VOLUME I

PERFORMANCE FLIGHT TESTING PHASE

CHAPTER 7
AERO PROPULSION

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7.1 INTRODUCTION

The steady progress of powered flight has closely followed the development of suitable aircraft powerplants. Unlike the question of the chicken and the egg, there is no doubt as to which was necessary first. Without a lightweight and yet adequately powerful engine, controlled flight of sufficient distance to serve a useful purpose would not be possible. Had it lacked an adequate means of propulsion, the machine conceived by Leonardo da Vinci could not have flown, even if it had been otherwise capable. Although Germany's Dr.N. A. Otto created the four-stroke internal combustion engine in 1876, it was not until twenty years later that Daimler was able to perfect the eight horsepower engine which enabled the Wolfert "Deutschland" to make the first gasoline powered dirigible flight. Wilbur and Orville Wright had to develop their own engine before they could achieve successful flight at Kitty Hawk in 1903. Later Glenn H. Curtiss met with outstanding success due largely to the engines which he was instrumental in developing. And so it has gone, down through the pages of aviation history; larger and more efficient engines lead to larger, faster, and higher flying aircraft.

7.2 THE FLIGHT SPECTRUM

The pros and cons of powerplant types for aircraft have been hotly debated since the earliest days of powered flight. The reciprocating engine,turboprop, turbofan, turbojet, ramjet, and the rocket each has its limitations as well as uses for which it is best suited.

The reciprocating engine, which has reached its ultimate size and horsepower, has long been with us as the workhorse of low and medium altitudes and airspeeds. The turboprop combines the advantage, inherent in propeller driven aircraft, of short takeoffs with the higher and faster flying capability of the gas turbine engine. The turbojet, with its increased efficiency at high altitudes and airspeeds, is ideal for high-flying, high performance military aircraft and fast, long-range airliners. The turbofan combines the advantages of both the turboprop and turbojet. It offers the high thrust at low airspeeds of the turboprop but without the heavy, complex reduction gearing and propeller, and improved fuel specifics at moderate airspeeds. On the horizon is yet a further advance, the propfan, which further combines turboprop and turbofan technology. A ramjet engine is particularly suited to high altitude and high speed, but it must be carried aloft by some means other than its own thrust to reach a velocity sufficient to allow the engine to start and operate.

Man is a creature who lives miles deep on the bottom of an ocean of air that forms a protective canopy over the surface of the earth. Place him in a vehicle a few miles above the bottom of his ocean, and he cannot survive unless some means are provided to duplicate, approximately, the air temperature and pressure of his normal environment. Above the altitude limitations of the human body, the vehicle must supply pressurized oxygen or air for its passengers and crew. Above the air limitations of the engine which propels it, the vehicle must carry all of its fuel and air (or other means of supporting combustion) with it, as is the case for the rocket.

Aircraft or missiles can be operated in continuous level flight only in a restricted area of the altitude flight speed spectrum. The minimum speed boundary of this level flight "corridor" is reached when the combined effect of wing lift and centrifugal force is no longer sufficient to support aircraft weight. Transient flight is possible at lower flight speeds by use of a ballistic-type flight path, where altitude is being varied throughout the flight, or by aircraft supported directly by powerplant thrust. Except at very high altitudes, the maximum speed for continuous flight occurs where the increase in aircraft and powerplant structural weight required to overcome the adverse effects of high ram air pressure and temperature becomes excessive. The effects of pressure predominate at low altitudes, whereas the rapid deterioration of the strength of structural materials at high temperatures is the primary factor at high altitudes. Development of better materials and improved construction techniques will tend to raise these maximum speed limits. At very high altitudes, the maximum speed for continuous level flight is limited to the orbiting velocity. Figure 7.1 shows the limits of the so-called continuous level-flight corridor.

7.3 PRINCIPLE OF JET PROPULSION

The principle of jet propulsion derives from an application of Newton's laws of motion. When a fluid is accelerated or given a momentum change, a force is required to produce this acceleration in the fluid, and, at the same time, there is an equal and opposite reaction force. This opposite reaction force of the fluid on the engine is called the thrust; therefore, the principle of jet propulsion is based on the reaction principle. A little thought will indicate that all devices or objects that move through fluids must follow this basic propulsion principle. The fish and human swimmer move themselves through the water by this principle, and, in the same manner, birds are able to propel themselves through the air. Even the reciprocating engine with its propeller (which causes a momentum change of air) obeys the same principle of momentum change.

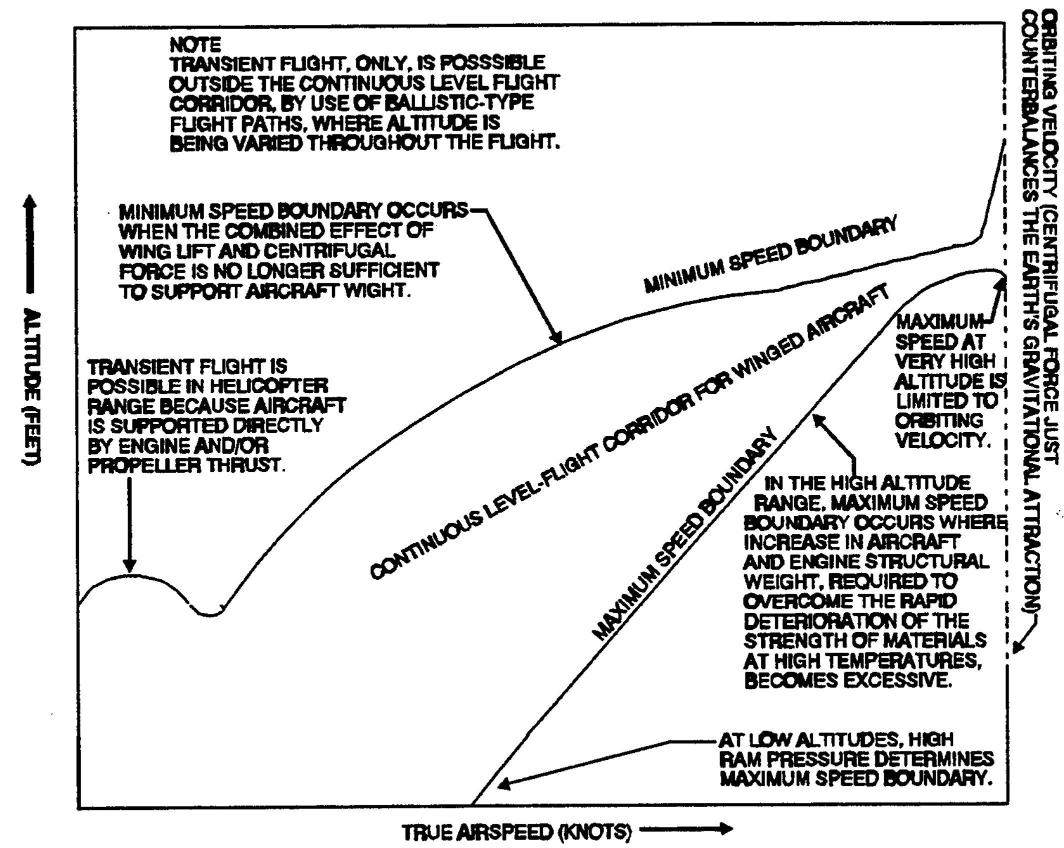


FIGURE 7.1. CONTINUOUS LEVEL FLIGHT CORRIDOR

Any fluid can be utilized to achieve the jet propulsion principle; thus, steam, combustion gases, or the hot gases generated by any heating process can be applied to propel a device through a fluid or space. Since many of these devices operate in the air, they change the momentum of the air for their propulsive thrust. These devices are called air-breathing engines because they utilize the air for their working fluid.

7.3.1 THE BASIC GAS TURBINE ENGINE

The gas turbine is an air-breathing engine. The term, "gas turbine," could be misleading because the word "gas" is so often used for gasoline. The name, however, means exactly what it says: a turbine type of engine which is operated by a gas, differentiated, for instance, from one operated by steam vapor or water. The gas which operates the turbine usually is the product of the combustion which takes place when a suitable fuel is burned with the air passing through the engine. In most gas turbines, the fuel is not gasoline at all, but rather, a low grade distillate such as JP-4 or commercial kerosene.

Both the reciprocating engine and the gas turbine develop power or thrust by burning a combustible mixture of fuel and air. Both convert the energy of the expanding gases into propulsive force. The reciprocating engine does this by changing the energy of combustion into mechanical energy which is used to turn a propeller. Aircraft propulsion is obtained as the propeller imparts a relatively small amount of acceleration to a large mass of air. The gas turbine, in its basic turbojet configuration, imparts a relatively large amount of acceleration to a smaller mass of air and thus produces thrust or propulsive force directly. Here, the similarity between the two types of engines ceases.

The reciprocating engine is a complicated machine when compared to the gas turbine. If only the basic, mechanically coupled compressor and turbine are considered, the gas turbine has only one major moving part. Air comes in through an opening in the front of the engine and goes out, greatly heated and accelerated, through an opening in the rear. Between the two openings, the engine develops thrust.

Fundamentally, a gas turbine engine may be considered as consisting of five main sections: an inlet, a compressor, a burner, a turbine, and a tailpipe having a jet nozzle. Turbojet versions of gas turbine engines are devices to generate pressures and gases which provide mass and acceleration.

Newton's Second Law states that a change in motion is proportional to the force applied. Expressed as an equation, force equals mass multiplied by acceleration (F = ma). Force is the net thrust. Acceleration is a rate of change of velocity; therefore, we can write

$$F=m\frac{dv}{dt} \tag{7.1}$$

The velocity change is between the low velocity of the incoming air, the zero velocity of the fuel, and the high velocity of the outgoing gases, all velocities being relative to that of the engine. Since momentum is defined as mass times velocity, when velocity changes are substituted in the equation in place of acceleration, the idea of momentum changes within the engine being equal to force or thrust can be understood.

Mass, in the case of the turbojet, is the mass of air plus the mass of fuel which pass through the engine. Acceleration of these masses is accomplished in two ways. First, the air mass is compressed, and pressure is built up as the air goes through the compressors with little change in velocity. Secondly, the fuel and part of the air are burned to produce heat. The heated gases expand in the burner section and accelerate through the turbine inlet nozzle at the outlet of the burner section. The turbines extract power to drive the compressors. This process decelerates the gases but leaves some pressure. The jet nozzle allows the gases to attain their final acceleration and generates the outgoing momentum.

The incoming momentum of the air and the zero momentum of the fuel entering the engine must be subtracted from the outgoing momentum of the gases in order to arrive at the overall change in momentum which represents thrust. The thrust developed by a turbojet engine, then, may be said to result from the unbalanced forces and momentums created within the engine itself. When the static pressure at the jet nozzle or the tailpipe exit exceeds the ambient outside air pressure, an additional amount of thrust is developed at this point. Figure 7.2 graphically represents the manner in which the internal pressures vary throughout the engine. These pressures and the areas on which they work are indicative of the momentum changes within the engine. Since engine pressure is proportional to engine thrust, Figure 7.2 indicates how the overall thrust produced by the engine is developed. The final unbalance of these pressures and areas gives, as a net result, the total thrust which the engine is developing. In practice, this unbalance may be measured or calculated in terms of pressure to enable the pilot to monitor engine thrust.

While turboprop engines function in a similar manner, the chief difference is that the jet thrust produced is held to a minimum. Their relatively large turbines are designed to extract all of the power possible from the expanding gases flowing from the burner section. This power is used to rotate the propeller which, in turn, accelerates a large mass of air to produce thrust to propel the aircraft.

7.4 ENGINE CLASSIFICATION

There are five basic air-breathing engines used for aircraft propulsion. These are the ramjet and the four basic gas turbine variants: turbojet, turboprop, turboshaft and turbofan.

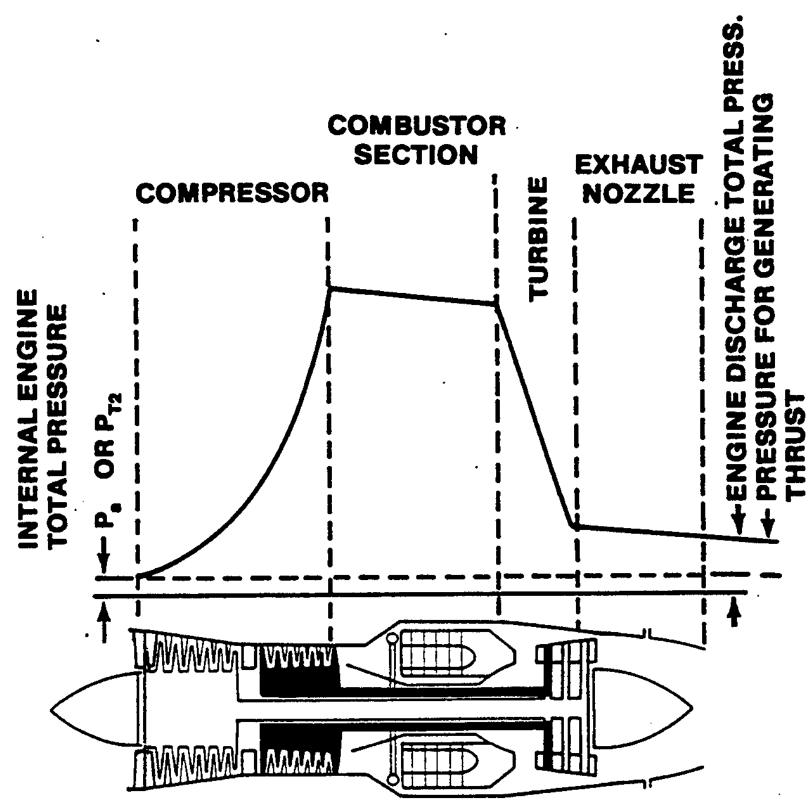


FIGURE 7.2. TYPICAL TURBOJET ENGINE INTERNAL PRESSURE VARIATIONS

7.4.1 THE RAMJET ENGINE

The simplest type of air-breathing engine is the ramjet engine, or, as it is sometimes called, the Athodyd (Aero-THermO-DYnamic-Duct) or Lorin engine (in honor of its original proponent).

This engine (Figure 7.3) consists of a diffuser, D, a combustion chamber, H, and a discharge nozzle, N. The function of a diffuser is to convert the kinetic energy of the entering air into a pressure rise by decreasing the air velocity. The diffuser delivers the air at a static pressure higher than atmospheric pressure to the combustion chamber, where fuel is mixed with the air and ignited.

The burning causes the specific volume of the air to increase; thus, the air is accelerated in the combustion chamber, where it burns at approximately constant

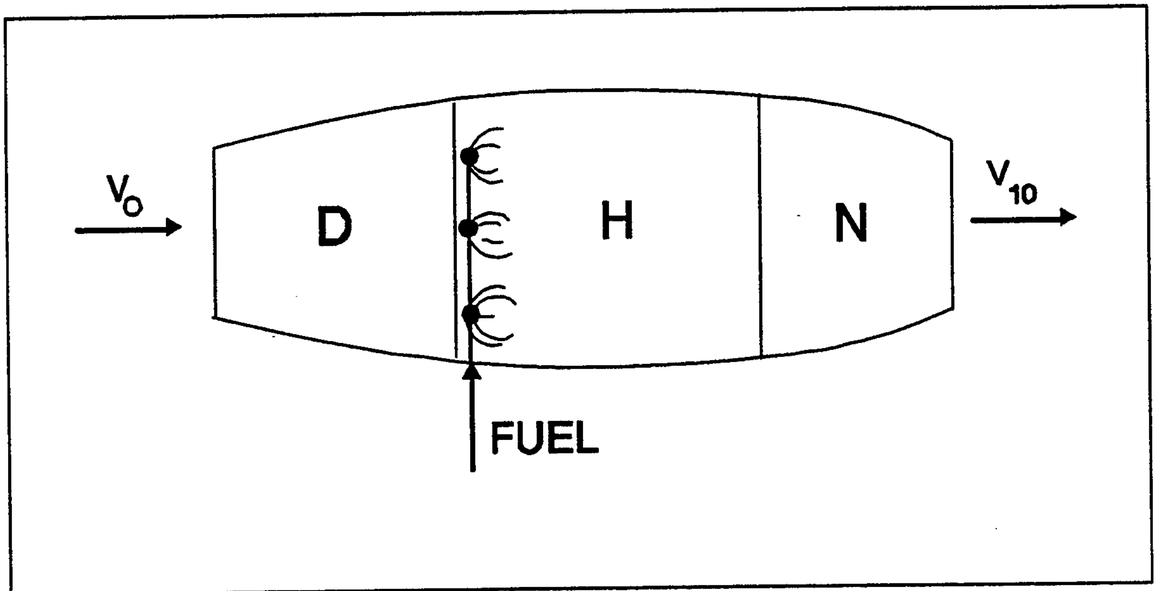


FIGURE 7.3. PRINCIPAL ELEMENTS OF A RAMJET ENGINE

pressure to a high temperature. The air temperature can also be raised by heat transfer from a heater such as a nuclear reactor. In this case, of course, the fuel consumption is effectively zero, since the required energy is derived from the nuclear fission in the reactor. Either way, high temperature and high pressure gases are delivered to the exhaust nozzle to produce an exit velocity greater than the entrance velocity.

Again, the process is one of changing the momentum of the working fluid from a low value at entrance to a high value at exit. The fuel used in this type of engine is usually a liquid hydrocarbon; however, solid fuels can be used to produce a propulsive thrust. Toward the end of World War II, the Germans were experimenting with ramjet engines which operated on coal and oil-cooked wood. It should be noted that the ramjet engine (in its basic form) cannot operate under static conditions, since there will be no pressure rise in the diffuser. Usually, a Mach of at least 0.2 is required for any operation at all, and performance improves as the flight speed is increased.

It is readily apparent why this engine is sometimes called the "flying stovepipe." An ignition system is required to start it. However, once started, the engine is a continuous firing duct in that it burns fuel at a steady rate and takes air in at a steady rate for any given flight velocity.

7.4.2 THE TURBOJET ENGINE

The ramjet engine is simple in construction; however, its application is limited, and to date it has not been used extensively. The most common type of air-breathing engine is the turbojet engine illustrated in Figure 7.4.

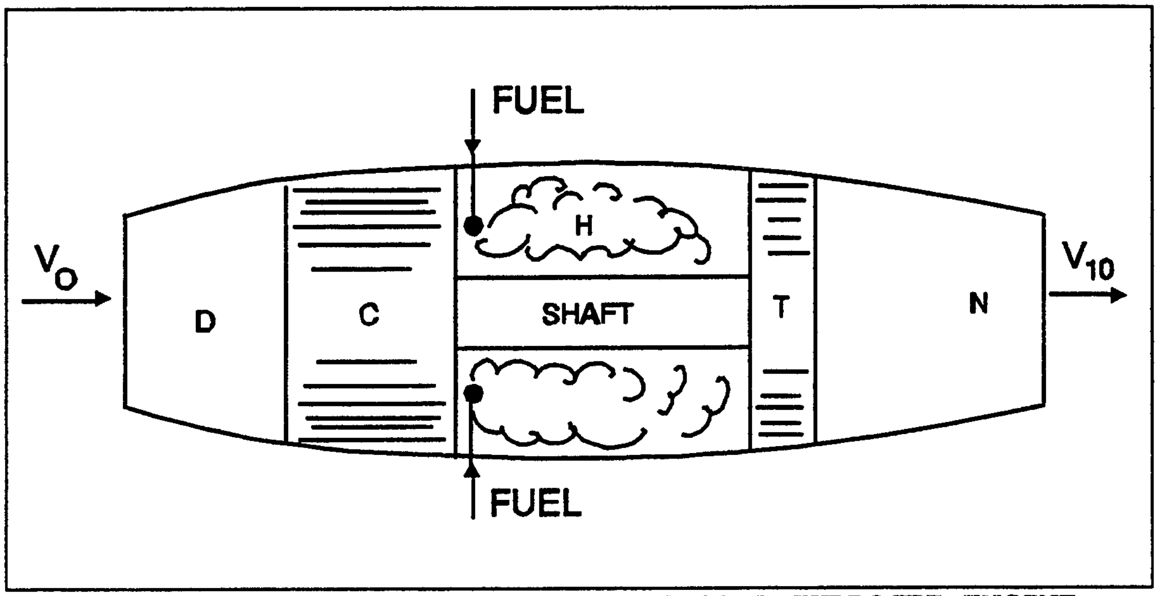


FIGURE 7.4. PRINCIPAL ELEMENTS OF A TURBOJET ENGINE

This engine consists of a diffuser, D, a mechanical compressor, C, a combustion chamber, H, a mechanical turbine, T, and an exhaust nozzle, N. Again, the function of the diffuser is to transform the kinetic energy of the entering air into a static pressure rise. The diffuser delivers its air to the mechanical compressor which further compresses the air and delivers it to the combustion chamber. There, fuel nozzles feed fuel continuously, and continuous combustion takes place at approximately constant pressure. Here also, the air temperature can be raised by heat transfer from a nuclear reactor. The high temperature and high pressure gases then enter the turbine, where they expand to provide driving power for the turbine. The turbine is directly connected to the compressor, and all the power developed by the turbine is absorbed by the compressor and the auxiliary apparatus. The main function of the turbine is to provide power for the mechanical compressor. After the gases leave the turbine, they expand further in the exhaust nozzle and are ejected with a velocity greater than the flight velocity to produce a thrust for propulsion. It is evident that this engine is not a great deal different from the ramjet engine. Here, a compressor and a turbine are used to provide the additional pressure rise which could not be obtained in a ramjet engine. Since this engine has a mechanical compressor, it is capable of operating under static conditions; however, increases in flight velocity improve its performance because of the benefit of ram pressure achieved by the diffuser. It is again pointed out that the overall pressure ratio of the cycle may be increased to a value greater than that which is possible in a ramjet engine. However, at very high flight speeds (Mach 3 or more), sufficient pressure rises can be obtained from the diffuser alone. Thus, at higher speeds, the ramjet engine may become more attractive than the turbojet engine.

Turbojet engines can be further classified by the type of compressor they employ. The centrifugal compressor works very well in the smaller turbojet and turboprop engines where a high compression ratio is not too essential. This design was standard for early aircraft gas turbines. Large, high performance engines require the greater efficiency and higher compression ratios attainable only with an axial flow type of compressor. Axial flow compressors have the added advantages of being lightweight and having a small frontal area. Either a single compressor (Figure 7.5a), a dual compressor (Figure 7.5b), or a triple-spool may be used. The latter types result in

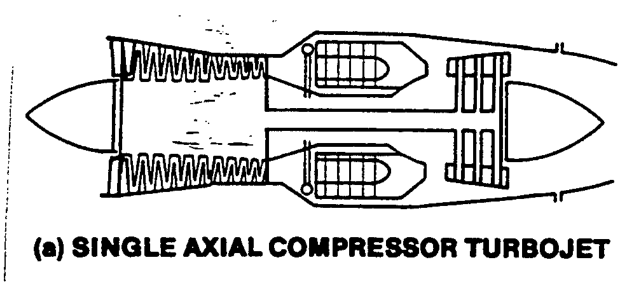


FIGURE 7.5A. SINGLE AXIAL COMPRESSOR TURBOJET

higher compressor efficiencies, compression ratios, and thrusts. In dual compressor engines, one turbine or set of turbine wheels drives the high pressure compressor, and another set drives the low pressure compressor. Both rotor systems operate independently of one another except for airflow. The turbine for the low pressure compressor, the rear turbine, is connected to its compressor by a shaft passing through the hollow center of the high pressure compressor and turbine assembly drive

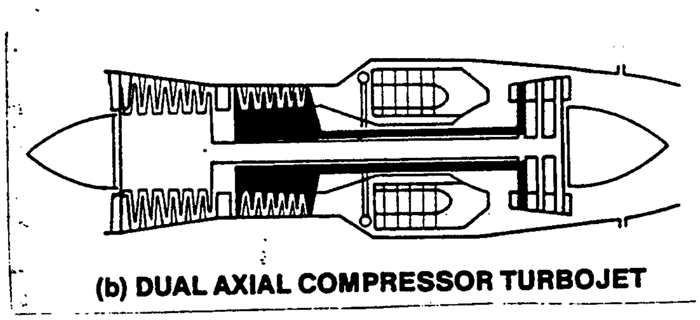


FIGURE 7.5B. DUAL AXIAL COMPRESSOR TURBOJET

shaft. The dual compressor configuration is often called a dual-rotor, two-spool, or twin-spool engine; the single compressor configuration is likewise called a single-rotor or single-spool engine.

Frequently, a turbojet engine is equipped with an afterburner for increased thrust (Figure 7.6). This increase in thrust can be accomplished regardless of the type of compressor used. Roughly, about 25% of the air entering the compressor and passing

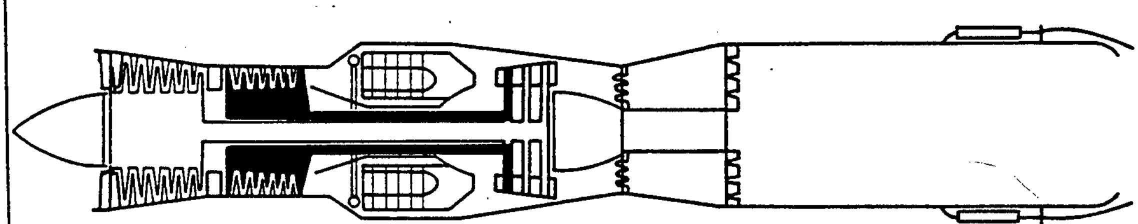


FIGURE 7.6. DUAL AXIAL COMPRESSOR TURBOJET WITH AFTERBURNER

through the engine is used for combustion. Only this amount of air is required to attain the maximum temperature that can be tolerated by the metal parts. The balance of the air is needed primarily for cooling purposes. Essentially, an afterburner is simply a huge stovepipe attached to the rear of the engine, through which all of the exhaust gases must pass. Fuel is injected into the forward section of the afterburner and is ignited. Combustion is possible because 75% of the air which originally entered the engine still remains unburned. The result is, in effect, a tremendous blowtorch which increases the total thrust produced by the engine by approximately 50% or more. Although the total fuel consumption increases two to ten times, the net increase in thrust is profitable for takeoff, climb, or acceleration. A turbojet aircraft with an afterburner can often reach a given altitude with the use of less fuel by climbing rapidly in afterburner than by climbing more slowly without the afterburner. The weight and noise of an afterburner, which is used only occasionally, precludes the device being employed on present day, transport type aircraft; however, afterburners are used to maintain cruise Mach on the SST.

7.4.3 THE TURBOPROP OR TURBOSHAFT ENGINE

In principle, this engine (Figure 7.7) is very similar to the turbojet engine, differing only in that it uses a propeller to provide most of the propulsive thrust.

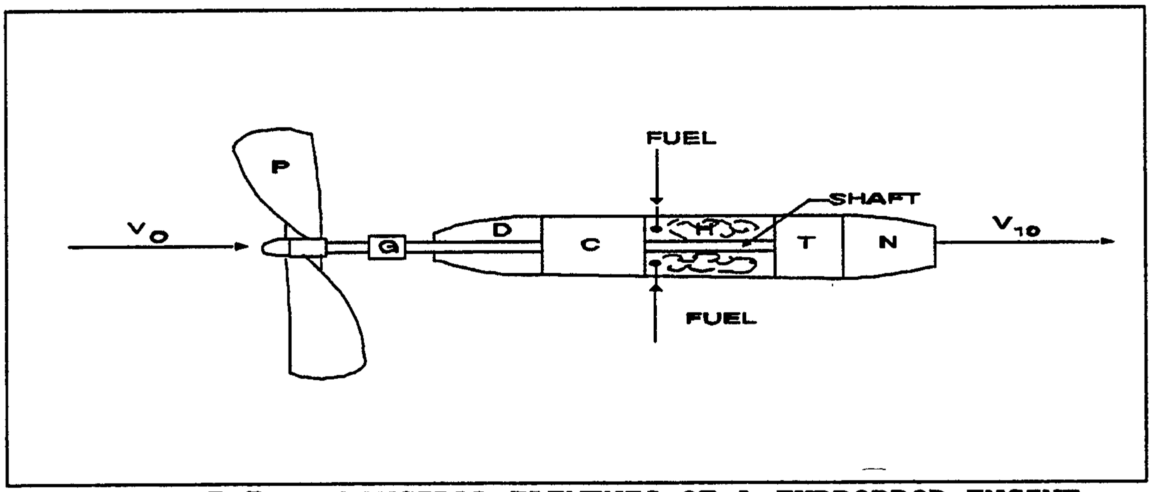
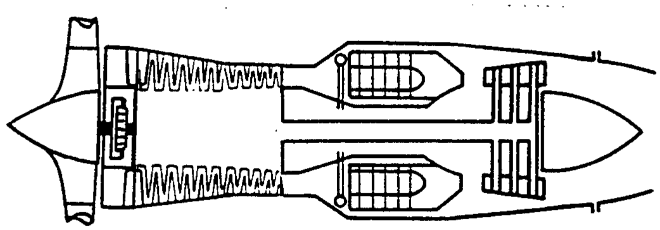


FIGURE 7.7. PRINCIPAL ELEMENTS OF A TURBOPROP ENGINE

The engine consists of a diffuser, D, a mechanical compressor, C, a combustion chamber, H, a turbine, T, an exhaust nozzle, N, reduction gearing, G, and a propeller, P. The diffuser, mechanical compressor, and combustion chamber function in the same manner as in the turbojet engine. However, in the turboprop engine, the turbine extracts much more power than it does in the turbojet engine because the turbine provides power for both the compressor and the propeller. When all of this energy is extracted from the high temperature gases, there is little energy left for producing jet thrust. Thus, the turboprop engine derives most of its propulsive thrust from the propeller and derives only a small portion (10 to 25% depending on the flight velocity) from the exhaust nozzle. Since the shaft rotation speed of gas turbine engines is very high (approximately 12,000 RPM), reduction gearing must be placed between the turbine shaft and the propeller to enable the propeller to operate efficiently. The turboprop engine is essentially a gas turbine power plant because, as pointed out before, little power is derived from the exhaust nozzle; still, as flight speeds are increased, the ratio of jet thrust to propeller thrust for maximum thrust tends to become higher. The propulsive thrust is provided by a dual momentum change of the air. First, the propeller increases the air momentum, and second, the overall engine, from diffuser to nozzle, provides an internal momentum increase. The sum of these two thrusts is the total thrust developed by the engine.

The conversion to a turboprop can be accomplished with either a single or multistage centrifugal compressor, a single axial compressor, or a dual axial compressor. In most cases, the propeller reduction drive gearing is connected directly to the compressor drive shaft (Figure 7.8a) or, when a dual axial compressor is used, to the low pressure compressor drive shaft (Figure 7.8b). On still another type, the propeller is driven independently of the compressor by a free turbine of its own (Figure 7.9).



(a) SINGLE AXIAL COMPRESSOR, DIRECT PROPELLER DRIVE TURBOPROP

FIGURE 7.8A. SINGLE AXIAL COMPRESSOR DIRECT PROPELLER DRIVE TURBOPROP

In one version of the free turbine turboprop, both an axial and a centrifugal compressor are used. A single stage turbine, operating by itself, supplies the power to drive both the compressors and the accessories. If a turbine of a gas turbine engine is connected to a drive shaft which, in addition to the compressor, drives something other than a propeller, the engine is referred to as a shaft turbine or turboshaft engine. Turboshaft engines are most often used to power helicopters.

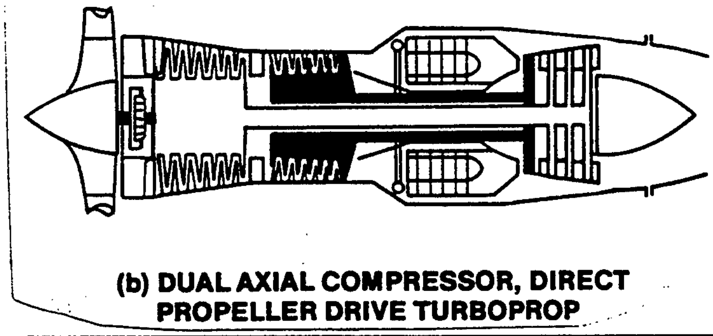


FIGURE 7.8B. DUAL AXIAL COMPRESSOR: TURBOPROP

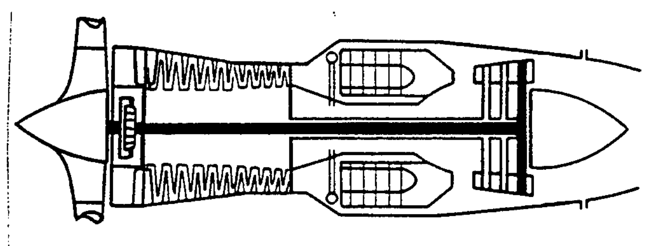


FIGURE 7.9. SINGLE AXIAL COMPRESSOR: FREE TURBINE PROPELLER DRIVE TURBOPROP

7.4.4 THE TURBOFAN ENGINE

The turbofan engine combines features of both the turbojet and turboprop engines. As a result, it has performance characteristics somewhere between the other two engines. Figure 7.10 schematically illustrates the principal elements of a front fan version of the turbofan engine.

The engine consists of a diffuser, D, a front fan, F, a mechanical compressor, C, a combustion chamber, H, a turbine, T, a bypass duct, B, and an exhaust nozzle or nozzles, N. As before, the function of the diffuser is to convert the kinetic energy of the entering air into a static pressure rise. The diffuser delivers its air to a fan, which further compresses it a small amount (a pressure ratio of approximately 1.5 to 2.0). The airflow is then split, and a portion enters the bypass duct, while the remainder continues into the mechanical compressor, combustion chamber, and turbine. The ratio of the airflow through the bypass duct to the airflow through the gas generator

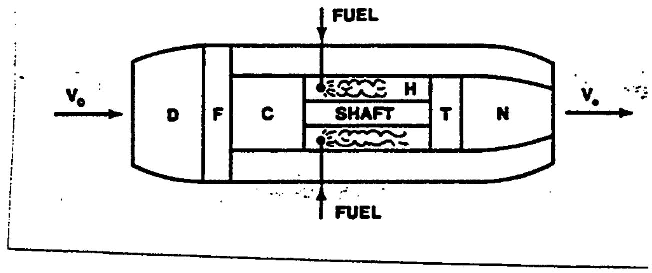


FIGURE 7.10A. PRINCIPAL ELEMENTS OF A TURBOFAN ENGINE (FRONT FAN)

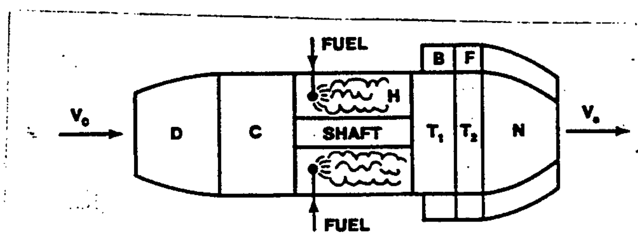


FIGURE 7.10B. PRINCIPAL ELEMENTS OF A TURBOFAN ENGINE (AFT FAN)

is defined as the bypass ratio. The turbine, as with the turboprop engine, provides the power for both the fan and the compressor. Unlike the turboprop engine, however, there is still considerable energy available in the gases downstream of the turbine. The exhaust gases are, therefore, further expanded in the exhaust nozzle to a velocity greater than the flight velocity, producing thrust for propulsion. The bypass air is also expanded, either through a common nozzle with the exhaust gases or through a separate nozzle, to a velocity higher than the flight velocity, producing additional thrust for propulsion. The turbofan engine thus derives its propulsive thrust from the high velocity exhausts of both the bypass air and the gas generator gases.

The version of the turbofan engine illustrated in Figure 7.10b differs from the front fan version in that the fan, F, is located aft of the gas generator turbine, T_1, and is driven by a separate turbine, T_2. Only bypass air, which can enter a common diffuser, D, or a separate diffuser, B, passes through the fan. However, the propulsive thrust of the engine is still derived from the high velocity exhaust of both the fan and the gas generator.

Although these are the two basic configurations of the turbofan engine, many variations are possible. Three different configurations of actual engines are illustrated in Figure 7.11.

As compared to the turbojet and turboprop engines, the turbofan engine derives its thrust from the acceleration of a medium amount of air through a medium velocity increment. The turbojet accelerates a small amount of air through a large velocity increment; the turboprop accelerates a large amount of air (through the propeller) through a small velocity increment.

As with the turbojet engine, significant thrust augmentation is also possible with the turbofan engine. Afterburning can be accomplished in either or both of the exhaust streams. In fact, since the bypass stream has no combustion products, very large temperature increases and, hence, exhaust velocity or thrust increases are possible with the turbofan engine.

7.5 THRUST

One speaks of horsepower when describing a reciprocating engine or a turboprop. Power is defined as work per unit of time, and work involves a force operating over a distance. Expressed as an equation

$$P = \frac{Fs}{t} \tag{7.2}$$

where: P = Power
F = Force
s = Distance
t = Time

One horsepower is the unit used to describe the equivalent of 33,000 foot-pounds of work performed in one minute, or 550 foot-pounds of work in one second. In a reciprocating engine or turboprop, it is possible to measure distance and time. Torque and RPM are used in computing horsepower. However, these same distance and time elements make the use of the terms "power" and "horsepower" unacceptable for a turbojet engine. When a turbojet engine is static, as in the case of an aircraft parked

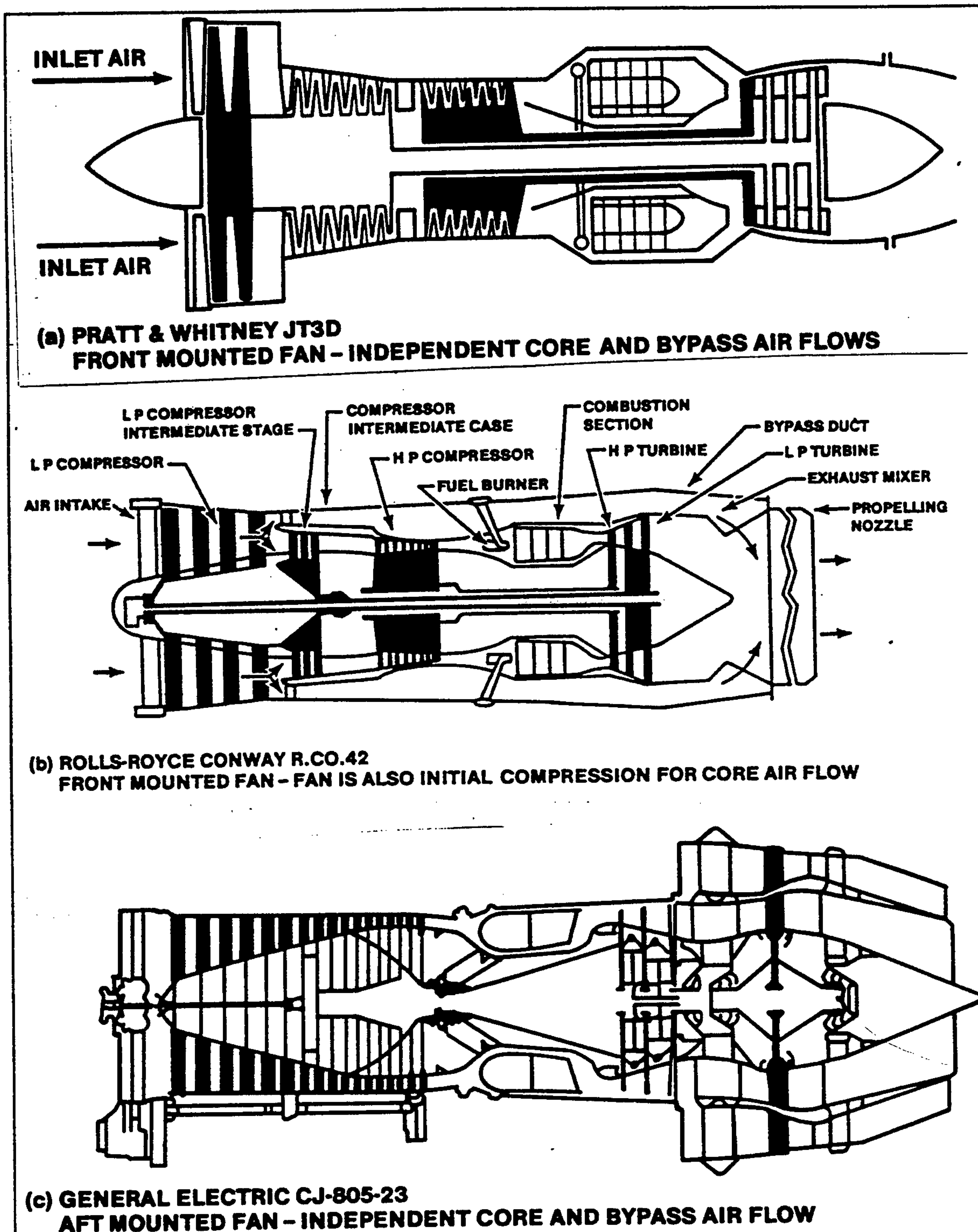


FIGURE 7.11. SCHEMATIC DIAGRAMS OF TURBOFAN ENGINES.

on the ground or when an engine is mounted in a ground test stand, distance and time are zero because no movement is involved that can be measured against a period of time. Although torque and RPM are produced by the turbine, the horsepower developed is used entirely within the engine itself. According to the definition and equation for power, none is being produced; yet, a forward force is being exerted when the engine is operating. It might be said that thrust is the measurement of the amount that an engine pushes against its attachment points. The propulsive force developed by a turbojet is measured in pounds of thrust.

In order to evaluate various propulsive devices and provide a basis for comparison, we will write an expression which gives a value for thrust. Consider, for example, an air-breathing engine that uses $\dot{m}$ slugs/sec or lb sec/ft of air per second, as shown

$$\dot{m} = \frac{\dot{w}\ lb/sec}{g\ ft/sec^2} \tag{7.3}$$

in Figure 7.12.

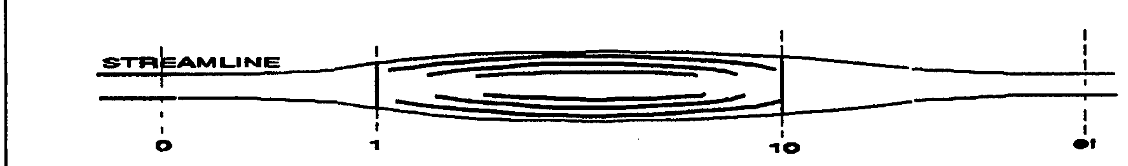


FIGURE 7.12. AIR-BREATHING ENGINE

We consider the air as it flows between the streamlines from entrance to exit as illustrated. All air-breathing engines take in air at approximately flight velocity and atmospheric pressure (in the absence of shock waves), compress it by some means, heat it by combustion, and discharge it through a nozzle so as to increase the momentum of the exit gases. The subscripts in Figure 7.12 have the following meaning: 0 refers to free stream conditions, 1 refers to the engine inlet section, 10 refers to the engine exit section, and ef refers to the section where the pressure of the engine exhaust gases is first equal to the pressure of the surrounding atmosphere. The thrust of such a device is given by the time rate of momentum change between sections where the pressure is equal (0 and ef). We can write the following expression for the net thrust acting on the engine.

$$F=ma=m\left(\frac{dv}{dt}\right)=\frac{m}{dt}\left(V_{ef}-V_o\right) \tag{7.4}$$

or

$$F=\dot{m}\left(V_{ef}-V_0\right) \tag{7.5}$$

To account for the change of mass flow due to the addition of fuel we must write

$$F=\dot{m}_{10}V_{ef}-\dot{m}_0V_0 \tag{7.6}$$

In most air-breathing engines, this addition of fuel is small (about 2%); however, to be analytically correct, we shall consider this factor in our thrust equation. When thrust is evaluated, measurements are usually made at the actual engine exit section and not at Section ef. Therefore, it is desirable to write the first term of Equation 7.6 in terms of conditions at Station 10. The pressure at 10 can be greater or less than the atmospheric pressure, and when this is true, the pressure unbalance will provide an additional force term to the thrust equation. When the thrust equation is written between the free stream condition 0 and the actual engine exit Station 10, it becomes

$$F_{actual}=\dot{m}_{10}V_{10}+A_{10}\left(P_{10}-P_0\right)-\dot{m}_0V_0 \tag{7.7}$$

Equation 7.6 or Equation 7.7 may be used to evaluate the net thrust of a propulsion device, and it has been found from flight measurements that either equation will give satisfactory results. The various terms contained in these equations are given specific names. First, there is the gross thrust, the thrust produced by the nozzle, which is defined as

$$F_{g_{ideal}}=\dot{m}_{10}V_{ef} \tag{7.8}$$

$$F_{g_{actual}}=\dot{m}_{10}V_{10}+A_{10}(P_{10}-P_0) \tag{7.8a}$$

Note that these are two forms for the gross thrust. The first is the momentum flux at the effective exit section, and the second is the sum of the momentum flux and the pressure thrust at the exit section. The latter form is the one preferred.

The other term of the thrust equation is called the negative thrust or ram drag. It is defined as

$$F_r=\dot{m}_o V_o \tag{7.9}$$

This force is a negative one because it represents the equivalent drag of taking on the flight-velocity air. The difference between these two terms (the gross thrust and the ram drag) is called the net thrust because it is the net force acting on the engine to produce propulsion power. Thus, we can write for net thrust

$$F_n=F_{g_{actual}}-F_r=\dot{m}_{10}V_{10}+A_{10}(P_{10}-P_0)-\dot{m}_o V_0 \tag{7.10}$$

or

$$F_n=\dot{m}(V_{10}-V_0)+A_{10}(P_{10}-P_0)$$

neglecting fuel added.

When

$$P_{10}=P_0$$

i.e., for an ideal nozzle

$$F_n = \dot{m}(V_{10} - V_0) \tag{7.11}$$

Sometimes it is more convenient, when evaluating thrust, to express the momentum flux as a function of Mach rather than velocity. This relation was derived from the continuity equation and from the definition of Mach.

$$\frac{\dot{m}V}{A} = P\gamma M^2 \tag{7.12}$$

The various forms of the thrust equation are summarized in Table 7.1.

TABLE 7.1
SUMMARY OF THRUST EQUATIONS

Gross Thrust

$$F_g = \dot{m}_{10} V_{10} + A_{10}(P_{10} - P_0)$$

$$F_g = A_{10} \; (P_{10}(\gamma_{10} M_{10}^{\;2} + 1) - P_0)$$

Ram Drag

$$F_r = \dot{m}_0 V_0 = A_0 P_0 \gamma_0 M_0^{\;2}$$

Net Thrust

$$F_n = F_g - F_r$$

It should be remembered that the net thrust is always the difference between the gross thrust and the ram drag; therefore, it is given by any combination of the various gross thrust and ram drag terms.

When the aircraft and engine are static, net thrust and gross thrust are equal. When the term, "thrust," is used by itself in discussing a gas turbine engine, the reference is usually to net thrust, unless otherwise stated.

Static engine thrust is measured directly in an engine test stand. Stands are usually constructed is such a manner that they float, pushing against a calibrated scale which accurately measures the thrust in pounds. Thrust stands are also available to measure the static thrust exerted by a complete aircraft and engine installation and are often used, although some additional complications are involved. Once an installed engine becomes airborne, direct measurement of thrust is not usually practical. Consequently, compressor RPM and turbine discharge pressure (or engine pressure ratio), that vary with the thrust being developed, are measured and used to indicate the propulsive force which an engine is producing in flight.

7.6 FACTORS AFFECTING THRUST

If a turbojet engine were operated only under static conditions in an air-conditioned room at standard day temperature, there would be no need to change the quantities used in the foregoing equations for net and gross thrust at any given throttle setting. However, all engines installed in aircraft must operate under varying conditions of airspeed and altitude. These varying conditions will radically affect the temperature and pressure of the air entering the engine, the amount of airflow through the engine, and the jet velocity at the engine exhaust nozzle. This means that, for any given throttle setting, different values must be entered in the thrust equations as the airspeed and/or altitude of the aircraft changes. Although some of these variables are compensated by the engine fuel control, many of the changes that will occur affect the thrust output of the engine directly. In actual practice, the equation presented previously will seldom be used directly to calculate engine thrust. Nevertheless, an understanding of the effect on the thrust equations of the several variables that will be encountered during normal engine operation will serve to illustrate how the changing conditions at the engine air inlet affect engine performance in flight and on the ground.

7.6.1 RAM EFFECT

As an aircraft gains speed going down a runway, the outside air is moving past the aircraft with increasing speed. The effect is the same as if the aircraft were stationary in a wind tunnel and air were being blown past the aircraft by means of a fan in the tunnel. The movement of the aircraft relative to the outside air causes air to be rammed into the engine inlet duct. Ram effect increases the airflow to the engine, which, in turn, means more thrust.

Ram effect alone, however, is not all that happens at the engine air inlet as airspeed increases. There are some changes in pressure and velocity which occur inside the air inlet duct because of the shape of the duct itself, as will be explained later. Neglecting these changes for the moment, it has been shown that, as an aircraft gains airspeed, the thrust being produced by the engine decreases for any given throttle setting because V_0 at the engine air inlet is increasing. Yet, because of ram effect, increasing the airspeed also increases the pressure of the airflow into the engine ($\dot{m}_a$).

What actually takes place, therefore, is the net result of these two different effects, as illustrated in Figure 7.13. In the sketch, the "A" curve represents the tendency of thrust to drop off as airspeed builds up, due to the increase in free stream velocity, V_0. The "B" curve represents the thrust generated by the ram effect that increases the airflow, $\dot{m}_a$, and, consequently, increases the thrust. The "C" curve is the result of combining curves "A" and "B". Notice that the increase in thrust due to ram as the aircraft goes faster and faster, eventually becomes sufficient to make up the loss in thrust caused by the increase in V_0. Ram will also compensate for some of the loss in thrust due to the reduced pressure at high altitude.

Ram effect is important, particularly in high speed aircraft, because eventually, when the airspeed becomes high enough, the ram effect will produce a significant overall increase in engine thrust. At the subsonic speeds at which aircraft powered by nonafterburning engines usually cruise, ram effect does not greatly affect engine thrust. At supersonic speeds, ram effect can be a major factor in determining how much thrust an engine will produce.

7.6.2 ALTITUDE EFFECT

The effect of altitude on thrust is really a function of density. As an aircraft gains altitude, the pressure of the outside air decreases, and the temperature of the air will, in general, become colder (Figure 7.14). As the pressure decreases, so does the thrust, but as the temperature decreases, the thrust increases. However, the pressure of the

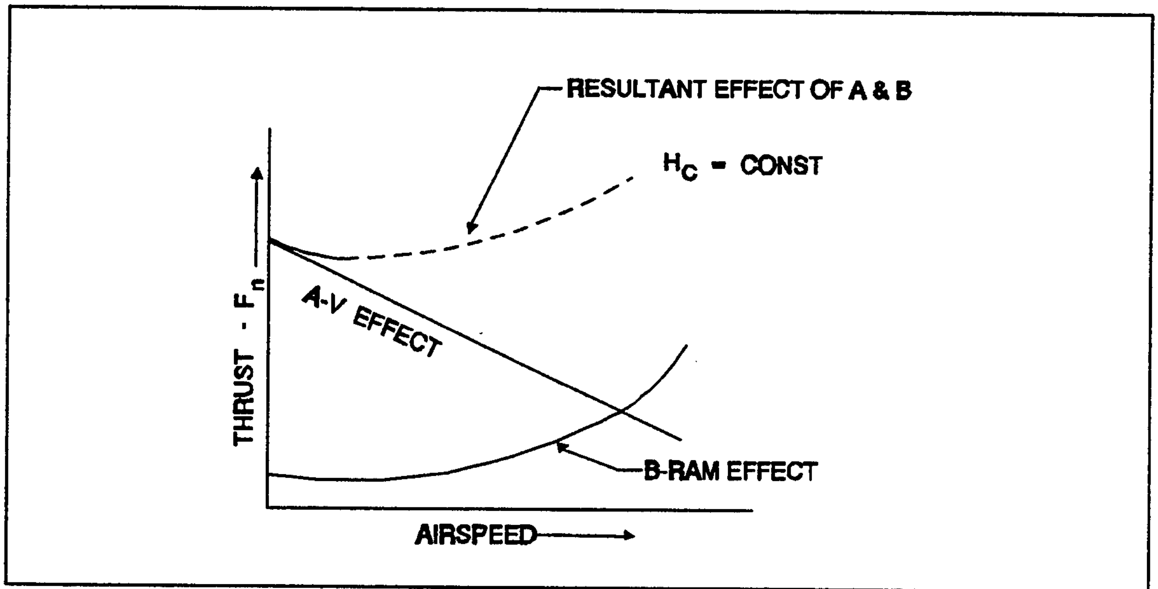


FIGURE 7.13. EFFECT OF RAM PRESSURE ON THRUST

outside air decreases faster than the temperature, so an engine actually produces less thrust as altitude is increased. The temperature becomes constant at about 36,000 feet. But the ambient pressure continues to drop steadily with increasing altitude. Because of this, thrust will drop off more rapidly above 36,000 feet.

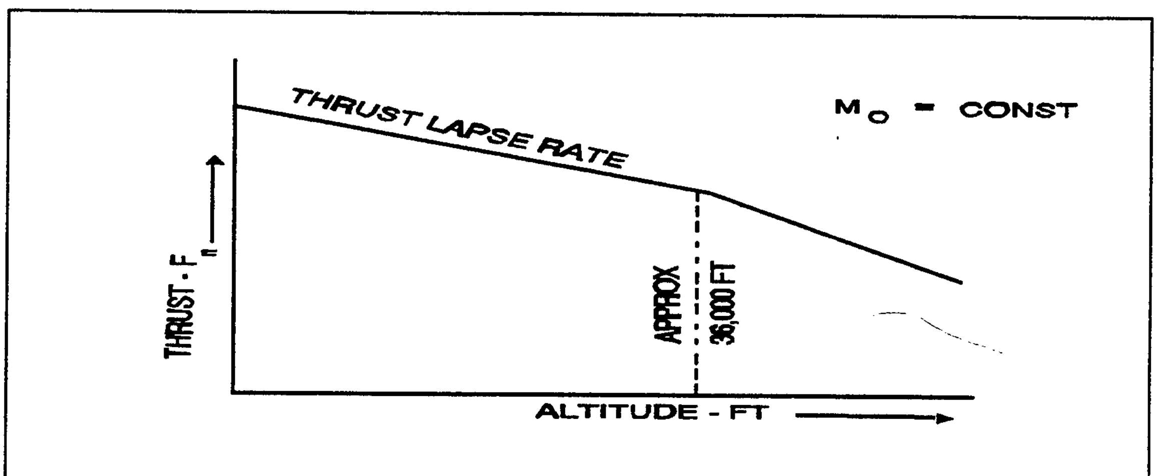


FIGURE 7.14. EFFECT OF ALTITUDE ON THRUST

7.7 SIMPLE CYCLE ANALYSIS

The thermodynamic cycle of the jet engine will be examined in order to obtain an insight into the factors affecting performance. An ideal cycle analysis of the turbojet and turbofan engine will be presented with a number of assumptions that will make the analysis simpler and easier to understand. Although the approach may appear somewhat restrictive, the results will be surprisingly close to those of the actual engine.

7.7.1 ENGINE STATION DESIGNATIONS

Figure 7.15 shows the engine station terminology that will be used throughout this chapter. This designation is normally used for a single-spool (single compressor-single turbine) turbojet engine. The system can be expanded to include dual axial compressors and turbines by adding Station Number 2.5 and 4.5 between the low and high pressure compressor and turbine respectively. Afterburner mechanization is designated by Station Numbers 6 to 9, as required.

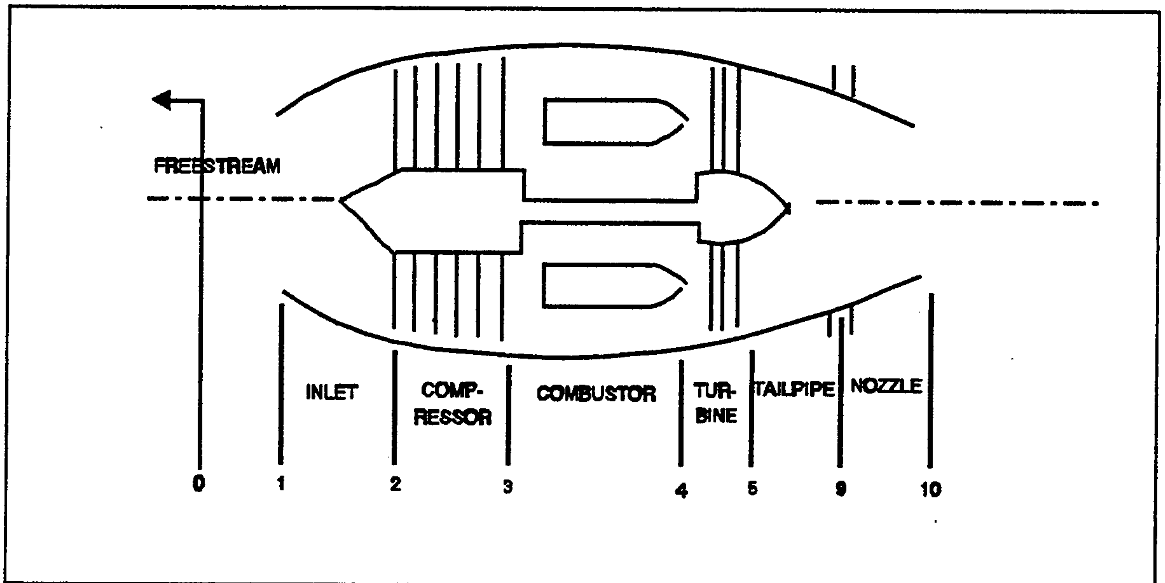


FIGURE 7.15. SINGLE-SPOOL TURBOJET ENGINE STATION DESIGNATIONS

7.7.2 BASIC EQUATIONS AND PROCESSES

The steady flow energy equation, Equation 7.13, will be the primary relationship used throughout the analysis.

$$\Delta Q - \Delta w = \Delta h_T \tag{7.13}$$

where

ΔQ is the heat energy added to the cycle less the heat energy rejected,

ΔW, net work output of the cycle,
ΔhT, net change in total enthalpy.

Enthalpy is a convenient term used in flow analysis because it includes not only the internal energy of the working gas but also the flow and expansion work potential. Total enthalpy is composed of a static term related to absolute temperature and a kinetic term resulting from the velocity of the gas

$$h_T = h + \frac{V^2}{2gJ} \tag{7.14}$$

$$h = C_p T \tag{7.15}$$

The term gJ = 25,050 F-lb/BTU is a conversion factor to keep the equations in standard heat engine units. The specific heat at constant pressure (C_p) is a function of temperature, varying from 0.24 to 0.27 BTU/lb°R within a typical cycle.

The function of each engine component along with the appropriate form of Equation 7.13 is listed in Table 7.2. All processes in an ideal cycle are reversible, meaning there are no friction losses. In addition, all ideal processes except for combustion are isentropic. Isentropic means that entropy does not change during the process.

Entropy can be defined in several contexts, but in general, most definitions seem to be rather abstract. Although a thorough understanding of entropy is not required to comprehend the thermodynamic cycle, the basic concept is useful in understanding the limits of any heat engine. Entropy is a measure of the relative amount of heat energy that can be converted into mechanical energy, the remaining heat being rejected as lost energy. The Second Law of Thermodynamics gives some insight into the relative amount of energy which can be converted and the efficiency of the process. A process can be isentropic only if there is no heat transfer. Consequently, a combustion process can never be isentropic.

TABLE 7.2
IDEAL NON-AFTERBURNING TURBOJET COMPONENT
PROCESS AND EQUATIONS

STATION #	NAME	PURPOSE	IDEAL PRO-CESS	IDEAL EQUATION
0	Free Stream			
0-1	Aerodynamic Inlet	Accel or Decel Air to Velocity at Inlet Face	Isentropic No Work	$h_{T0}=h_{TI}=h_0+\frac{V_0 2}{2gJ}$
1	Inlet Face			
1-2	Geometric Inlet	Accel or Decel Air to Vel Req'd by Comp	Isentropic No Work	$h_{TI}=h_{T2}=h_{T0}$
2	Comp Face			
2-3	Compressor	Incr Total Pressure of Airflow Absorb Work of the Turbine	Isentropic with Work	$h_{T2} + W_c = h_{T3}$

3	Comb Face			
3-4	Combustor	Incr Total Energy of Flow	Const. Pres Combus-tion	$h_{T3} + Q_{IN} = h_{T4}$
4	Turb Inlet			
4-5	Turbine	Extract Work to Drive Compressor	Isentropic with Work	$h_{T4} = h_{T5} + W_T$
5	Tailpipe Entry			
5-9	Tailpipe	Deliver Gas to Nozzle	Isentropic No Work	$h_{T5} = h_{T9}$
9	Noz Entr			
9-10	Nozzle Discharge	Incr Kinetic Engy of Gas	Isentropic Expan-sion No Work	$h_{T5} = h_{T10}$ $V_{10}=\sqrt{2gJ(h_{T5}-h_{10})}$

7.7.3 THE IDEAL CYCLE

A thermodynamic cycle is a series of processes that are repeated in a given order. The working fluid passes through various state changes, returning periodically to the initial state. An ideal cycle is one composed entirely of reversible processes.

The cycle can be constructed with any two independent variables, but a plot of enthalpy versus entropy is most useful. A typical h-s diagram for air is shown in Figure 7.16. Enthalpy and temperature are related by Equation 7.15; however, note that on the diagram the temperature variations of C_p have been included. The lines of constant pressure are given by the equation

$$ds = C_p \ln dT \ (P = \text{constant}) \qquad (7.16)$$

The ideal cycle for a turbojet engine is easily constructed using the equations from Table 7.2. A typical cycle is shown in Figure 7.17. The ideal cycle consists of the following processes in which the working gas is assumed to have negligible velocity at the compressor and turbine inlet and exit:

- 0-3 Air is compressed adiabatically
- 3-4 Air is heated at constant pressure
- 4-10 Gas is expanded isentropically
- 10-0 Gas is cooled at constant pressure within the atmosphere

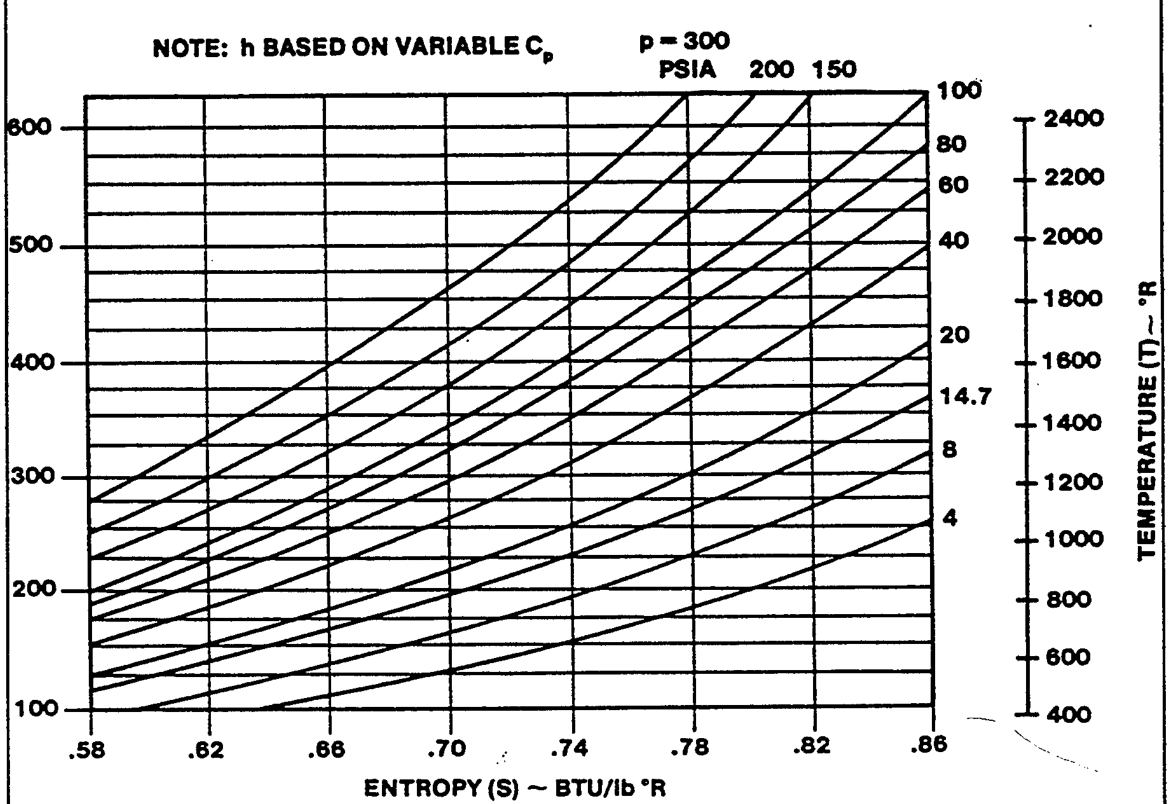


FIGURE 7.16. h-s DIAGRAM FOR AIR

The energy relationships which follow directly from Equation 7.14 are:

Compressor work, $W_c = h_{T3} - h_{T2} = C_p (T_{T3} - T_{T2})$ (7.17)

Turbine work, $W_T = h_{T4} - h_{T5} = C_p (T_{T4} - T_{T5})$ (7.18)

Net work out, $W_N = h_{T5} - h_{10} = C_p (T_{T5} - T_{10})$ (7.19)

Heat added, $Q_{IN} = h_{T4} - h_{T3} = C_p (T_{T4} - T_{T3})$ (7.20)

Heat rejected, $Q_{REJ} = h_{10} - h_0 = C_p (T_{10} - T_0)$ (7.21)

An energy balance of the cycle yields

$$W_c + Q_{IN} = W_T + W_N + Q_{REJ} \tag{7.22}$$

The work done by the turbine is equal to the work required by the compressor in the ideal cycle: $W_c = W_T$. Rearranging Equation (7.22), the net work out is then

$$\begin{aligned} W_N &= Q_{IN} - Q_{REJ} \\ &= (H_{T4} - H_{T3}) - (H_{10} - H_0) \\ &= C_P (T_{T4} - T_{T3} - T_{10} + T_0 \end{aligned} \tag{7.23}$$

7.7.3.1 NOTE ON TEMPERATURE MEASUREMENT

Equation 7.23 suggests that the output energy of a turbojet engine could be calculated by measuring the turbine inlet temperature (TIT = T_{T4}), compressor exit temperature (T_{T3}), nozzle exit temperature (T_{10}), and the ambient free stream temperature (T_0). The net work output could then be easily calculated with a simple calculator. This is in fact done for some engines. However, TIT is very difficult to measure due to temperatures sometimes in excess of 2400°R.

Another approach follows directly from the ideal relationship $W_c = W_T$. Substituting Equations 7.17 and 7.18

$$C_P (T_{T3} - T_{T2}) = C_P (T_{T4} - T_{T5})$$

Rearranging

$$T_{T4} - T_{T3} = T_{T5} - T_{T2}$$

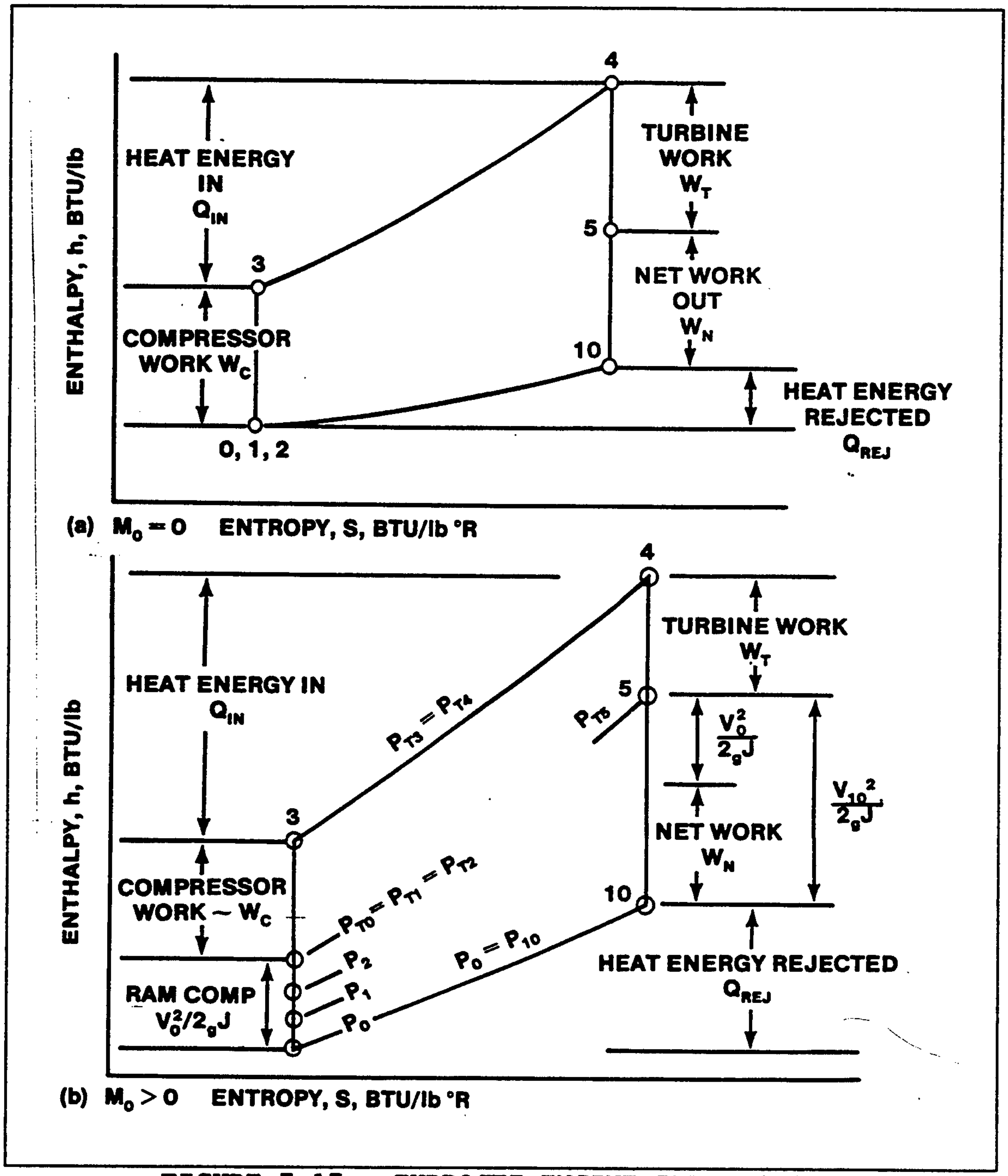


FIGURE 7.17. TURBOJET ENGINE IDEAL CYCLE

Substituting into Equation 7.23

$$W_N = C_P (T_{T5} - T_{T2} + T_0 - T_{10}) \tag{7.24}$$

where

T_{T5} = EGT, exhaust gas temperature, and

T_{T2} = CIT, compressor inlet temperature.

Since EGT is considerably lower then TIT, this method is more easily applied in practice and more often used. However, it is not as accurate as the first method because of the assumption $W_c = W_T$.

7.7.4 THERMAL EFFICIENCY

Thermal efficiency is a measure of how efficiently heat energy can be converted into net work. By definition,

$$\eta_{TH} = \frac{W_N}{Q_{IN}}$$

$$= \frac{C_P(T_{T4} - T_{T3} - T_{10} + T_0)}{C_P(T_{T4} - T_{T3})}$$

$$\eta_{TH} = 1 - \frac{T_{10} - T_0}{T_{T4} - T_{T3}} \tag{7.25}$$

Equation 7.25 is not very transparent in terms of engine design parameters. From the relationship for an ideal gas undergoing an isentropic expansion,

$$\left[\frac{P_1}{P_2}\right]^{\frac{\gamma-1}{\gamma}} = \frac{T_1}{T_2} \tag{7.26}$$

we can write

$$\left[\frac{P_{T3}}{P_0}\right]\left[\frac{P_{10}}{P_{T4}}\right]^{\frac{\gamma-1}{\gamma}} = \left[\frac{T_{T3}}{T_0}\right]\left[\frac{T_{10}}{T_{T4}}\right] \tag{7.27}$$

In the ideal cycle $P_{10} = P_0$ and $P_{T4} = P_{T3}$ so the right side of Equation 7.27 is unity. Hence,

$$\left[\frac{T_{T3}}{T_0}\right]\left[\frac{T_{10}}{T_{T4}}\right]=1 \quad or \quad \frac{T_{10}}{T_0}=\frac{T_{T4}}{T_{T3}} \tag{7.28}$$

PROBLEM: Using Equation 7.28, show that:

$$\frac{T_0}{T_{T3}}=\frac{T_{10}-T_0}{T_{T4}-T_{T3}} \tag{7.29}$$

Substituting Equation 7.29 into 7.25

$$\eta_{TH} = 1-\frac{T_0}{T_{T3}}$$

and applying Equation 7.26 again

$$\eta_{TH}= 1 - \left[\frac{P_0}{P_{T3}}\right]^{\frac{\gamma-1}{\gamma}} \tag{7.30}$$

Note that

$$\frac{P_{T3}}{P_{T0}} = \left[\frac{P_{T3}}{P_{T2}}\right]\left[\frac{P_{T2}}{P_{T0}}\right]\left[\frac{P_{T0}}{P_0}\right]$$

where

$\frac{P_{T3}}{P_{T2}}$ *is the compression ratio, CR,*

$\frac{P_{T2}}{P_0}$ *, inlet recovery factor (which is unity in the ideal cycle)*

$\frac{P_{T0}}{P_0} = \left[1+\frac{\gamma-1}{\gamma}M_0^2\right]^{\frac{\gamma-1}{\gamma}}$ M_0 *is the free stream Mach*

Thus

$$\eta_{TH} = \left[\left(1+\frac{\gamma-1}{\gamma}M_0^2\right)(CR)\right]^{-\frac{\gamma-1}{\gamma}} \tag{7.31}$$

The significance of this equation is that thermal efficiency is now shown to be a function of two design variables: compression ratio and Mach. Efficiency increases with an increase in either parameter. These variations are shown in Figure 7.18 for two different Mach.

Turbine inlet temperature (T_{T4}) also has a small effect on thermal efficiency. Efficiency decreases slightly as TIT is increased, but the effect is smaller than Mach variations. The effects of TIT on thermal efficiency are shown in Figure 7.19.

Thermal efficiency does not tell the entire story because the operating temperatures of the cycle must also be considered. Ambient air temperature (T_0) is fixed by the flight condition. The maximum TIT is also fixed by the metallurgy of the turbine blades. The current state of the art limits TIT to about 3000°R, but higher limits may be permissible with a better technique for cooling the turbine blades.

Fixing T_0 and TIT, a variety of compression ratios are possible with each one yielding a different thermal efficiency. Figure 7.20 shows three cycles. Cycle 0-3-4-0 has a low pressure ratio, a low efficiency, and a low work capacity as denoted by the small enclosed area of the cycle. In the limit (compression ratio = 0), the work capacity and efficiency would be zero. At the other extreme, cycle 0-3" - 4"-0 would have a very high compression ratio and high thermal efficiency, but the work capacity would again be low. In the limit as the compression ratio is increased, the work would be zero, but the efficiency 100%. Obviously, neither of these cycles would be satisfactory in any practical application. These trends are cummarized in Table 7.3.

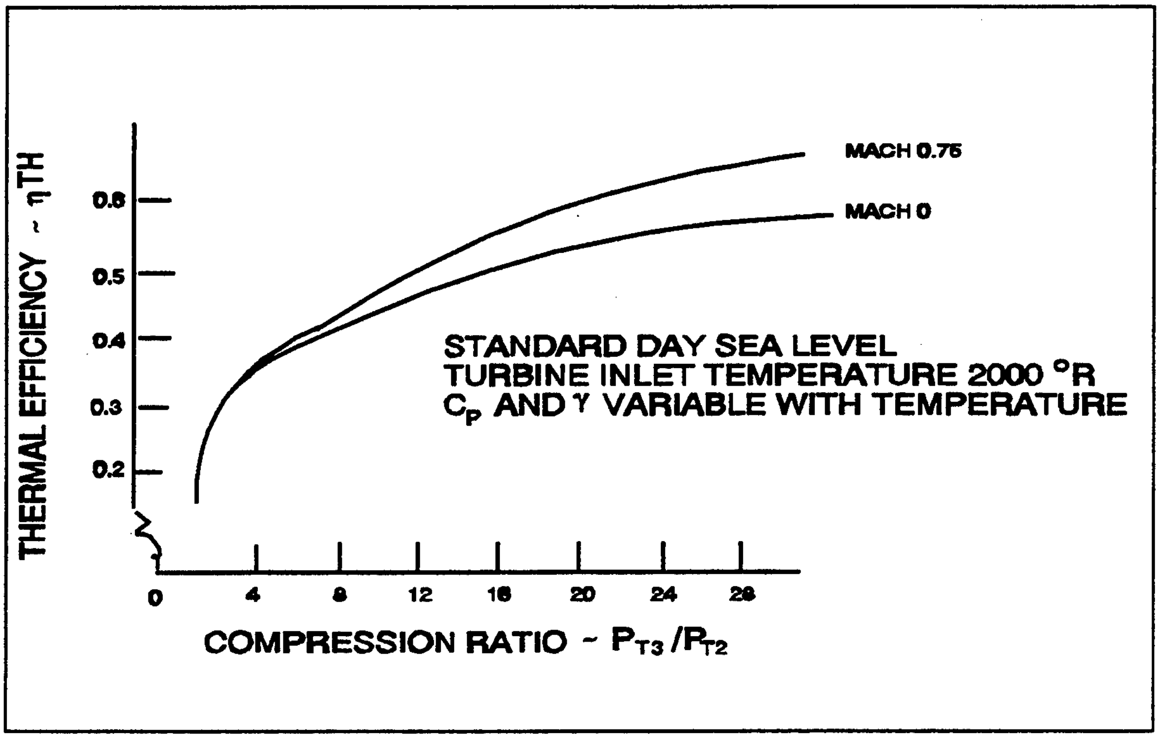


FIGURE 7.18. IDEAL TURBOJET THERMAL EFFICIENCY

TABLE 7.3

EFFECTS OF COMPRESSION RATIO ON ηTH AND W_N

CYCLE	CR	TH	W_N
0 - 3 - 4 - 0	LOW	LOW	LOW
0 - 3' - 4' - 0	MED	MED	HIGH
0 - 3" - 4" - 0	HIGH	HIGH	LOW

What is needed is a compromise compression ratio which will give an adequate work capacity at a reasonable thermal efficiency. The optimum compression ratio is derived in Appendix F for maximum net work, and results in

$$T_{T3}=\sqrt{T_0 \times T_{T4}}$$

.

7.7.5 IDEAL TURBOJET PERFORMANCE

We are now ready to determine the ideal cycle for an actual turbojet engine. However, let's slow down a moment and look at the plan of attack.

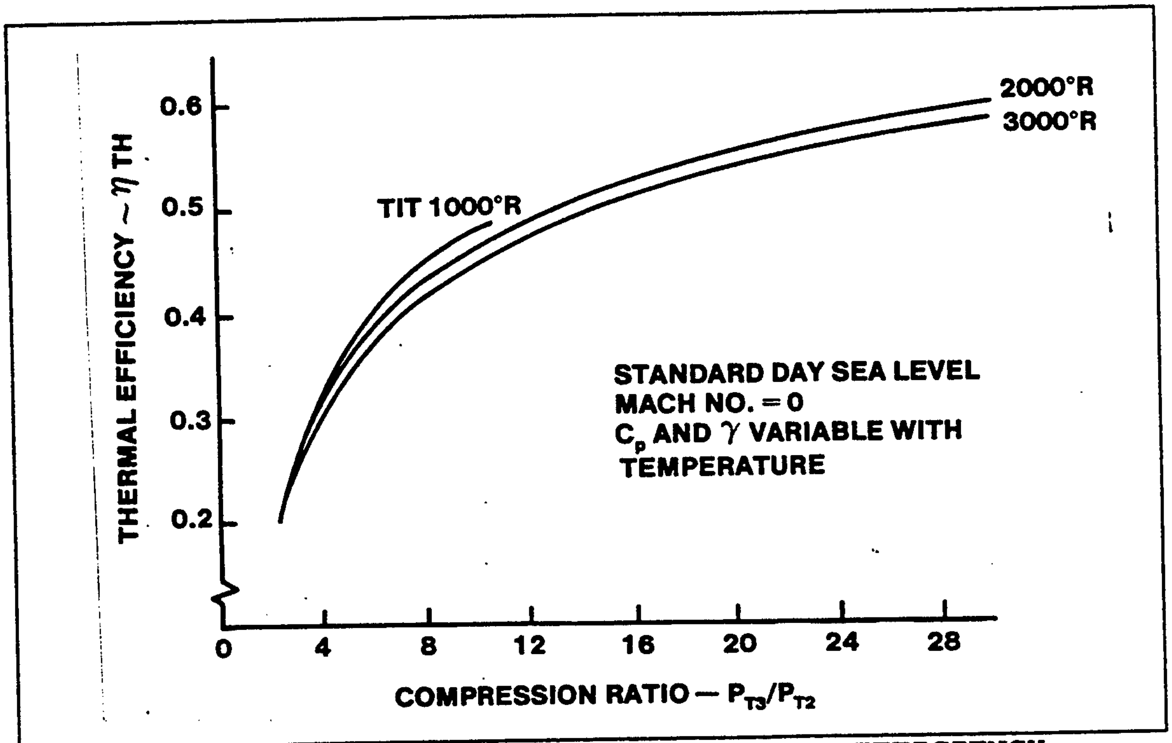


FIGURE 7.19. IDEAL TURBOJET THERMAL EFFICIENCY

We would like to determine the net thrust (F_n) and thrust specific fuel consumption (TSFC) of the turbojet engine. The two basic equations are:

$$F_n = \frac{\dot{w}_a}{g} (V_{10} - V_0) + A_{10} (P_{10} - P_0) \quad (7.32)$$

$$TFSC = \frac{\dot{w}_f}{F_n} \quad (7.33)$$

where $\dot{w}_a$ and $\dot{w}_f$ are the air and fuel flow rates respectively. We will then be able to examine trends and tradeoffs as a function of the variables.

The variables can be divided into flight conditions and engine parameters. The significant flight conditions (V_0, T_0 and P_0) define the free stream. The engine parameters are compression ratio, turbine inlet temperature, and airflow rate. An actual cycle analysis would also include the individual component efficiencies.

These assumptions will be made:

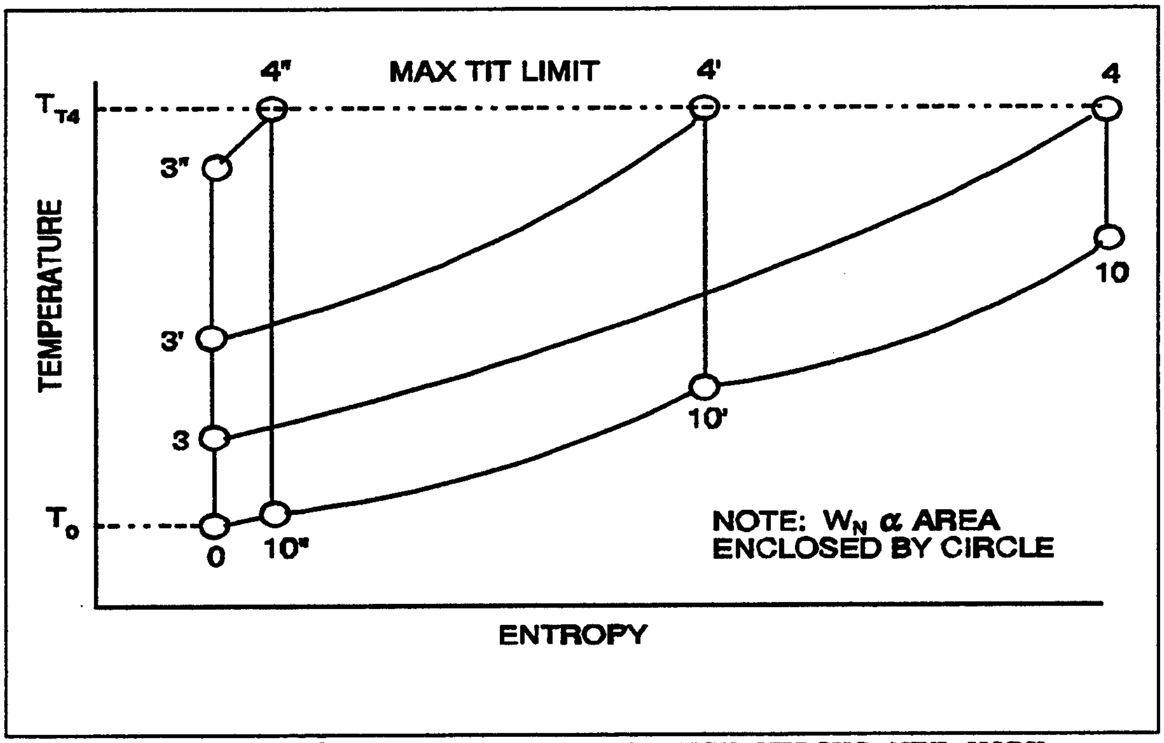


FIGURE 7.20. THERMAL EFFICIENCY VERSUS NET WORK

1. Individual components are 100% efficient
2. $W_T = W_C$ (no auxiliary drives)
3. No bleed air
4. Nozzle perfectly expands gas to ambient pressure
5. Addition of fuel to mass flow rate is negligible

The problem (to determine F_n and TSFC) can be solved analytically or graphically using the h-s diagram. The analytical approach is presented in Appendix F. The remainder of this chapter will be concerned with the graphical approach. The latter approach not only requires less mathematics but also gives a better insight into the actual processes occurring in the engine.

7.7.5.1 IDEAL TURBOJET CYCLE ANALYSIS

In this section we will construct the h-s diagram for a J-79 turbojet and then calculate F_n and TSFC. The specific flight conditions and engine parameters are listed in Table 7.4.

TABLE 7.4
FLIGHT CONDITIONS AND ENGINE PARAMETERS
FOR J-79 TURBOJET ANALYSIS

FLIGHT CONDITIONS			ENGINE PARAMETERS		
V_0	T_0	H_0*	TIT_{MAX}	CR	$\dot{W}_a$
230K	40°F	16,000 ft	1810°	13.5	170 lb/sec

* FREE STREAM ALTITUDE

SOLUTION

The problem will be solved in a series of steps. The resulting h-s diagram is shown in Figure 7.21 so that the reader may more easily follow the actual construction.

STEP 1: LOCATE STATION 0

H_0 = 16,000 ft => P_0 = 8.0 PSIA

T_0 = 40°F = 500°R

Enter h-s diagram with P_0 and T_0; read h_0 and s_0 directly

h_0 = 0.63 BTU/lb°R
S_0 = 0.63 BTU/16°R

NOTES

Only ambient temperature and pressure are required to locate station 0.

From the TPS Performance Manual we find 16,000 ft corresponds to δ = 0.5420. Hence P_0 = (14.7) δ PSIA

Always remember to convert °F to absolute.

(°R = °F + 460°)

STEP 2: LOCATE STATION 1,2

$$h_{T1}=h_1+\frac{V_1^{\,2}}{2gJ}=h_{T2}$$

NOTES

The purpose of the inlet is to slow the free stream airflow, thereby converting the kinetic energy of the flow into a pressure rise. This is a consequence of Bernoulli's equation.

$$=h_0+\frac{V_0^2}{2gJ}$$

$$=120+\frac{[(1.69)(230)]^2}{50,100}$$

$$h_{T1}=h_{T2}=123\ BTU/lb$$

$s_1 = s_2 = s_0 = 0.63$ BTU/lb°R

Read P_{T2} directly from h-s diagram (interpolate)

$P_{T2} = 8.7$

In an ideal inlet operating on design, $V_0 = V_1$. Don't forget to convert knots into ft/sec: (1K = 1.69 ft/sec.)

Since the inlet processes are all isentropic, there can be no loss in total quantities. Hence
$h_{T0} = h_{T1} = h_{T2}$
$s_0 = s_1 = s_2$.
However, $V_1 = V_2$ as the kinetic and static values may vary.

STEP 3: LOCATE STATION 3

$P_{T3} = (CR)(P_{T2})$

where $CR = \dfrac{P_{T3}}{P_{T2}}$

NOTES

The compressor increases the total pressure of the airflow. Since the ideal process is isentropic, $s_2 = s_3$.

$P_{T3} = (13.5)(8.7)$
$= 117.45$ PSIA

Read h_{T3} from h-s diagram
$h_{T3} = 255$ BTU/lb
$s_3 = 0.63$ BTU/lb°R

STEP 4: LOCATE STATION 4

NOTES

$P_{T4} = P_{T3} = 117.45$ PSIA

$T_{T4} = TIT_{MAX} = 1810°F = 2270°R$

The ideal combustion process occurs at constant pressure. The exit temperature of the gas is limited to 1810°F which corresponds to MIL POWER.

Locate on h-s diagram; read h & s directly
$h_{T4} = 565$ BTU/lb
$s_4 = 0.82$ BTU/lb°R

STEP 5: LOCATE STATION 5

NOTES

From Steps 2 and 3
$W_c = h_{T3} - h_{T2}$
$= 255 - 123$
$= 132$ BTU/lb

The turbine, located between Stations 4 and 5 drives the compressor.

Hence,
$W_T = 132$ BTU/lb
$h_{T5} = h_{T4} - W_T$
$= 565 - 132$
$= 433$ BTU/lb
$s_5 = s_4 = 0.82$ BTU/lb°R

The addition of auxiliary drives does not add a significant error in the results if neglected because they require only a small percentage of the energy required by the compressor.

STEP 6: LOCATE STATION 10

$P_{10} = P_0 = 8$ PSIA

$s_{10} = s_5 = 0.82$ BTU/lb°R

Locate on h-s diagram; read h_{10} directly

$h_{10} = 270$ BTU/lb

$W_N = h_{T5} - h_{10}$

$= 433 - 270$

$= 163$ BTU/lb

NOTES

Up to this point, the results could apply equally well to a turboprop, turboshaft, or turbofan. In other words, the output energy ($W_N = h_{T5} - h_{10}$) could be used to drive a second turbine or a fan.

The turbojet uses a nozzle to convert the static enthalpy at Station 5 into a high velocity gas at Station 10. The ideal nozzle isentropically expands the gas to ambient pressure ($P_{10} = P_0$).

SPECIAL NOTE:

The cycle is closed in the atmosphere as the gas at the nozzle exit cools at ambient pressure to the ambient temperature. The enthalpy, pressure, and temperature at Stations 0 and 10 are static quantities (these are total quantities at all other stations). The total enthalpy at Stations 0 and 10 are:

$$h_{T0} = h_{T1} = h_{T2}$$

$$h_{T10} = h_{T5}$$

This shows that the diffuser (inlet) and nozzle perform exactly opposite functions.

STEP 7: CALCULATE EXIT VELOCITY

NOTES

Since $h_{T10} = h_{T5}$ and h_{T10}

$$= h_{10} + \frac{V_{10}^2}{2gJ}$$

V_{10} is easily determined.

$$V_{10}=\sqrt{2gJ(h_{T5}-h_{10})}$$

$$=\sqrt{(50,100)(163)}$$

V_{10} = 2858 ft/sec

STEP 8: CALCULATE NET THRUST

NOTES

$$F_n=\frac{\dot{w}_a}{g}(V_{10}-V_0)$$

The fuel contribution has been neglected as it is typically small compared to air flow

$$(\dot{w}_f<0.02\dot{w}_a).$$

$$=\frac{170}{32.2}(2858-389)$$

F_n = 13035 lb

If the gas at the nozzle is not expanded to the ambient pressure, then Equation 7.10 must be used.

STEP 9: CALCULATE FUEL FLOW RATE

$$\dot{w}_f = 0.195\ \dot{w}_a (h_{T4} - h_{T3})$$

$$= 0.195\ (170)\ (565\text{-}255)$$

$$\dot{w}_f = 10276\ lb/hr$$

NOTES

The heat energy input in the cycle (Q_{IN}) is obtained from combustion of hydrocarbons. The average heating value (H. V.) for hydrocarbons is 18,500 BTU/lb fuel. Each pound of air requires a heat input of $Q_{IN} = h_{T4} - h_{T3}$. The total heat input per second is then

$$\dot{w}_a (h_{T4} - h_{T3})\ .$$

The total heat added is

$$\dot{w}_a (h_{T4} - h_{T3}) = \dot{w}_f\ H.V.$$

Hence

$$\dot{w}_f = \frac{\dot{w}_a}{H.V.} (h_{T4} - hT)$$

However, fuel flow is normally given in pounds per hour, so the last equation must be multiplied by 3600 sec/hr.

STEP 10: CALCULATE TSFC

$$TSFC = \frac{\dot{w}_f}{F_n}$$

$$= \frac{10276}{13035}$$

$$TSFC = 0.788 \frac{(lb\text{-}fuel)}{(lb\text{-}thrust)\ (hr)}$$

NOTES

TSFC is another measure of thermal efficiency and is more commonly given with engine specifications. In this particular example, η_{TH} - 0.54, which does not convey nearly the information that is contained in TSFC.

This completes the ideal cycle analysis of the turbojet engine. The results were:

F_n = 13,035 lb

$$TSFC = 0.788 \frac{(lb\text{-}fuel)}{(lb\text{-}thrust)\ (hr)}$$

$$\dot{w}_f = 10276 lb/hr$$

Do these values seem reasonable for the J-79?

7.7.5.2 PROPULSIVE EFFICIENCY

All jet propulsion devices develop thrust by changing the velocity of the working fluid, and it is desirable to define an efficiency factor which shows how efficiently the process is carried out. This efficiency factor is called the propulsive efficiency, and it is indicative of how efficiently the kinetic energy of the engine is used. It is defined as the ratio of useful thrust power output to the available propulsive energy, which, in turn, is equal to the useful output plus the kinetic energy loss at the exit. That is,

$$\eta_p = \frac{THP\ output}{THP\ output + KE\ losses\ at\ exit}$$

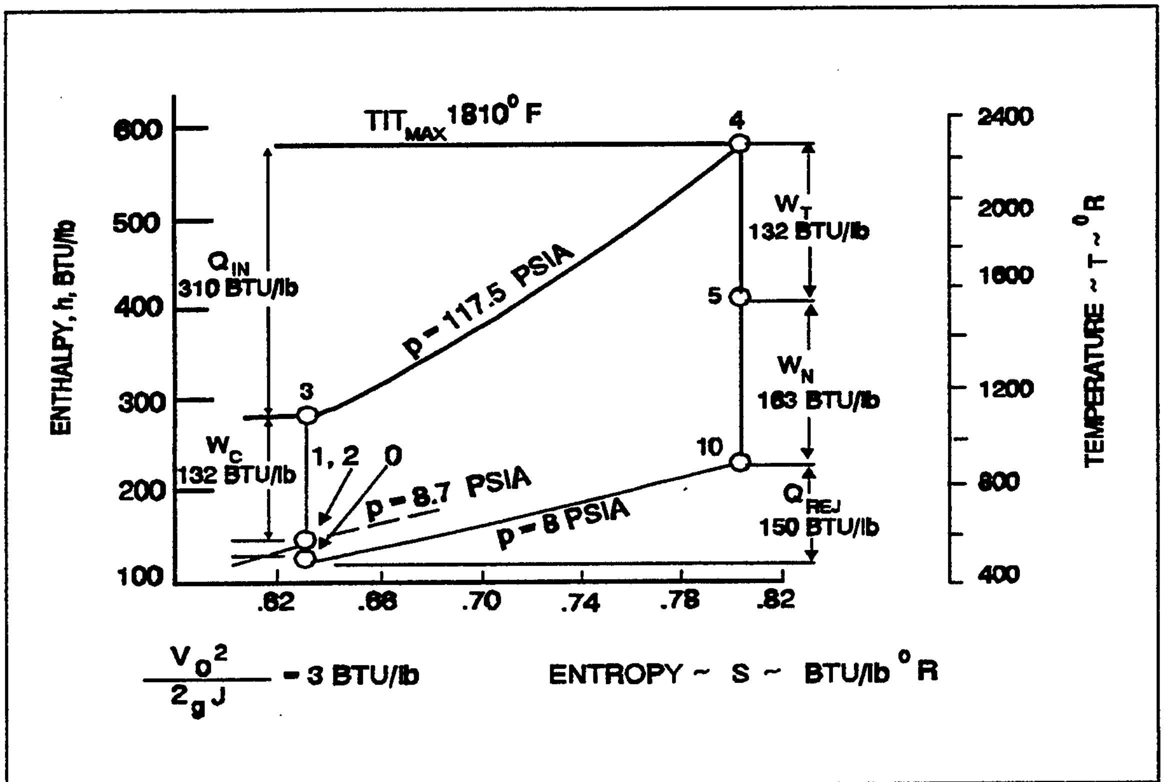


FIGURE 7.21. IDEAL CYCLE FOR THE J-79 TURBOJET (PER LB OF AIR V_o = 230 KTS, T_o = 40°, H_o = 1600 FT

$$\eta_p = \frac{F_n V_0}{F_n V_0 + \dfrac{\Delta KE}{\Delta t}} \tag{7.34}$$

$$\eta_P = \frac{\dot{m}\,(V_{10} - V_0)\, V_0}{\dot{m}\,(V_{10} - V_0)\, V_0 + \dot{m}\left[\dfrac{V_{10} - V_0}{2}\right]^2} \tag{7.35}$$

$$\eta_P = \frac{2 V_0}{V_{10} + V_0} \tag{7.36}$$

It should be noted that this definition ignores the heat losses which occur at the nozzle exit and considers only the kinetic energy loss at that particular section.

Equation 7.36 applies to the air breathing engine. Note that when flight velocity is zero, there is no useful power; therefore, the propulsive efficiency is zero. Propulsive efficiency is equal to unity when the effective exhaust velocity is equal to the flight velocity. The latter case has no physical meaning because, in this condition, the thrust is zero (no momentum change).

Figure 7.22 illustrates variation of η_p with V_0 for the different air-breathing aircraft engines.

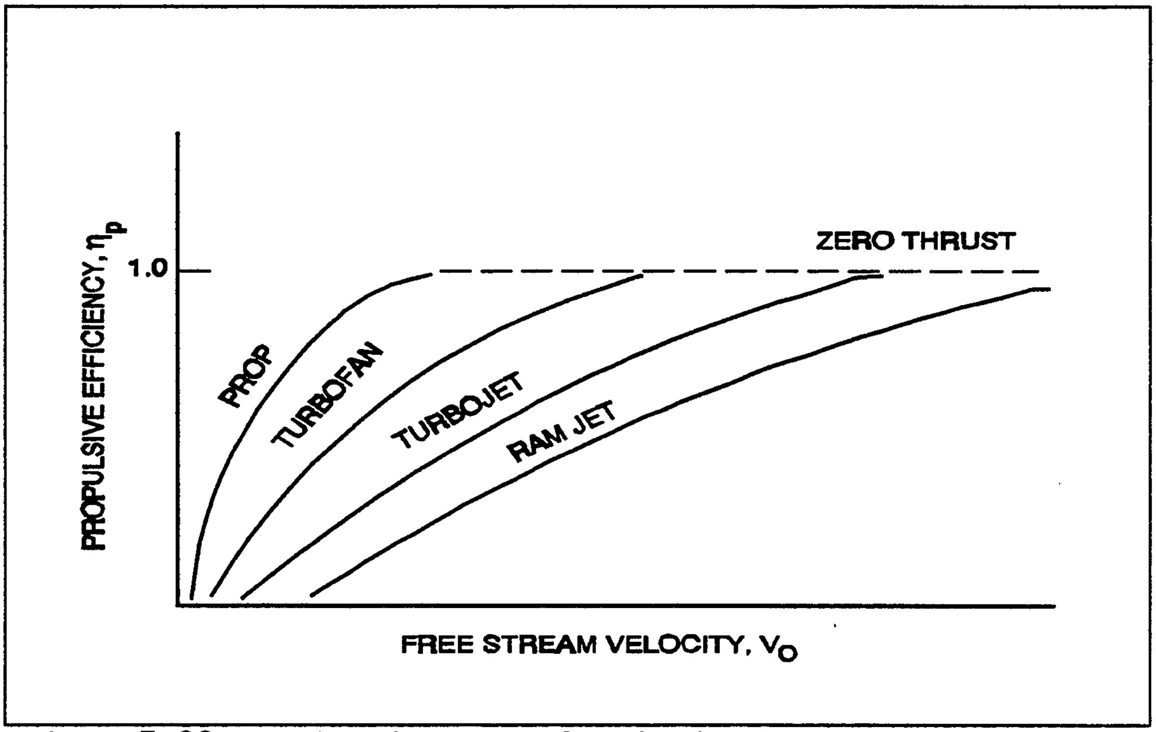


FIGURE 7.22. PROPULSIVE EFFICIENCY OF AIR-BREATHING ENGINES

7.7.5.3 OVERALL EFFICIENCY

The product of the propulsive and thermal efficiencies yields a further criterion for judging the performance of jet propulsion engines. It is called overall efficiency and is written

$$\eta_0 = \eta_P \eta_{TH} \tag{7.37}$$

7.7.5.4 IDEAL TURBOJET TRENDS: NET THRUST.

The ideal net thrust per unit mass flow of a turbojet engine is given by

$$F_n \,/\, \frac{w_a}{g} = (V_{10} - V_0) \tag{7.38}$$

$$= \sqrt{2gJC_p\left[TIT\left[1-(CR\ f(m))^{-\frac{\gamma-1}{\gamma}}\right]-T_0\ \left[(CR\ f(m))^{+\frac{\gamma-1}{\gamma}}\right]\right]-1+V_0^{\,2}}-V_0 \tag{7.39}$$

where

$$f(M) = 1-\frac{\gamma-1}{2}M_0^{\,2} \text{ and } V_0 = M_0\sqrt{\gamma R T_0}$$

Equation (7.39) is derived in Appendix F. This equation shows that the net thrust per unit mass flow is a function of tow design variables (TIT and CR) and tow flight parameters (M_0 and T_0). The variations in net thrust per unit mass flow with these parameters are shown in Figures 7.23 and 7.24. The results are summarized in Table 7.5.

TABLE 7.5
SUMMARY OF NET THRUST TRENDS
Ideal Turbojet

VARIABLE INCREASE	$\left[F_n / \frac{\dot{w}_a}{g}\right]$
TURBINE INLET TEMPERATURE	INCREASE
COMPRESSION RATIO	OPTIMUM
MACH NUMBER - M_0	DECREASE
ALTITUDE - M_0	INCREASE

To fully appreciate the results, it must be understood that the variations tabulated above are valid only when all other variables are held constant. For example, the increase in $\left[F_n / \frac{\dot{W}_a}{g}\right]$ with altitude does not mean an increase in net thrust. The net thrust actually decreases with increasing altitude because the airflow through the engine decreases.

A particular variable of interest is Mach, the decrease in $\left[F_n / \frac{\dot{W}_a}{g}\right]$ with increasing M_0 is primarily due to the increase in V_0, which is the ram drag per unit mass flow. If ram rag is deducted from the net thrust, the gross thrust per unit mass flow would result, which increases with Mach.

One observation worthy of note can be seen by looking at Figures 7.23 and 7.24. The optimum compression ratio (TIT constant) for maximum thrust decreases with increasing Mach number. The limiting case is for CR=1. This engine is called a ramjet.

7.7.5.5 IDEAL TURBOJET TRENDS: THRUST SPECIFIC FUEL CONSUMPTION

The fuel consumption of an engine is usually given in terms of the amount of fuel required to produce a given amount of thrust. It is the key parameter for comparing engines. For example, a particular flight condition for any given aircraft produces a drag which the engine(s) must overcome. If Engine A has better TSFC (lower) than Engine B for the same flight conditions, Engine A will yield better range or require less fuel since both engines must develop the same thrust.

Thrust specific fuel consumption is defined as

$$TSFC = \frac{W_f}{F_n} \quad \frac{lbs\ fuel}{lbs\ thrust\ hr} \tag{7.40}$$

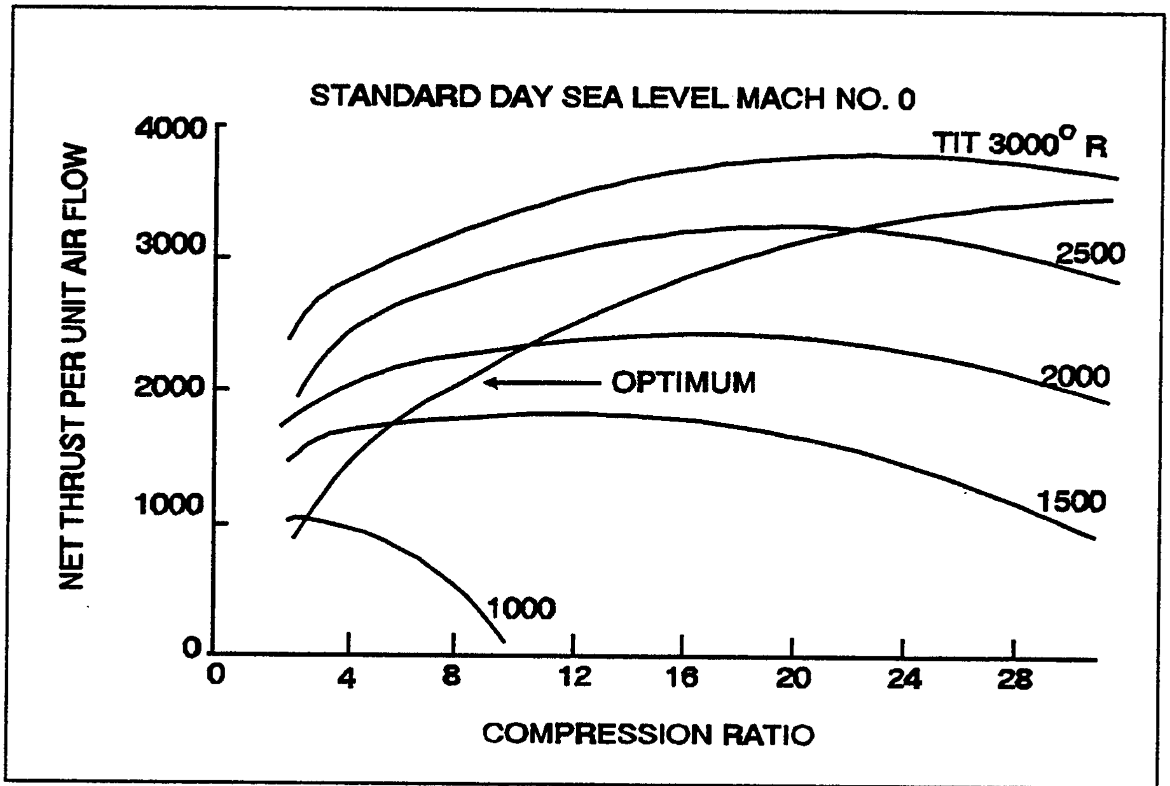


FIGURE 7.23. IDEAL TURBOJET NET THRUST

$$= \frac{\dot{W}_a (H_{T4} - H_{T3})}{H.V.\ F_n}$$

$$= \frac{C_P\ g\ (T_{T4} - T_{T3})}{H.V.\ \left[F_n / \frac{\dot{W}_a}{g}\right]}$$

Comparison of Equation 7.41 with 7.39 shows that TSFC is a function of the same variables as net thrust per unit mass flow. Note that the compressor discharge temperature is established by the CR, altitude, and Mach. The effects of these variables are shown in Figure 7.25 and 7.26 and are summarized in Table 7.6.

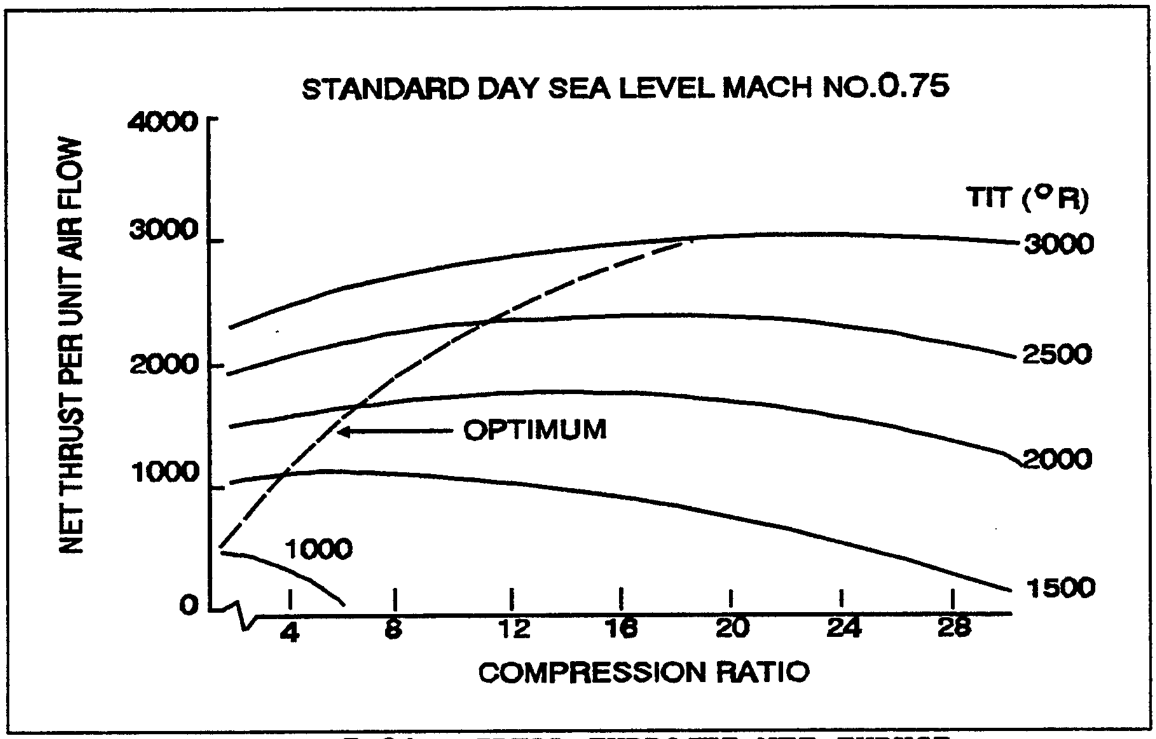


FIGURE 7.24. IDEAL TURBOJET NET THRUST

TABLE 7.6
SUMMARY OF TSFC TRENDS
Ideal Turbojet

VARIABLE INCREASE	TSFC
TURBINE INLET TEMPERATURE	INCREASE
COMPRESSION RATIO	DECREASE
MACH - M_0	INCREASE SLIGHTLY*
ALTITUDE - H_0	DECREASE SLIGHTLY

* This effect is not the inverse of h_{TH}, due to the difference in the (V_{10} - V_0) terms.

The definition of TSFC can be related to the overall efficiency by converting the jet thrust to jet thrust horsepower. The following expression is obtained:

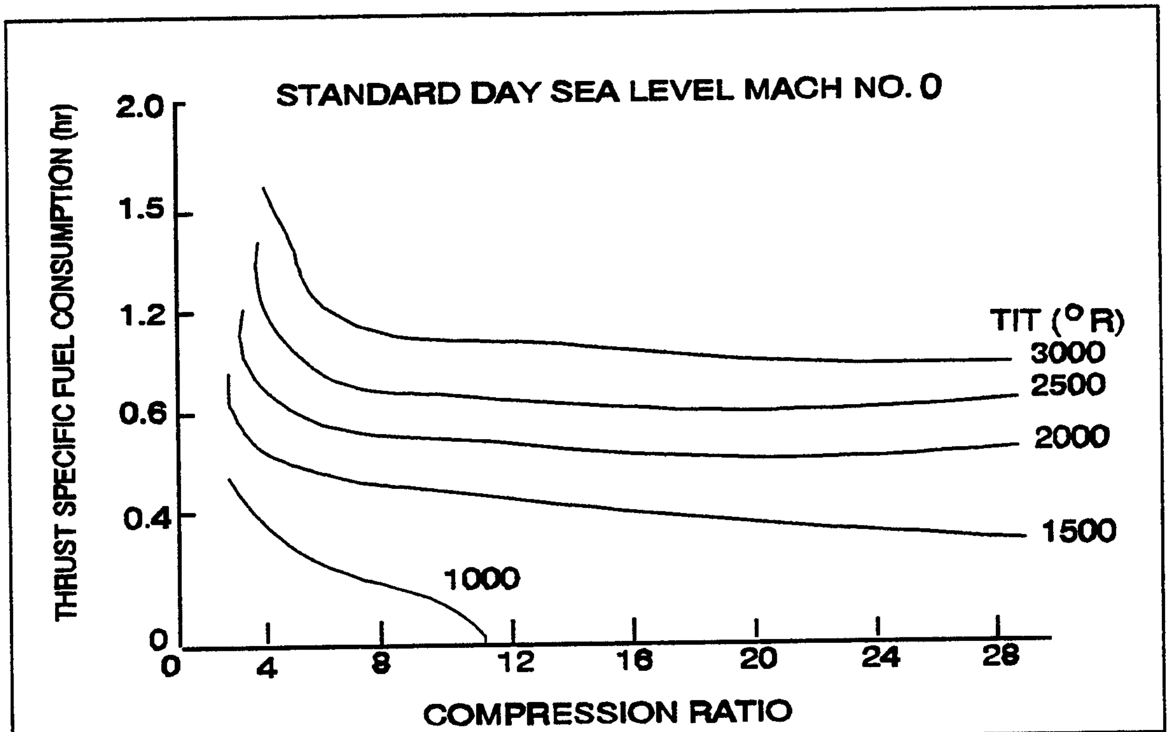


FIGURE 7.25. IDEAL TURBOJET THRUST SPECIFIC FUEL CONSUMPTION

$$\eta_0 = \frac{V_0}{sfc\ (H.V.)} \tag{7.42}$$

where H.V. is the lower heating value of the fuel. Here also, the overall efficiency and sfc are indicative of the same thing. A typical value of sfc for a turbojet engine at sea level static is 0.9 lb per lb-hr. For a turbofan engine a typical sfc for the same condition is 0.7 lb per lb-hr.

Increasing TIT increases both thrust and TSFC (lower efficiency). Also, the optimum compression ratio for optimum thrust is lower than for optimum TSFC. Consequently, the selection of an engine operating point is a compromise. In fact, each design point of every engine component is a compromise within itself. In addition, off-design consideration must also be considered as performance tends to degrade much faster for the off-design condition in some areas. The nozzle is a good example of off-design considerations dictating the operating point.

7.7.6 IDEAL TURBOFAN PERFORMANCE

The turbofan engine has two primary advantages over the turbojet engine: higher net thrust and lower thrust specific fuel consumption. In this section we will demonstrate this by converting the J-79 turbojet engine into a turbofan engine and then calculate

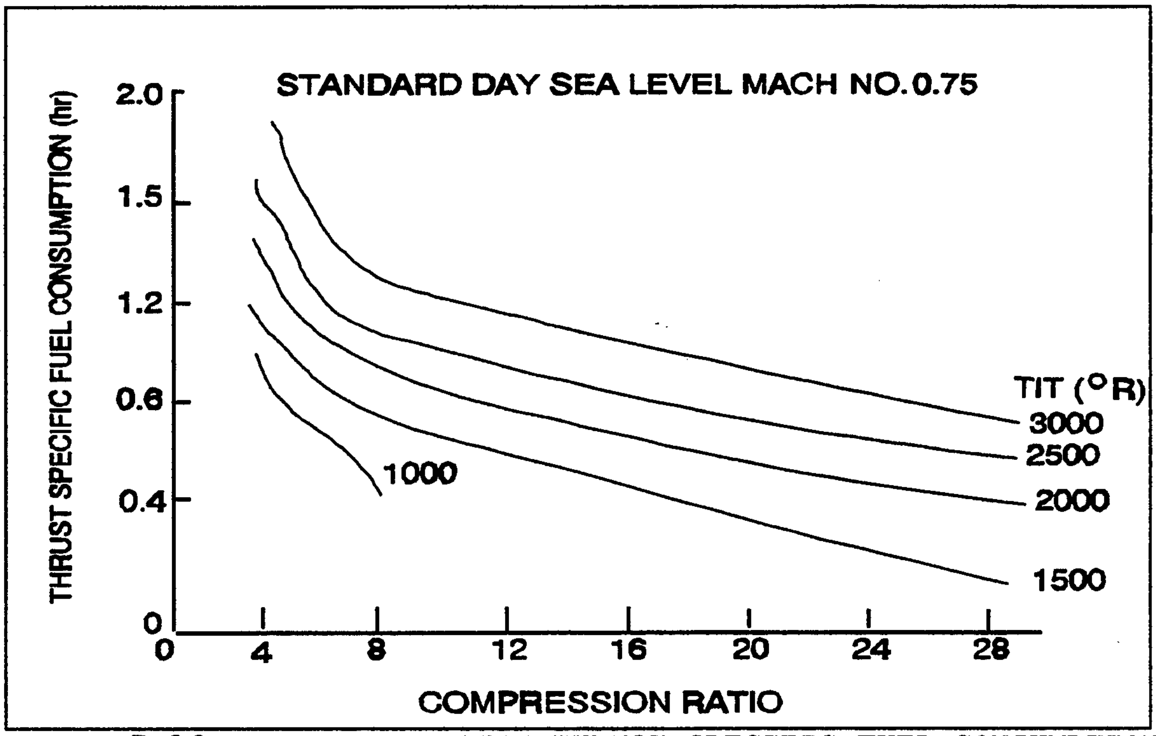


FIGURE 7.26. IDEAL TURBOJET THRUST SPECIFIC FUEL CONSUMPTION

F_n and TSFC. Although the results will be optimistic, the overall improvement is considerable. However, the turbofan does have some disadvantages when compared to the turbojet. We will examine the relative merits of each in a subsequent section.

7.7.6.1 TURBOFAN OPERATION

The fan stage consists of two primary components: an inlet and a fan compressor. The purpose of the inlet is the same as in the core engine--slow the free stream and thereby convert the kinetic energy of the flow into a pressure rise. In some turbofan configurations, the core and fan stage inlets are identical. The fan compressor increases the total pressure of the bypassed airflow. The ideal process for both components is obviously isentropic. Before starting the turbofan cycle analysis we need to discuss the interaction of bypass ratio (β) with fan compression ratio (CR_f). Bypass ratio is defined by

$$\beta = \frac{\dot{w}_{a_{duct}}}{\dot{w}_{a_c}} \tag{7.43}$$

where $\dot{w}_{a_{duct}}$ is the air flow rate through the fan duct that doesn't go through core and $\dot{w}_{ac}$, the air flow rate through the core engine. The effect of fan compression ratio and bypass ratio on TSFC is shown in Figure 7.27. The turbine work limit is reached when all of the net energy output of the core engine is used to drive the fan stage (no core engine thrust). The optimum TSFC (and net thrust) is obtained when the core exhaust gas velocity is equal to the bypassed exhaust gas velocity. TSFC improves as the bypass ratio is increased, the limit being a shrouded turboprop engine. However, high bypass engines suffer from lack of performance at higher Mach. Note that the optimum fan compression ratio decreases as bypass ratio increases.

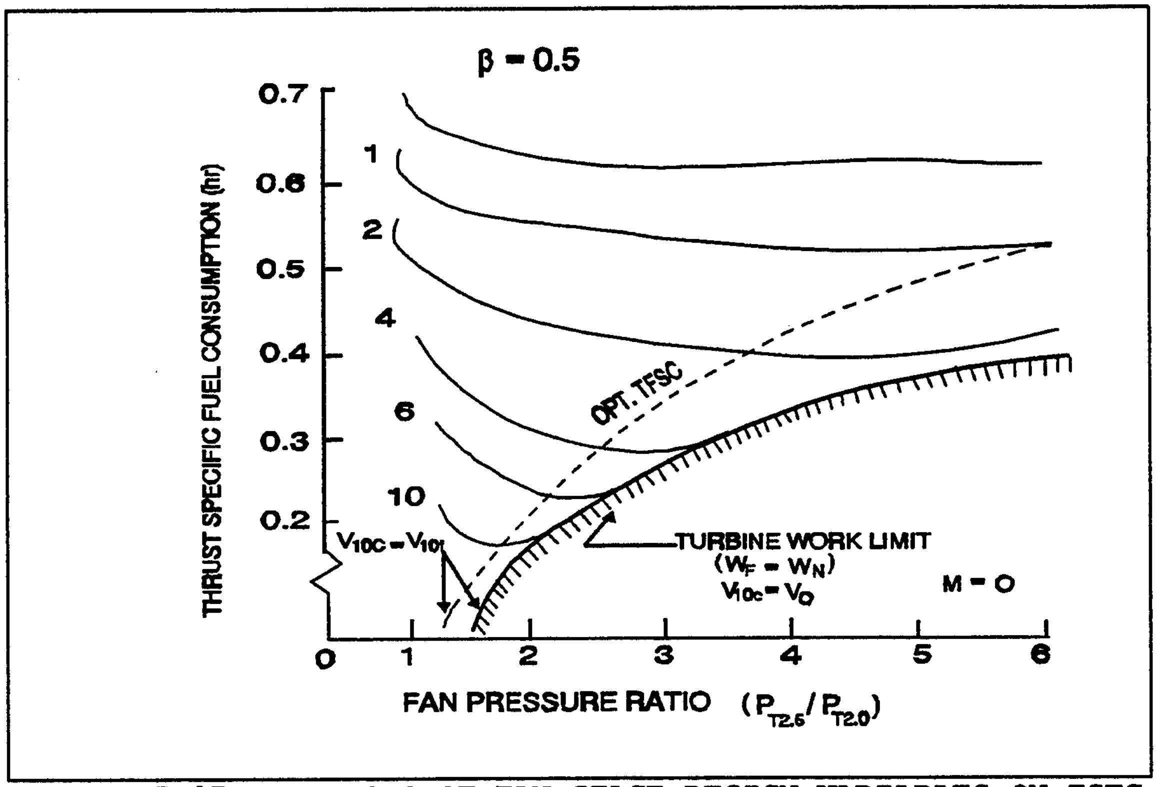


FIGURE 7.27. EFFECTS OF FAN STAGE DESIGN VARIABLES ON TSFC

NOTE: EACH BYPASS RATIO HAS AN OPTIMUM FAN COMPRESSION RATIO. THE RATIO OF TSFC OF THE TURBOFAN ENGINE TO THE TURBOJET ENGINE IS

$$\frac{TSFC_{TF}}{TSFC_{TJ}} = \frac{1}{1+\beta}$$

WHERE THE OPTIMUM FAN COMPRESSION RATIO IS USED WITH β. FOR EXAMPLE, A BYPASS RATIO OF TWO WITH THE ASSOCIATED OPTIMUM FAN

The effect of bypass ratio on net thrust is shown in Figure 7.28. The curve shows that net thrust continues to increase with bypass ratio, but the relative increase becomes smaller for the higher ratios.

Core compression ratio also affects thrust and net TSFC as shown in Figures 7.29 and 7.30. TSFC improves with increasing core compression ratio, whereas there is an optimum core compression ratio required to optimize net thrust. These are the same trends displayed by the core engine.

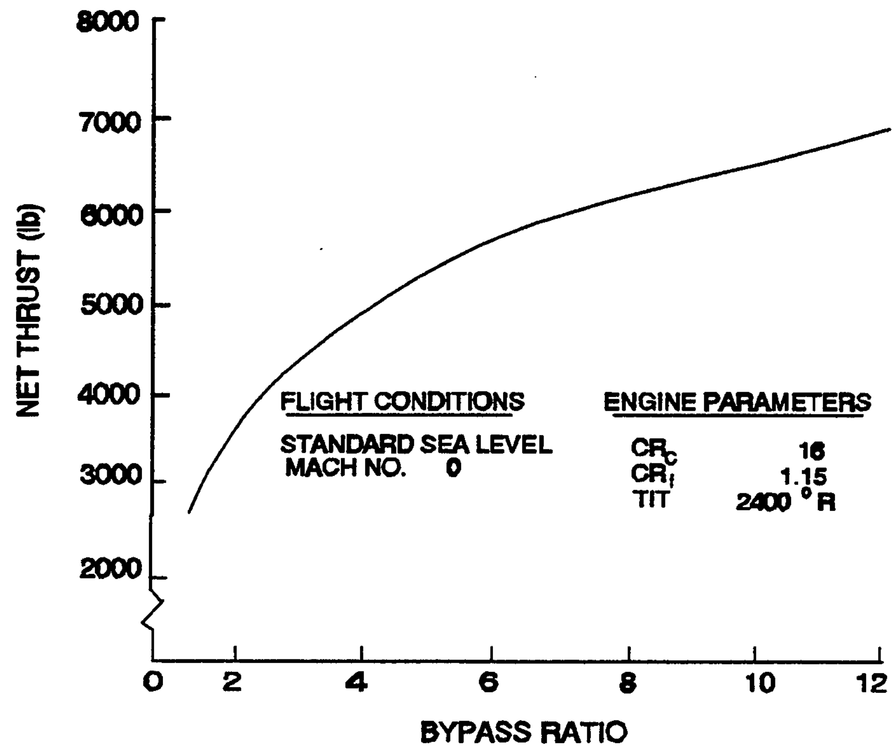


FIGURE 2.28. NET THRUST IMPROVEMENTS WITH BYPASS RATIO

NOTE: FOR A SPECIFIC CORE ENGINE AND FAN COMPRESSION RATIO

7.7.6.2 VARIATION IN TSFC OF A TURBOFAN WITH MACH

As Mach increases, the optimum (lowest) TSFC occurs at a progressively lower bypass ratio. This trend is shown in Figure 7.31. In addition, TSFC degrades with increasing Mach. In designing an engine, the propulsion engineer optimizes the performance for the specific mission of the aircraft. For instance, a transport aircraft designed to cruise at Mach 0.8 might have a bypass ratio and fan compression ratio of two.

A fighter type aircraft presents a more complex problem since the overall mission is divided into several phases, each requiring a different Mach/altitude combination. Two solutions are possible: (1) compromise engine, and (2) a variable cycle engine.

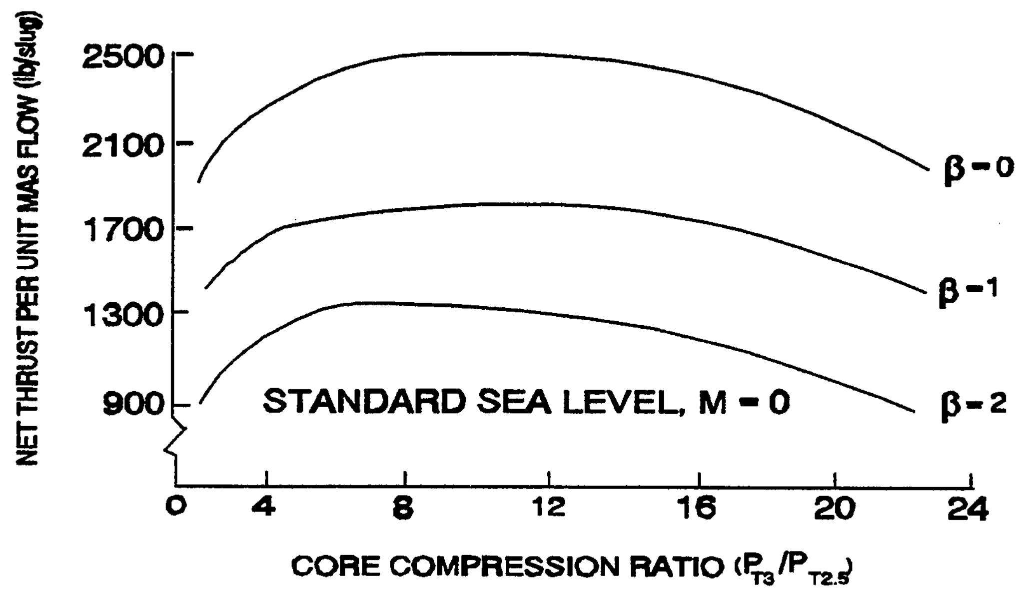


FIGURE 7.29. EFFECT OF CORE COMPRESSION RATIO ON NET THRUST FOR THE TURBOFAN (TIT=2400°R AND CR_f=2)

Current production engines compromise overall performance while attempting to retain adequate performance in the most crucial phases of the mission.

7.7.6.3 THE VARIABLE CYCLE ENGINE

The variable cycle engine is basically a variable bypass engine. The amount of bypass air is varied over a wide range and programmed so that the engine has the optimum bypass ratio for every flight speed. It also has the potential for substantially reducing installation losses in both the inlet and the nozzle. Engine technology required to implement variable cycle engines includes (1) variable-pitch, variable-camber fans (similar in basic principle to the variable-pitch propeller but more complex), (2) variable-area turbine inlet nozzles, (3) variable-area convergent-divergent (C-D) exhaust nozzles, and (4) a propulsion control system capable of integrating all the variable-area components with a fuel control.

7.7.6.4 IDEAL TURBOFAN CYCLE ANALYSIS

In this section we will construct the h-s diagram for a turbofan engine using the J-79 turbojet as the core. We will then calculate F_n and TSFC and compare these with our

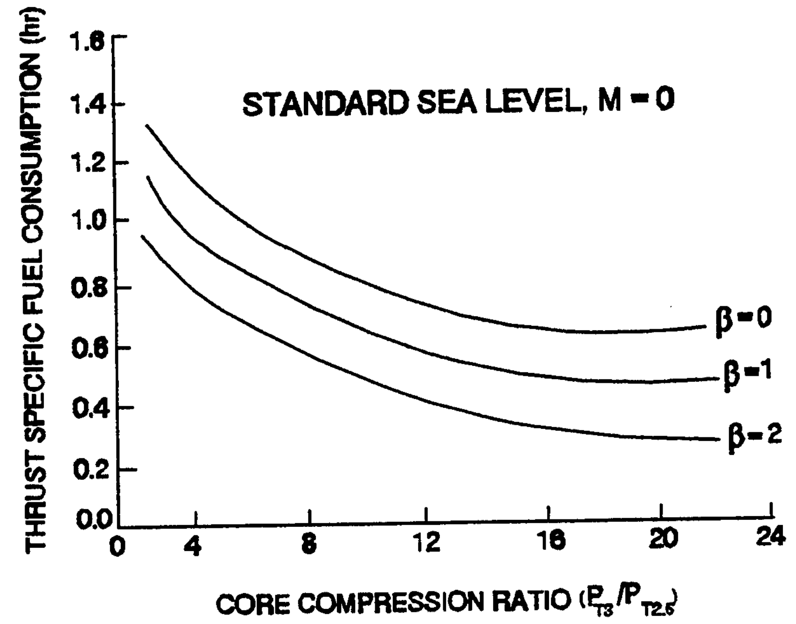


FIGURE 7.30. EFFECT OF CORE COMPRESSION RATIO ON TSFC FOR THE TURBOFAN (TIT=2400°R AND CR_f=2)

original values for the core engine. The specific flight conditions and core engine parameter will be the same as in Table 7.4. We will arbitrarily pick a bypass ratio of two and a fan compression ratio of three. The flight conditions and engine parameters are summarized in Table 7.7.

TABLE 7.7

FLIGHT CONDITIONS AND ENGINE PARAMETERS FOR CONVERTED J-79 TURBOFAN ANALYSIS ENGINE PARAMETERS

FLIGHT CONDITIONS			CORE ENGINE		FAN ENGINE		
V_0	T_0	H_0	TIT_{MAX}	CR_c	$\dot{w}_{ac}$	ρ	CR_f
230K	40°F	16,000F	1810°F	13.5	170 lb/sec	2	3

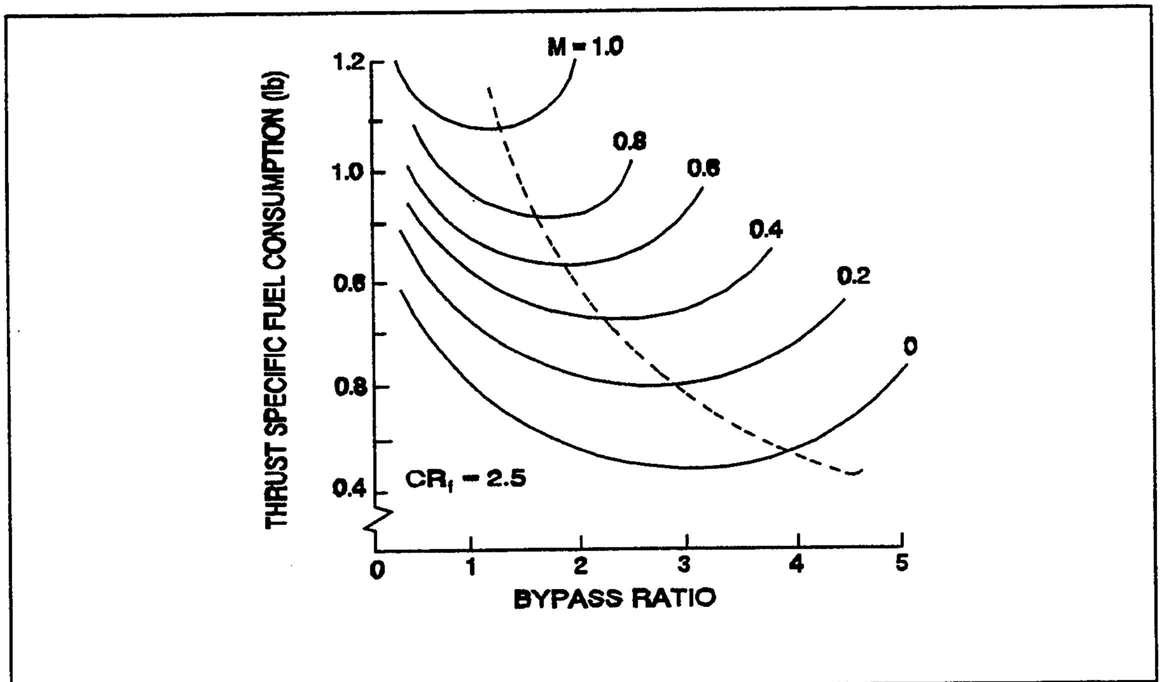


FIGURE 7.31. MACH EFFECTS FOR AN ACTUAL TURBOFAN ENGINE

The subscript "c" refers to core engine parameters, and "f," fan engine parameters.

SOLUTION

The solution to the problem is identical to the ideal turbojet engine analysis up to Step 7. The net energy output of the cycle was found to be 158 BTU/lb. Part of this energy will now be used to drive the fan, while the remainder will be expanded in the core engine nozzle to produce the core thrust. Continuing the analysis from Step 6 of the turbojet engine, we must next construct the h-s diagram for the fan section.

STEP 7A: LOCATE FAN STATION 0

Locate on h-s DIAGRAM

P_0 = 8 PSIA
T_0 = 500°R

Read h and s directly
h_0 = 120 BTU/lb
s_0 = 0.63 BTU/lb°R

NOTES

This step is identical with Step 1 for the turbojet because the free stream is identical.

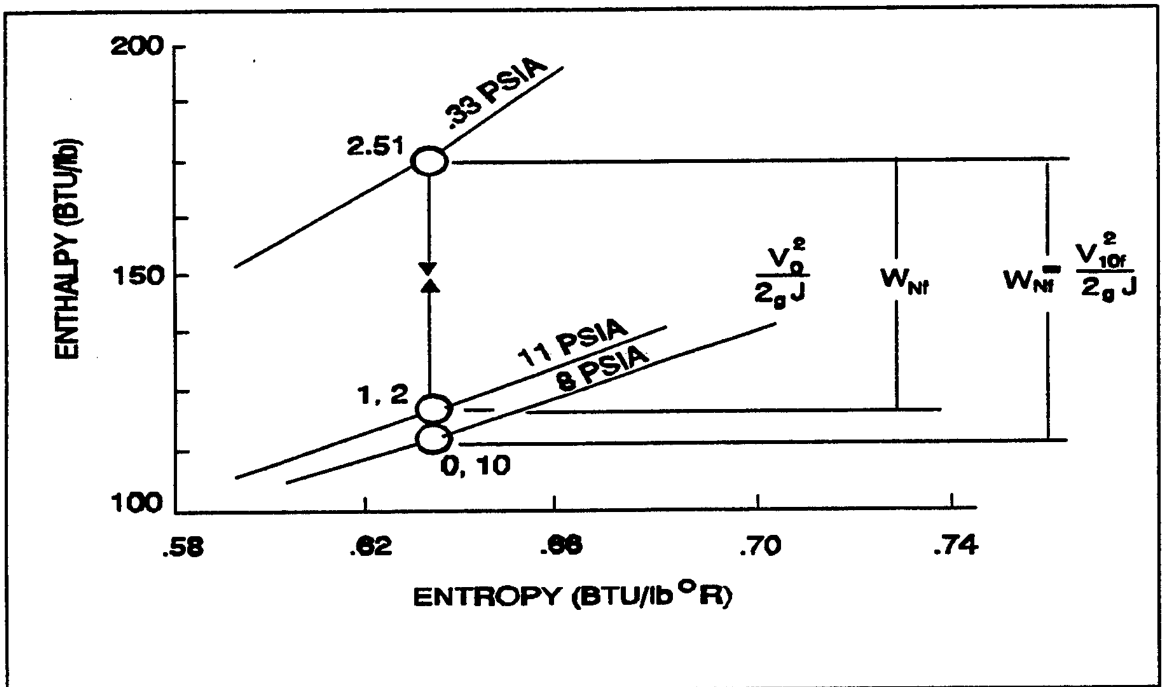


FIGURE 7.32. h-s DIAGRAM FOR THE FAN STAGE

STEP 8A: LOCATE FAN STATION 2

$$h_{T1F}=h_{T2f}=h_0+\frac{V_0^2}{2gJ}$$

$$=120+\frac{(1.69)\,(230)^2}{50,100}$$

= 123 BTU/lb

$s_1 = s_2 = s_0 = 0.63$BTU/lb°R

$P_{T2} = 11$ PSIA

NOTES

Again this step is identical with Step 2 for the turbojet analysis because the fan and core inlets see the same free stream velocity.

STEP 9A: LOCATE FAN STATION 3

$P_{T2.5} = (CR_f)\,(P_{T2f})$

$$where\ CR_f = \frac{P_{T3f}}{P_{T2f}}$$

$P_{T2.5} = (3)\,(11)$
$= 33$ PSIA

Read h directly

$h_{T2.5} = 175$ BTU/lb
$s_{2.5} = s_{2f} = s_{1f} = s_{0f}$

NOTES

This is the first step which differs.

STEP 10A: LOCATE FAN STATION 10

$P_{10f} = P_{0f} = 8$ PSIA
$h_{10f} = h_0 = 120$ BTU/lb
$s_{10f} = s_0$

NOTES

The high pressure gas is now expanded to the ambient pressure without any additional processing. The ideal fan section is isentropic; hence, the entropy does not change throughout the cycle.

STEP 11A: CALCULATE FAN EXIT VELOCITY

$$V_{10f} = \sqrt{2gJ(h_{T3t} - h_{10f})}$$

$= (50{,}100)\,(175\text{-}120)$
$V_{10f} = 1660$ fps

NOTES

A duct burning fan would add heat energy at this point instead of expanding the flow. The analysis would then follow the turbojet cycle, but there would not be a turbine to drive.

STEP 12A: CALCULATE WORK THE CORE ENGINE MUST SUPPLY TO DRIVE THE FAN

$W_F = \beta_{wf} = \beta(h_{T3f} - h_{T2f})$
$= 2\ (175 - 123)$
$W_F = 104\ BTU/lb_{core}$

NOTES

This step is straightforward but requires some thought. The fan acts on an airflow equal to $\beta \dot{w}_{ac}$. Each pound of this air requires an amount of work equal to W_f. The total work required by the fan is thus $\beta \dot{w}_{ac} W_f$. The core engine supplies an amount of work equal to W_F per pound of core airflow. The total work supplied by the core engine is then $w_{ac}W_F$. Since this must be equal to the work required by the fan,

$$w_{ac}W_F = w_{ac}W_f$$

$$\therefore W_F = \beta W_f$$

STEP 13A: LOCATE STATION 5 OF CORE ENGINE ON CORE h-s DIAGRAM

$HT_{4.5} = H_{t4} - Wc$
$H_{T4.5} = 413$
$H_{T5} = H_{T4.5} - W_F$
$= 413 - 104$
$H_{T5} = 309\ BTU/lb$

NOTES

The turbine which drives the core compressor is located between Stations 4.0 and 4.5. The turbine which drives the fan is located between Stations 4.5 and 5.0. Generally, the rotor speed of the core turbine is higher than the fan turbine. Why? (Think about blade tip Mach effects versus diameter.)

STEP 14A: CALCULATE CORE GAS EXIT VELOCITY

$$V_{10c}=\sqrt{2gJ(h_{T5}-h_{10})}$$

$$=\sqrt{(50,100)(309-255)}$$

V_{10c} = 1645 fps

NOTES

The two exit velocities, V_{10f} and V_{10c}, are almost identical. This means TSFC and F_n are nearly optimized for the particular β chosen.

STEP 15A: CALCULATE F_n

$$F_n=\frac{\dot{w}_{ac}}{g}[(V_{10c}-V_0)=\beta(V_{10f}-V_0)]$$

$$=\frac{170}{32.2}[(1645-389)+2(1660-389)$$

F_n 20,039 lb

NOTES

Actually we have just combined the two thrust equations

$$F_{nc}=\frac{\dot{w}_{ac}}{g}(V_{10c}-V_0)$$

$$F_{nF}=\frac{\beta\dot{w}_{ac}}{g}(V_{10f}-V_0)$$

where

$F_n = F_{nc} + F_{nf}$

STEP 16A: CALCULATE TSFC

$$TSFC=\frac{\dot{w}_f}{F_n}$$

$$=\frac{9448}{20,039}$$

TSFC = 0.47

NOTES

Here lies the beauty of the turbofan. We have increased the net thrust at absolutely no penalty in fuel. No additional fuel is required because the bypass air is not heated. Who says you can't get something for nothing?! But this is infeasible? Why?

7.7.7 COMPARISON OF THE CYCLE TURBOJET AND TURBOFAN IDEAL CYCLE ANALYSIS

The ideal cycle analysis results for the J-79 turbojet and J-79 turbofan are shown in Table 7.8.

TABLE 7.8
COMPARISON OF RESULTS

	(J-79 TURBO JET)	(J-79 TURBO FAN)	(IMPROVEMENT)
F_n (lbs)	12800	20,039	57%
$TSFC\left[\frac{lbs\text{-}fuel}{lbs\text{-}thrust\text{-}hr}\right]$	0.74	0.47	36%

You may ask, "If this much improvement can be made by just adding a fan stage, then why hasn't it been done?" It was . . . the CJ805-23A turbofan with F_n = 16,000 lb and TSFC - 0.53. But there are more problems associated with reconfiguring an old core engine (J-79 is 1956 vintage) than starting from scratch, which permits use of the latest technology in compressor and turbine design. When a turbojet aircraft is refitted with a turbofan, inlet compatibility becomes a serious problem. The inlet was designed for an airflow rate of $\dot{w}_a$. The turbofan requires an airflow rate of $(\beta + 1)\, \dot{w}_{ac}$. Since the inlet cannot be redesigned without major aircraft modifications in the case of fighter type aircraft, the retrofit is not generally practical. However, a retrofit would be practical for transport type aircraft that use engine pods.

7.7.8 COMPARISON OF TURBOJET AND TURBOFAN ENGINES

The relative merits and disadvantages of the turbofan engine are summarized in Table 7.9.

TABLE 7.9
CHARACTERISTICS OF THE TURBOFAN ENGINE
DISADVANTAGES OF TURBOFAN OVER TURBOJET

CHARACTERISTIC	SIGNIFICANCE
ADVANTAGES OF TURBOFAN OVER TURBOJET	
Bypass air is not heated	Lower TSFC
Accelerates larger air mass at a lower velocity	Yields higher propulsive efficiency at lower airspeeds
More thrust at lower airspeed	Shorter takeoff roll or higher gross weight potential
	Lower engine noise
Lower average exhaust velocity	
DISADVANTAGES OF TURBOFAN OVER TURBOJET	
Addition of fan	More mechanical complexity and bigger FOD potential
Larger mass flow rate	Relight harder in flight
Fan tip losses	Lower airspeed limit

In summary, the turbofan engine is more efficient in producing thrust for a given amount of fuel. The gas generator (compressor-combustor-turbine) produces a specified energy output. The energy can be used in many ways, but the ultimate purpose is to produce thrust. The nozzle is a means of converting a high pressure flow into a high velocity thrust. However, nozzles are not as efficient as propellers at low flight velocities as the propeller is a momentum transfer device. These trends are shown in Figure 7.33. Note that the turboprop engine is the most efficient at low velocities.

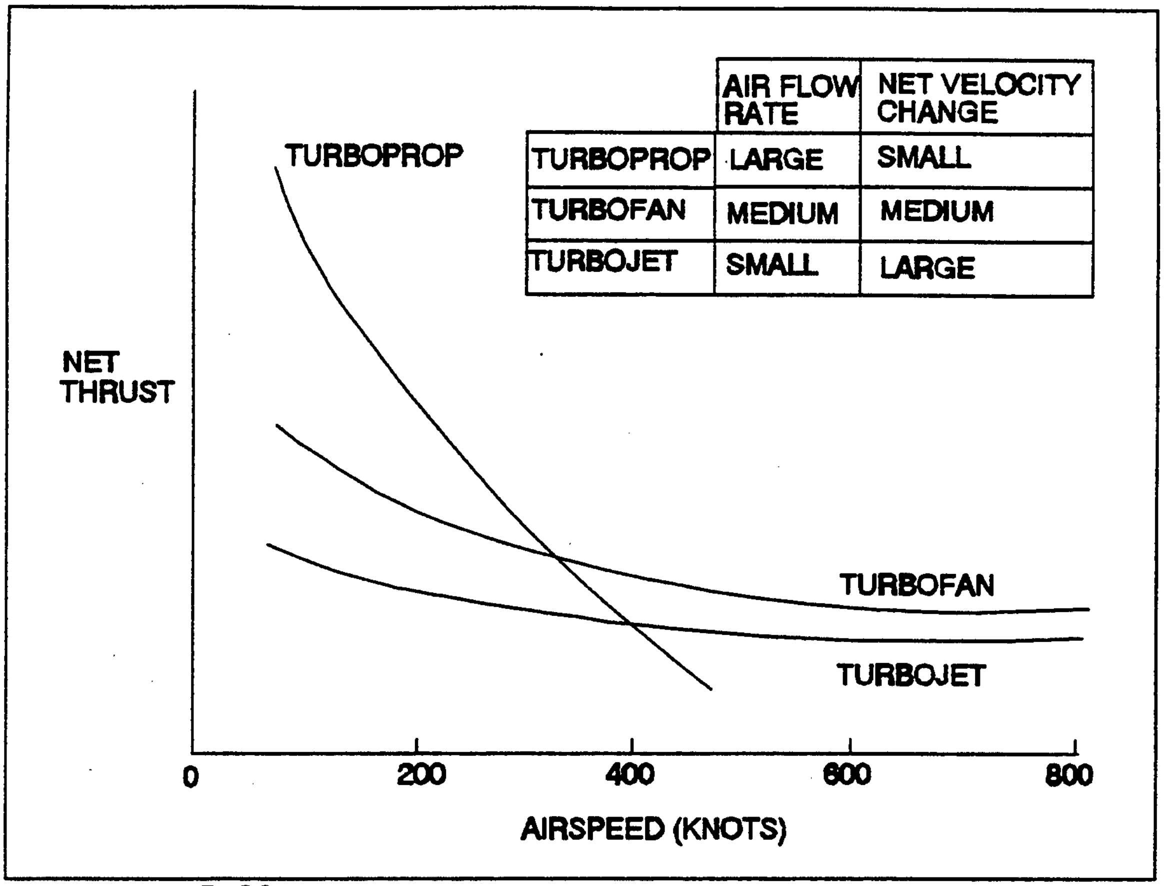


FIGURE 7.33. COMPARISON OF NET THRUST VERSUS AIRSPEED FOR THE TURBOPROP, TURBOFAN AND TURBOJET ENGINE

In most present day applications, the overall characteristics of the turbofan engine are superior to the turbojet. The biggest advantage of the turbofan, of course, is significantly more net thrust output at a lower TSFC. However, in some applications such as Mach 2 to 3 cruise, the turbojet is still employed. Sometimes the relative merits of each are about the same. The F-16 uses a turbofan engine while the YF-17 used two turbojet engines even though both aircraft were designed for identical missions!

7.8 ENGINE COMPONENTS

The physical features, functions, and performance of the major components included in the various types of gas turbine engines will be discussed in the order of their location, front to rear, on the engine.

7.8.1 AIR INLET DUCT

The engine inlet and the inlet ducting serve the function of a diffuser and furnish a relatively distortion-free, high-energy supply of air, in the required quantity, to the face of the compressor. A uniform and steady airflow is necessary to avoid compressor stall and excessive internal engine temperatures at the turbine. The high energy enables the engine to produce an optimum amount of thrust. Normally, the air inlet duct is considered an airframe part, and not a part of the engine. However, the duct itself is so important to engine performance that it must be considered in any discussion of the complete engine.

A gas turbine engine consumes six to ten times as much air per hour as a reciprocating engine of equivalent size. The air entrance passage is correspondingly larger. Furthermore, it is more critical than a reciprocating-engine air scoop in determining engine and aircraft performance, especially at high airspeeds. Inefficiencies of the duct result in successively magnified losses through other components of the engine. The inlet duct, or diffuser, has two engine functions and one aircraft function. First, it must be able to recover as much of the total pressure of the free airstream as possible and deliver this pressure to the front of the engine with a minimum loss of pressure or differential. This recovery is known as "ram recovery" or, sometimes, as "total pressure recovery." Secondly, the duct must uniformly deliver air to the compressor inlet with as little turbulence and pressure variation as possible. As far as the aircraft is concerned, the duct must hold the drag effect it creates to a minimum.

Pressure drop or differential is caused by the friction of the air along the sides of the duct and by the bends in the duct system. Smooth flow depends upon keeping the amount of turbulence as the air enters the duct to a minimum. The duct must have a sufficiently straight section to ensure smooth, even airflow within. The choice of configuration of the entrance to the duct is dictated by the location of the engine within the aircraft and the airspeed, altitude, and attitude at which the aircraft is designed to operate. A detailed discussion of the diffuser will help in understanding how the above design requirements can be met.

7.8.2 DIFFUSER

In the aeronautical sense of the word, a diffuser is a device which reduces the velocity and increases the static pressure of a fluid, such as a gas or air passing through a gas turbine engine. A diffuser operates on the principle of physics stated by Bernoulli's theorem which says that at any point in a fluid stream tube, the sum of the pressure energy, the potential energy, and the kinetic energy is a constant; that is, if one of the energy factors in a gas flow changes, one or both of the other variables must also change in order that the total energy may remain constant. Specifically, if velocity decreases, the pressure increases.

The primary purpose of the jet propulsion engine diffuser is to increase the static pressure of the free stream fluid. This function is accomplished by converting the available kinetic energy of the free stream air into a pressure rise. The basic function of a diffuser is exactly the same as the function of a mechanical compressor; thus, anything that can be done to improve the diffuser performance will benefit an engine's overall performance in the same way as an improvement in mechanical compressor design will benefit the overall engine performance. Since modern jet propulsion engines are operated, for the most part, at subsonic flight velocities, consider first the subsonic diffuser.

7.8.2.1 SUBSONIC DIFFUSER

Figure 7.34 illustrates schematically a subsonic diffuser with a simple inlet that is operating in an airstream of velocity V_0.

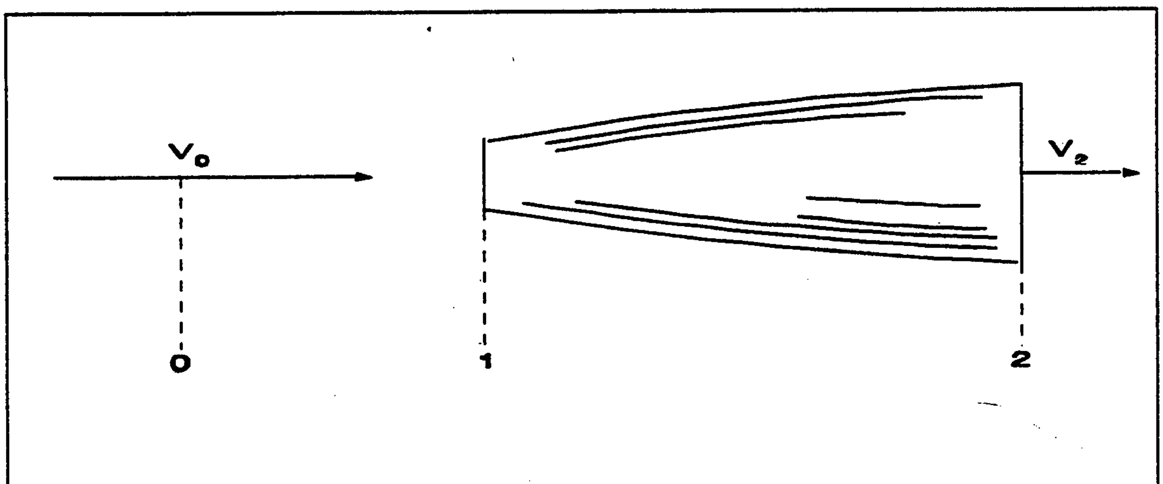


FIGURE 7.34. SUBSONIC DIFFUSER

Since the diffuser is designed to transform kinetic energy into a pressure rise, it is necessary to evaluate the total energy in the free stream and express it in terms of the total pressure that might be available for an ideal diffuser. The total-to-static temperature ratio can be expressed in terms of the flight Mach, as was done before and then the isentropic pressure-temperature relation can be applied to give the following expression for total-to-static pressure ratio

$$\frac{T_{T0}}{T_0} = 1 + \frac{\gamma - 1}{2} M_0^{\,2} \tag{7.44}$$

$$\frac{P_{T0}}{P_0} = \left[1 + \frac{\gamma - 1}{2} M_0^{\,2}\right]^{\frac{\gamma}{\gamma - 1}} \tag{7.45}$$

The total pressure in Equation 7.45 is the maximum available pressure energy that can be derived from the free stream Mach, M_0. It is the job of the diffuser to slow down the fluid so as to increase the static pressure at Station 2; however, it is still necessary to try to achieve high static pressures at the diffuser exit, and if we have an ideal diffuser, the total pressure at the diffuser exit will be equal to the total pressure in the free stream. This obviously cannot be achieved in practice because every diffuser has certain losses, primarily due to the friction that exists between the fluid and the diffuser walls. Figure 7.35 illustrates the diffuser process on a T-s plane and shows the relative pressure rise accomplished by an ideal and an actual diffuser.

In Figure 7.35, the free stream condition corresponds to Point 0. An ideal diffuser will accomplish isentropic compression to Point 2'. The actual diffuser, which has losses and therefore causes an increase in entropy, will follow a curved path from 0 to 2. The total temperature for the ideal and actual diffuser will, of course, be the same because we are assuming an adiabatic flow process (See Equation 7.44). From this diagram it is readily apparent that the total pressure in the ideal process is greater than the total pressure for the actual process, and the total energy and temperature in both processes are exactly the same for the adiabatic assumption. In the actual process, some of the available pressure energy goes into friction, which appears as heat, bringing the total temperature back to the value achieved by isentropic compression.

The losses in the diffuser are usually accounted for by an efficiency factor. In these notes the total pressure recovery factor is defined as

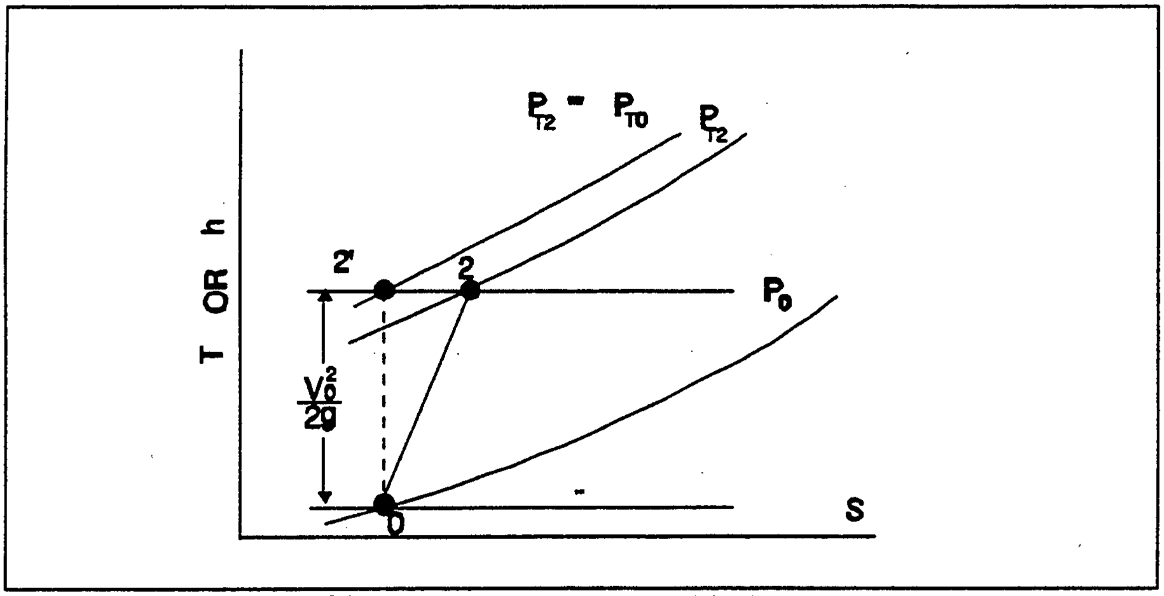


FIGURE 7.35. DIFFUSER PROCESS ON A T-s PLANE

$$\eta_r = \frac{P_{T2}}{P_{T0}} \tag{7.46}$$

where P_{T0} is the ideal free stream total pressure. This equation can be expressed in terms of a flight Mach by the use of Equation 7.45 to give

$$\frac{P_{T_2}}{P_0} = \eta_r \left[1 + \frac{\gamma - 1}{2} M_0{}^2\right]^{\frac{\gamma}{\gamma-1}} \tag{7.47}$$

Figure 7.36 presents graphically the solution of Equations 7.44 and 7.47, allowing direct determination of the pressure and temperature ratios across a diffuser for a given ηr and Mach.

For subsonic diffusers, consider only Mach less than one. It is readily apparent that the maximum pressure rise occurs when ηr = 1. Note that one curve applies to the total temperature ratio across a diffuser. This relationship is true because the total temperature, which is representative of total energy, is independent of the amount of friction in a process. Figure 7.36 is very useful for finding the total pressure at a diffuser exit when the flight Mach and diffuser ram recovery factor are known.

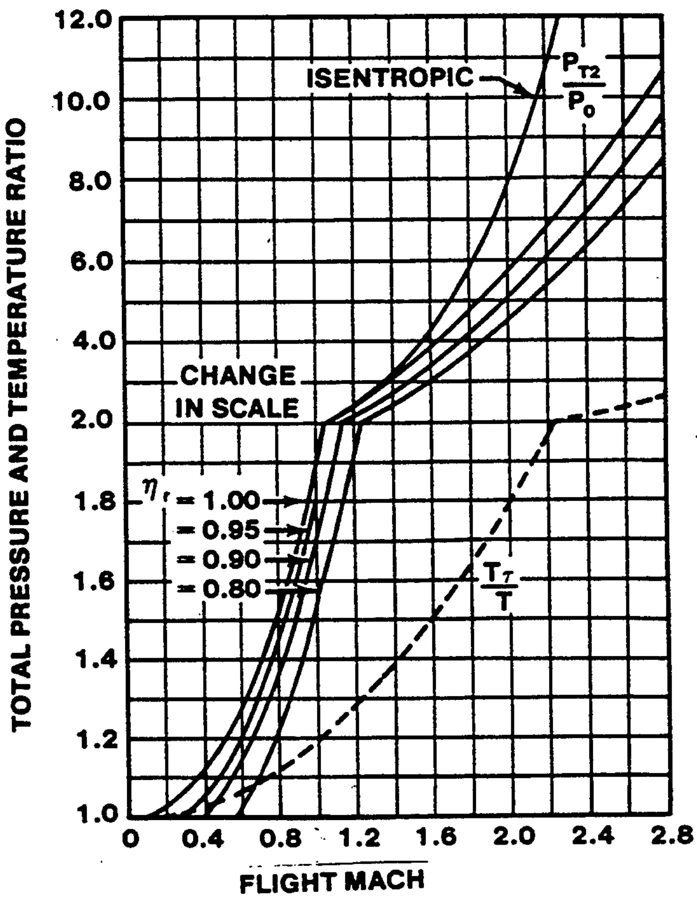


FIGURE 7.36. VARIATION OF TOTAL PRESSURE RATIO AND TOTAL TEMPERATURE RATIO FOR VARIOUS VALUES OF DIFFUSER RECOVERY FACTORS ASSUMING A NORMAL SHOCK FOR MACH GREATER THAN ONE.

Example

A turbojet engine is operated at a flight Mach of 0.6 at standard sea level conditions. If the diffuser ram recovery factor is 0.90, what is the total pressure and temperature at the diffuser exit?

Solution

From Figure 7.36 for $\eta r = 0.90$, read

$$\frac{P_T}{P}=1.148 \quad and \quad \frac{T_T}{T}=1.072$$

$$Thus \;\; P_{T2} = 1.148 \; x \; 14.7 = 16.9 PSIA$$

$$and \;\; T_{T2} = 1.072 \; x \; 520 = 557°R$$

7.8.2.2 SUBSONIC DUCT LOSSES

The fundamental causes of pressure losses in subsonic duct components are skin friction and flow separation. Skin friction is present in all flows and is the primary contributor to pressure losses in straight, constant area ducts. Flow separation losses can be much larger, however, and major effort in subsonic duct design is directed to minimizing such losses. Flow separation tends to occur when forces arise in the stream which oppose the direction of flow (adverse pressure gradient). The pressure rise in a diffuser due to the flow deceleration causes an adverse pressure gradient. Bends in the duct also produce forces that tend to separate the flow from the inner surface of the bend. The total pressure recovery for a subsonic duct is determined by analyzing the duct for each of these areas of pressure loss.

7.8.2.3 SUPERSONIC DIFFUSER

Jet propulsion devices designed to operate at supersonic Mach present an even more complex problem for the inlet designer. At these high flight speeds the available total pressure is higher, but the drag associated with the diffuser can become prohibitive. Couple these two factors with the necessity to operate well down into the subsonic flight regime for landing and it becomes difficult to fulfill the design objectives of high total pressure recovery and minimal ram drag over such a wide range of flight conditions. One method of classifying these complex diffusers is by geometry. The two basic geometric shapes are: two-dimensional and three-dimensional. Figure 7.37 shows two types of two-dimensional inlets.

The axisymmetrical inlet of the Lightening, illustrated in Figure 7.38, is typical of the three-dimensional supersonic inlet.

Another, and perhaps more useful, means of classifying supersonic inlets is according to how the compression takes place. Basically, there are three types of inlets under

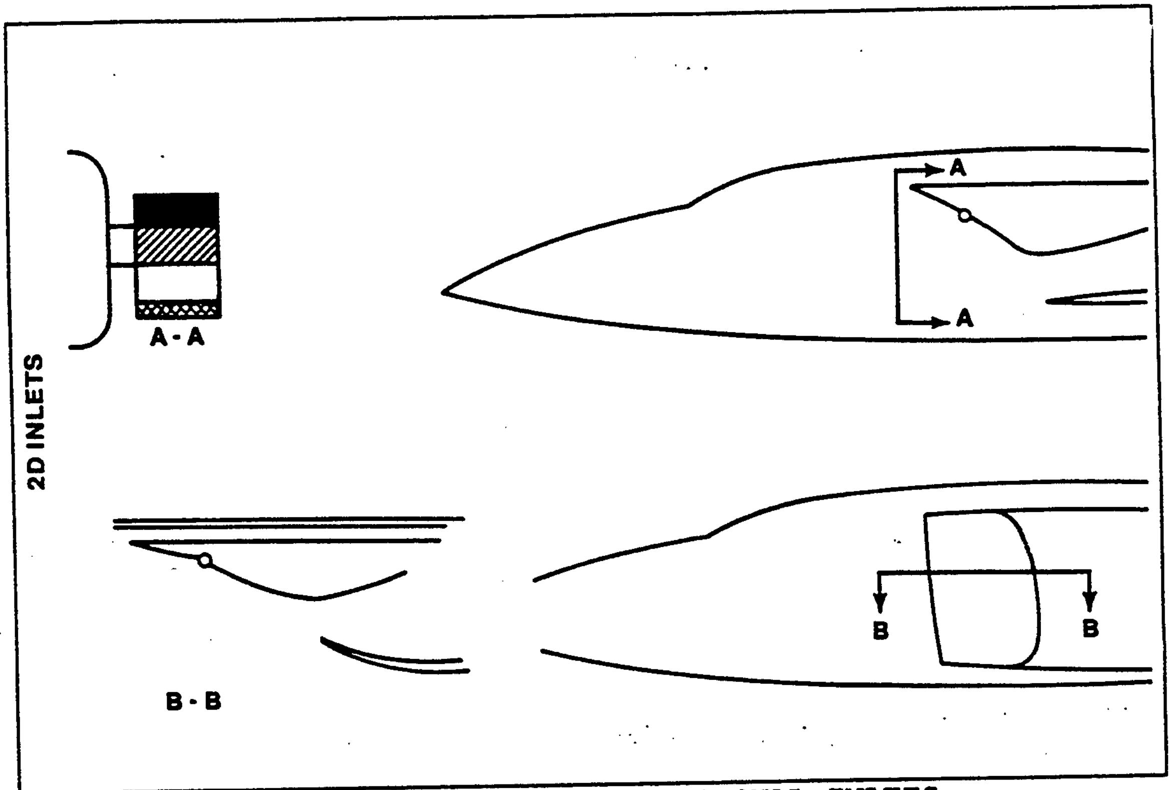


FIGURE 7.37. TWO-DIMENSIONAL INLETS

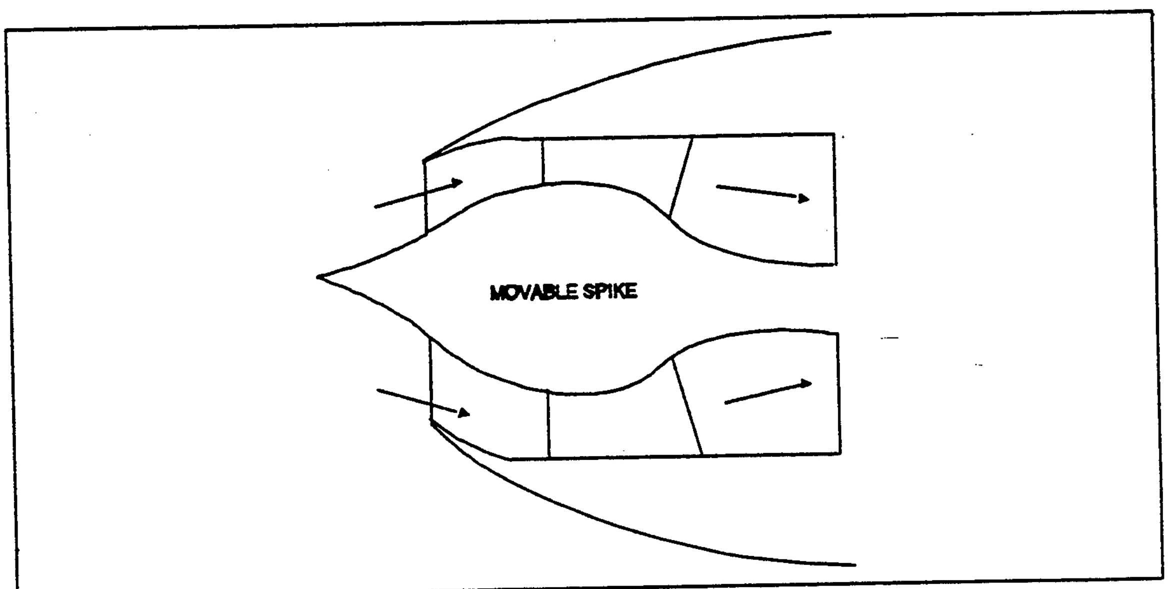


FIGURE 7.38. AXISYMMETRIC INLET

this scheme of classification: (1) normal shock inlets, (2) internal compression inlets, and (3) external compression inlets. However, mixed compression inlets, that combine internal and external compression, appear to be the most attractive design for most supersonic applications of the future.

7.8.2.3.1 Normal Shock Inlets. The normal shock inlet is very similar to the subsonic diffuser of Paragraph 7.8.2.1. The chief differences are that it operates in a supersonic flow region and the lips are usually somewhat sharper than those of a subsonic inlet. But it is simply a diverging duct, as shown in Figure 7.39 operating in supersonic conditions. In this figure, there are two distinct effects of compression: (1) the static pressure rise across the normal shock, and (2) the diffusion process which follows the normal shock.

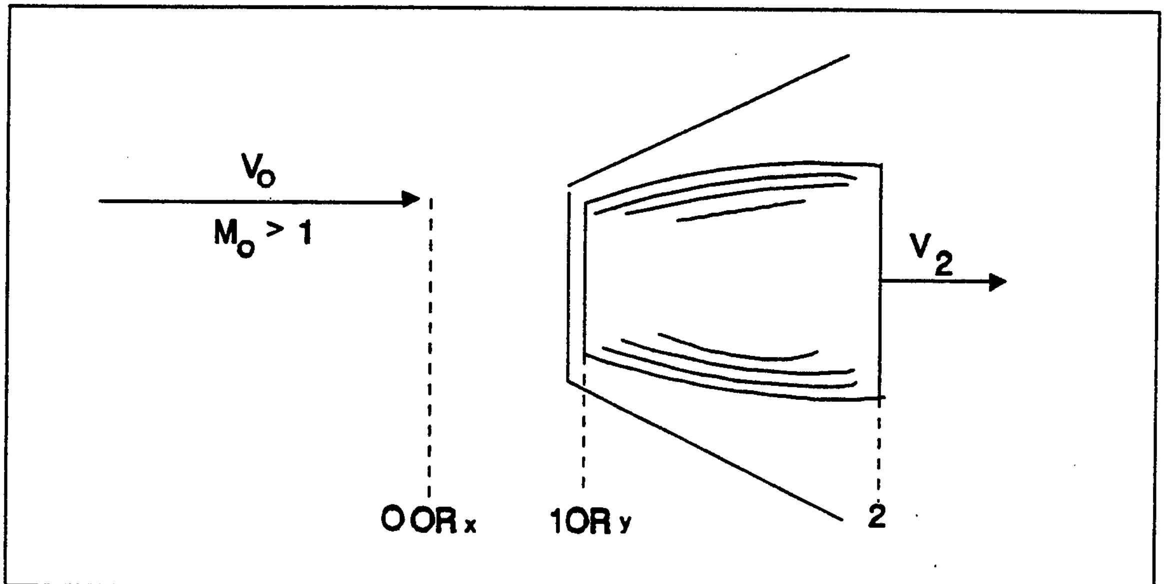


FIGURE 7.39. SUBSONIC DIFFUSER OPERATING IN A SUPERSONIC STREAM

It can be shown that the Mach after a normal shock, M_y, is always subsonic; therefore, the process from y to 2 is merely a duplication of the subsonic diffuser which has already been discussed. The pressure ratio across the normal shock can be found in any normal shock tables.

For supersonic flow in a simple diffuser, the curve for $\eta_r = 1$ is the maximum possible pressure ratio that can be achieved with a normal shock at a diffuser entrance and isentropic compression inside the diffuser after the normal shock.

Even if $\eta_r = 1$, however, the total pressure loss through the normal shock becomes prohibitive for $M_0 > 1.5$. Figure 7.40 illustrates the total pressure recovery through

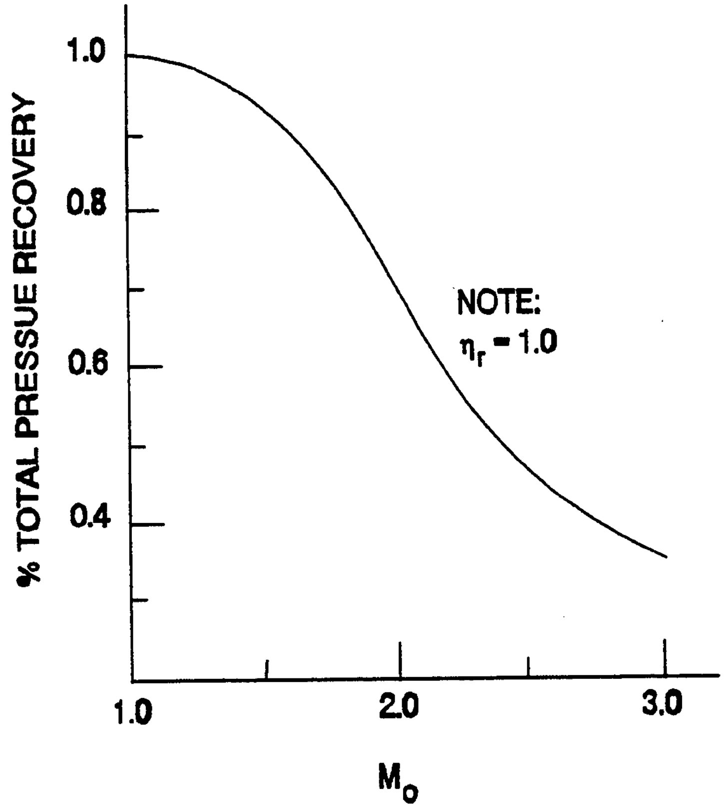


FIGURE 7.40. TOTAL PRESSURE LOSS IN NORMAL SHOCK INLET

a normal shock (with $\eta_r = 1$) as a percentage of the total pressure recovery expressed by Equation 7.45. At $M_0 = 1.5$, the total pressure recovery is 93%, whereas the total pressure recovery is 72% at $M_0 = 2$. Because of this loss in total pressure recovery, the normal shock diffuser is not used for aircraft designed to fly in excess of $M_0 = 1.5$.

The performance of the normal shock inlet deteriorates rapidly when operated at off-design conditions. If more air is required by the engine than the inlet is delivering, the flow adjustment must take place within the inlet, since pressure signals cannot move upstream of the normal shock wave. The normal shock wave is "swallowed" as shown in Figure 7.41 to make this flow adjustment.

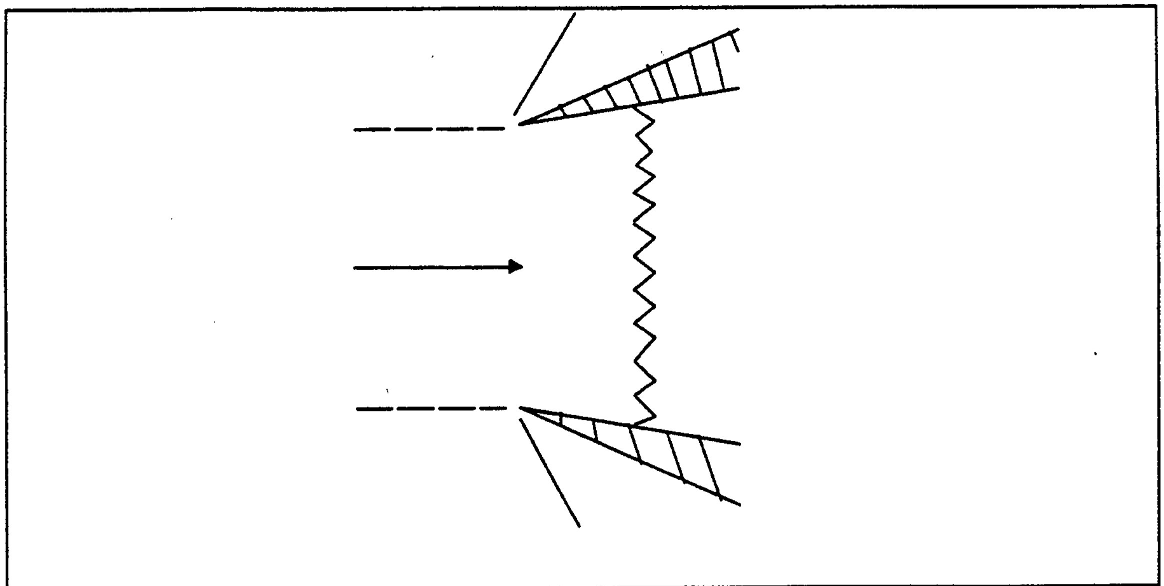


FIGURE 7.41. NORMAL SHOCK INLET WITH SWALLOWED SHOCK

In this case, though, the flow is accelerated in the diverging duct, and the total pressure recovery is reduced because of the stronger shock wave. On the other hand, if the engine requires less air than the inlet is delivering, the flow adjustment is made by a repositioning of the shock forward of the inlet lip as shown in Figure 7.42. The total pressure recovery remains high, but air is compressed by the shock wave and spilled around the inlet. This spillage causes additional drag and therefore degrades the overall engine-inlet performance. These disadvantages, along with other less important ones, force the inlet designer to look for alternatives to the normal shock inlet.

7.8.2.3.2 Internal Compression Inlets. Figure 7.43 shows three ways inlet designers have approached the problem. In this section, the internal contraction or internal compression inlet will be discussed. This type of diffuser is in essence a reversed supersonic nozzle. The convergent section up to the throat slows the supersonic flow to sonic velocity (ideally) and then further slows the flow in the diverging section. Theoretically, this type of inlet would provide very high total pressure recovery when operating at its design Mach because the compression would occur without shock waves. At off-design conditions, even this idealized inlet would suffer serious losses. If the free stream Mach is greater than design Mach, for example, a strong shock wave could develop in the divergent section downstream of the throat, giving high total pressure losses. Conversely, a strong shock wave could develop in the converging section upstream of the throat. This shock wave could

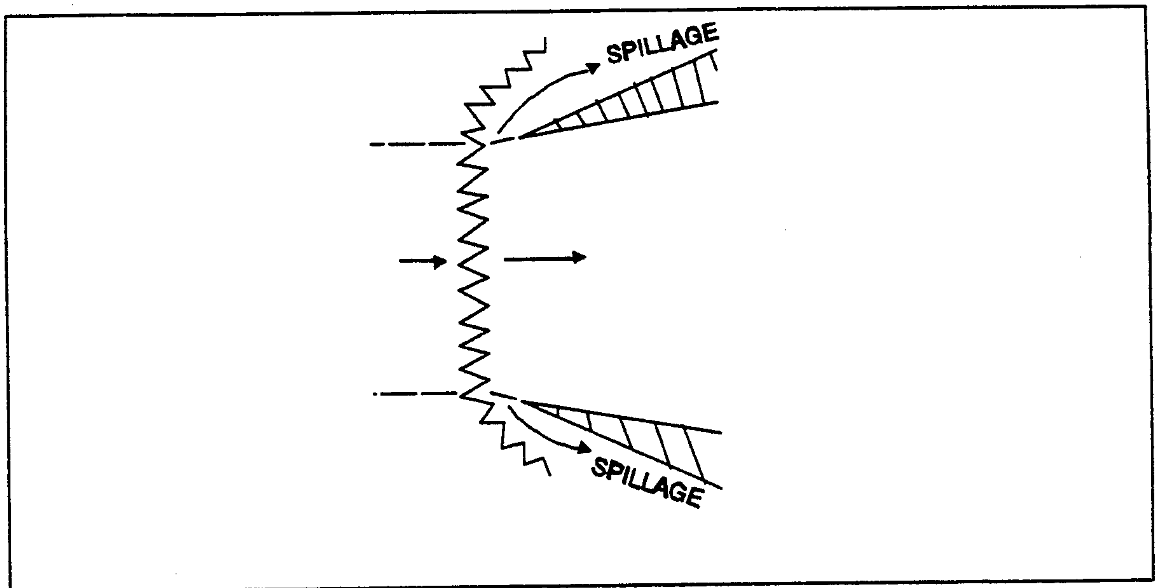


FIGURE 7.42. NORMAL SHOCK INLET WITH EXPELLED SHOCK

easily be expelled, resulting in high ram drag.

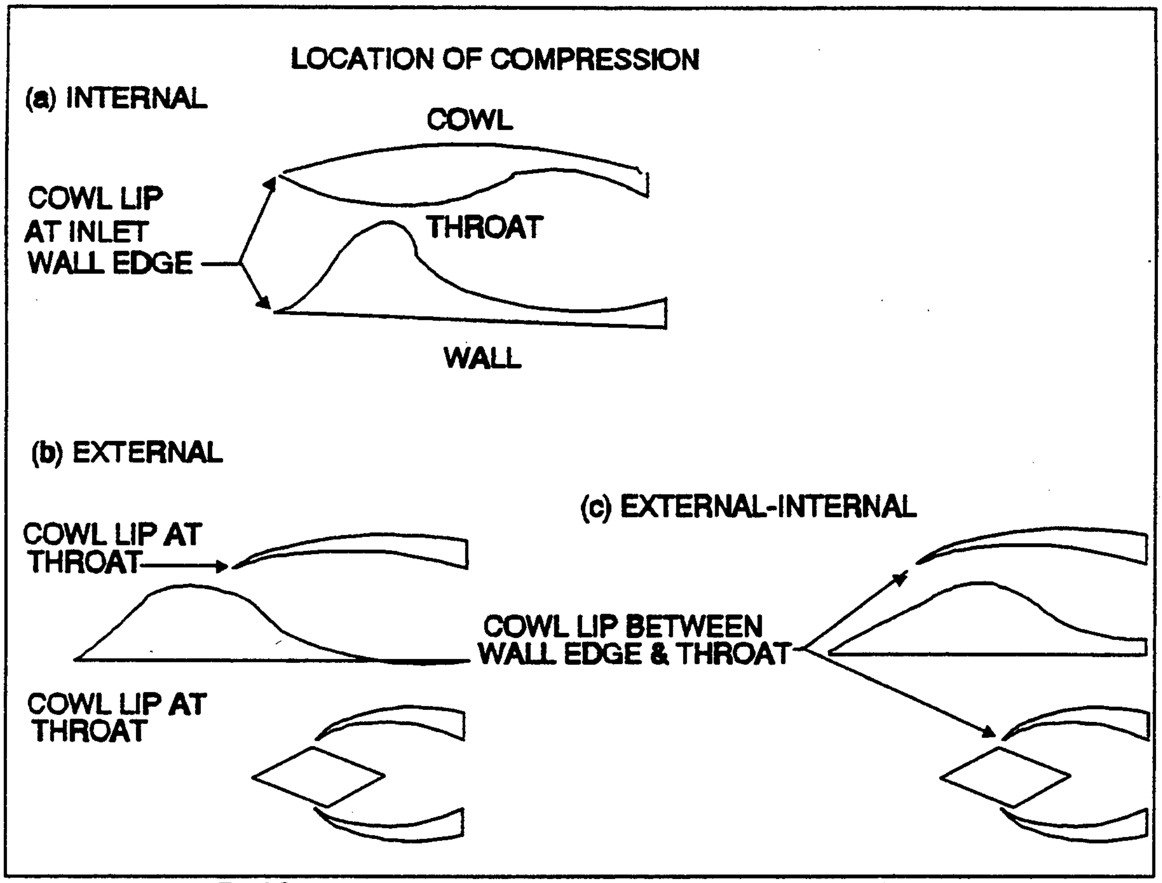


FIGURE 7.43. TYPES OF SUPERSONIC INLETS (REFERENCE)

From a practical viewpoint, the internal compression inlet has several other disadvantages. The boundary layer in such an inlet is very difficult to predict since an adverse pressure gradient exists along the length of the duct. The boundary layer thickness alters the flow area and directly affects performance of the inlet. Another problem is "starting" such an inlet. If the inlet is accelerated from subsonic speeds (as in an aircraft), the converging section will accelerate the flow. Choked flow will exist with a normal shock ahead of the throat until the design Mach is reached or exceeded, when the normal shock will be swallowed and thus disappear. The losses through this normal shock are not acceptable for an aircraft, generally, and could even prevent the vehicle from reaching the design Mach. Figure 7.44 shows one answer to the starting problem - variable geometry.

At flight velocities lower than the design Mach, the throat is enlarged for less flow restriction. As Mach is increased, the throat area is decreased, thereby allowing the inlet to function shock-free over a range of Mach.

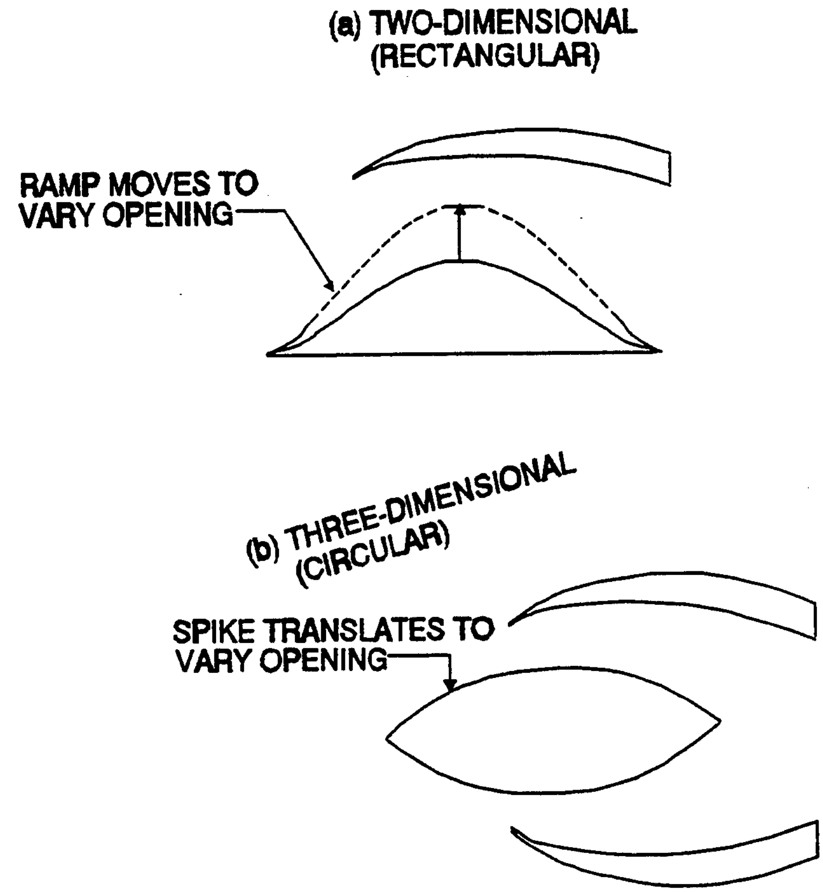


FIGURE 7.44. TWO CONCEPTS ILLUSTRATING VARIABLE GEOMETRY

7.8.2.3.3 External Compression Inlets. The problems of the internal compression inlet are such that a better solution was sought. Such aircraft as the F-8U, the F-104, and the B-58 use an inlet similar to those labelled "external compression" in Figure 7.43b. At the design operating conditions, the oblique shock wave generated by the leading edge of the compression surface (Figure 7.45) should intersect the cowl lip. The flow is slowed due to passing through the oblique shock wave and the turning of the flow to parallel the compression surface. At the cowl lip (where minimum area also occurs), a normal shock wave slows the flow to a subsonic Mach, after which it is slowed still further in the duct delivery to the compressor face.

A shock wave system utilizing two shock waves, one oblique and one normal, recovers significantly more total pressure than the single normal shock for free stream Mach greater than 1.5. Further, such an inlet avoids the starting problem since the

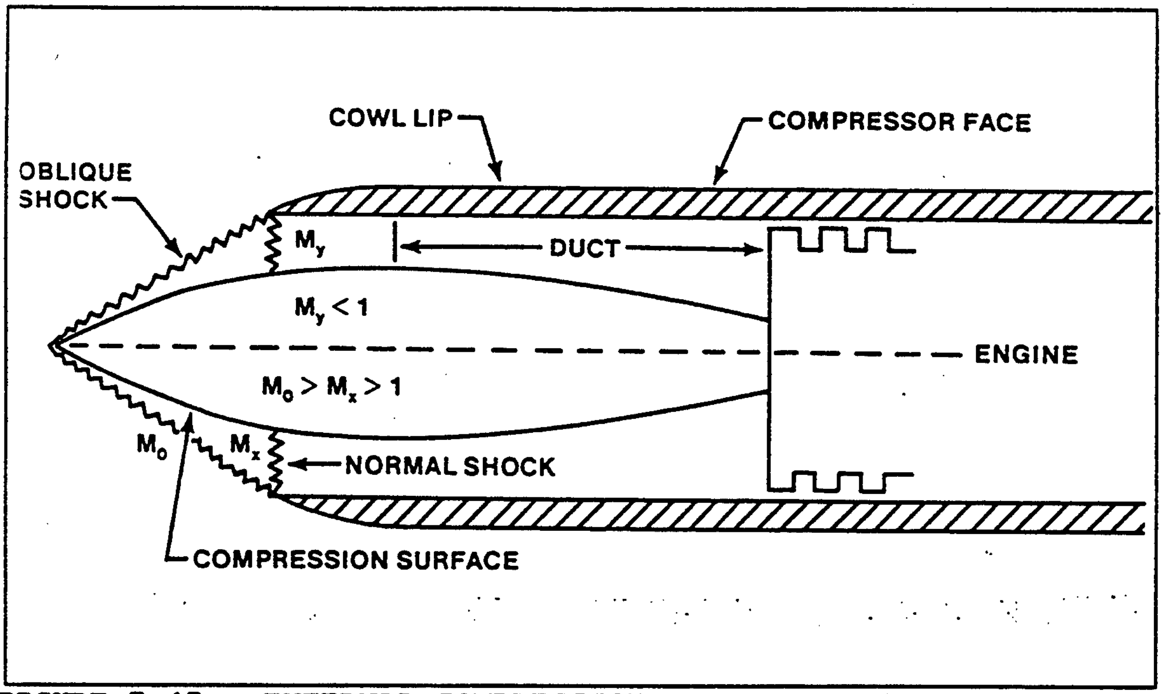


FIGURE 7.45. EXTERNAL COMPRESSION INLET AT DESIGN CONDITIONS

normal wave is forward of the cowl lip for $M_0 \leq M_{DES}$. However, this type of inlet also suffers a deterioration in performance at off-design flight conditions. Furthermore, the total pressure recovery can be increased still further by increasing the number of oblique shocks.

7.8.2.3.4 Mixed Compression Inlets. Typical examples of two-shock, three-shock, and multiple shock compression schemes are shown in Figure 7.46. The inlets utilizing a pattern of three or more shocks are frequently called mixed compression or external-internal contraction inlets. (See Figure 7.43c). These inlets capitalize on the better total pressure recovery ratios available with oblique shock waves. Figure 7.47 graphically illustrates the improvement in total pressure recovery from mixed compression inlets.

However, mixed compression inlets are susceptible to starting difficulties and may expel the normal shock (or "unstart") if the inlet is operated too far from the design conditions. Consequently, variable geometry is frequently used with mixed compression inlets. Mixed compression inlets can also be susceptible to inlet buzz or other forms of instability.

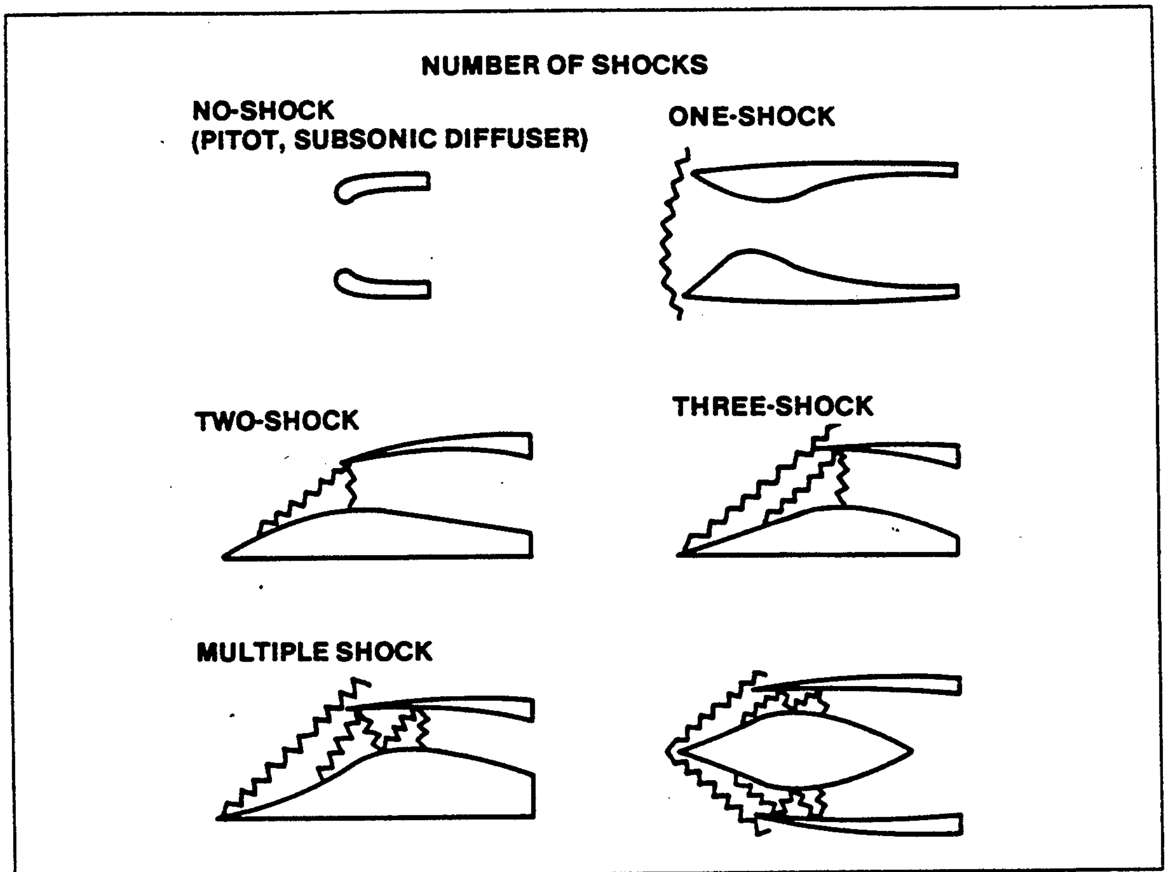


FIGURE 7.46. MULTIPLE SHOCK SYSTEMS

7.8.2.4 MASS FLOW

The criterion of diffuser performance discussed thus far has dealt solely with the ram recovery factor. This factor is important, but does not, in itself, dictate the overall performance of a diffuser. In addition to having a high ram recovery, a good diffuser must have air-handling characteristics which are matched with the engine, as well as low drag and good flow stability. For example, if a given installation had an η_r value of 0.95 for the air which it handled but supplied only 80% of the air required by the engine, it would not be a good diffuser. The importance of the airflow matching characteristics can be shown from the area considerations of Figure 7.48 which is a sketch of a typical subsonic diffuser and a typical ramp-type supersonic diffuser.

If an inlet were designed for $M_0 = M_2$, there would be no requirement for A_0 to be different than A_2. However, most of the time the subsonic inlet will be operating with M_0 greater than M_2. M_2 is a function of compressor RPM and inlet geometry, and the compressor will usually demand airflow at .3 to .4 Mach. By use of the Area Ratio

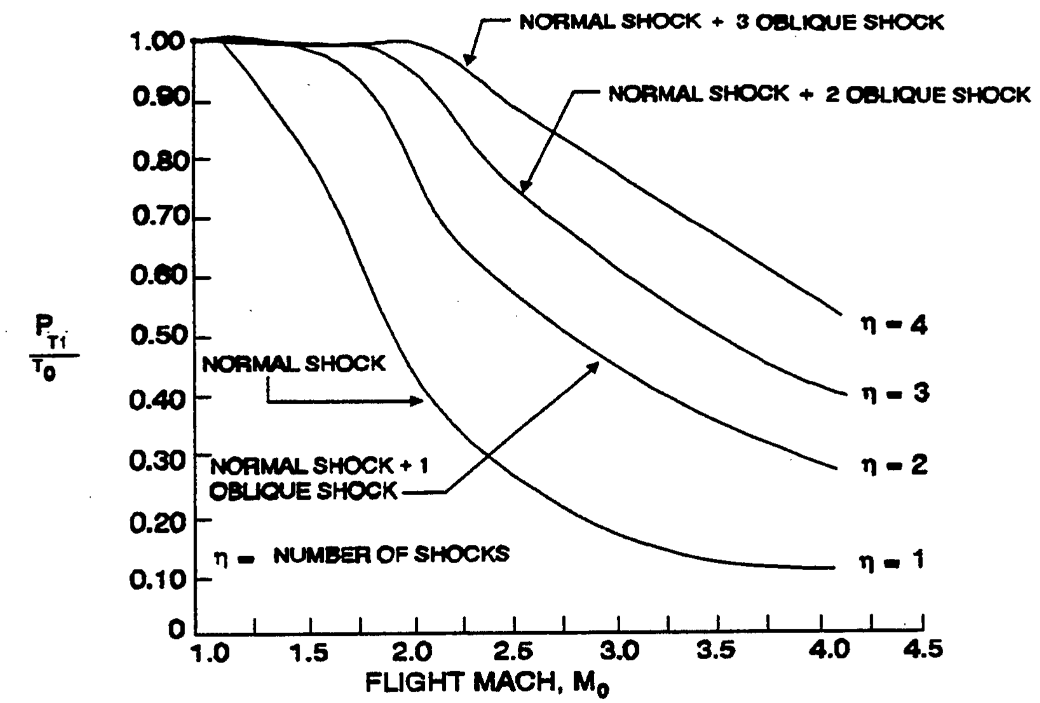


FIGURE 7.47. EFFECT OF NUMBER OF SHOCKS ON TOTAL PRESSURE RECOVERY

relation found in shock tables, Equation 7.48, which was derived in Supersonic Aero, one is able to relate inlet Mach to inlet areas.

$$\frac{A_x}{A_y} = f(M_y/M_x) \tag{7.48}$$

For a typical fixed geometry, subsonic diffuser operating on design at $M_0 = 0.8$ and $M_2 = 0.3$, the inlet would be like that shown in Figure 7.49.

For the given design conditions, $A_0 = A_1$ and $A_2/A_1 = 1.961$.

Once the area ratio of a subsonic inlet is determined for a specific design condition, the inlet contours are usually fixed, and there will exist numerous flight/engine conditions during which the inlet is operating "off-design." These off-design conditions, $A_1/A_0 \neq 1$, may affect the aircraft and/or engine performance.

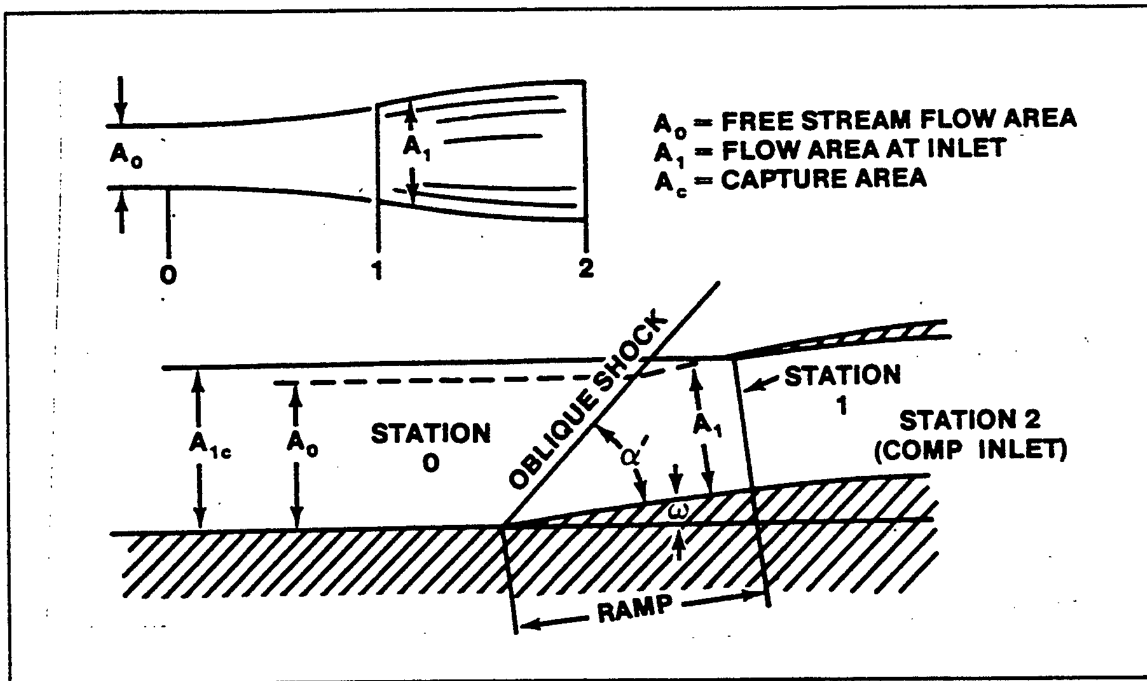


FIGURE 7.48. TYPICAL SUBSONIC AND SUPERSONIC DIFFUSERS

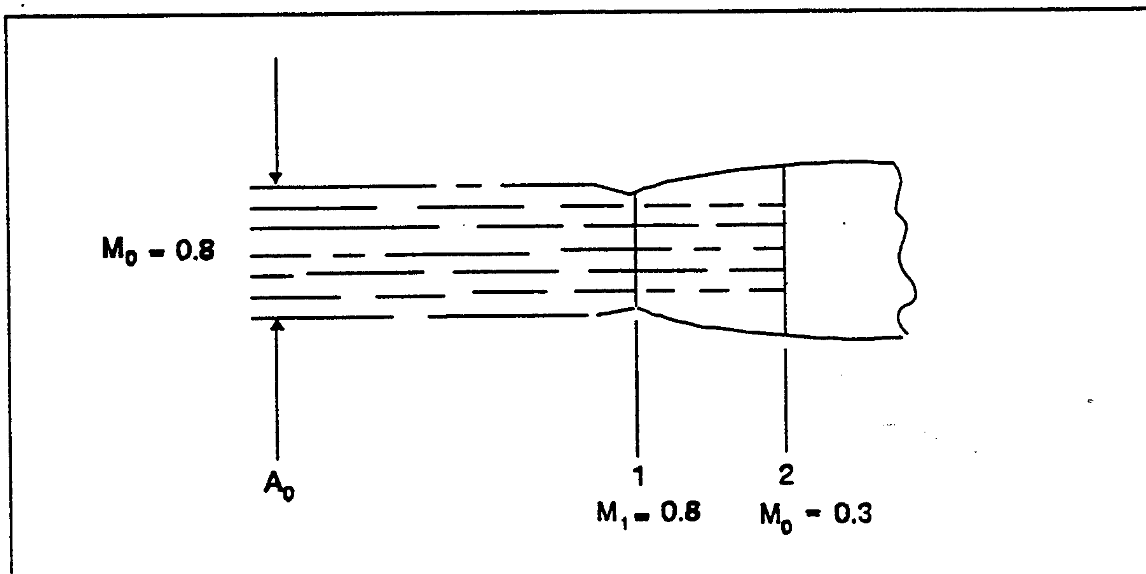


FIGURE 7.49. SUBSONIC DIFFUSER OPERATING ON DESIGN

For the fixed geometry subsonic inlet, the engine RPM will dictate the Mach that will exist at both compressor face and inlet entrance as long as $M_0 < 1.0$. For an inlet operating at a flight Mach greater than design, $A_1 > A_0$, spillage will occur and the drag increase. This off-design condition is shown in Figure 7.50a. The spillage and external flow separation will seldom affect engine operation but will adversely affect aircraft performance due to the increased drag.

If the inlet is operating off-design with the flight Mach less than design, Figure 7.50b, the engine will have to "suck in" the air. This may result in internal flow separation and can result in flow distortion at the compressor face and low pressure recovery. Distortion can cause compressor stall. The internal flow separation may become critical during the takeoff phase when the aircraft is rotated. One solution for this off design condition, Figure 7.50c, is the installation of auxiliary inlet doors, "sucker doors", near the inlet entrance to effectively increase A_1. The doors may be mechanically actuated but are usually opened automatically by the static pressure imbalance.

For a given set of operating flight conditions, the airflow requirements are fixed by the pumping characteristics of the engine, Figure 7.51. For the subsonic diffuser, if A_1 is too small to handle the air, the engine must "suck in" the lacking amount of air, resulting in a decreased ram recovery. If A_1 is too large, the diffuser will supply more air than the engine can use, resulting in excess drag because the excess air must be bypassed around the engine or "spilled" back out of the inlet. Too much air or too little air is detrimental to diffuser performance.

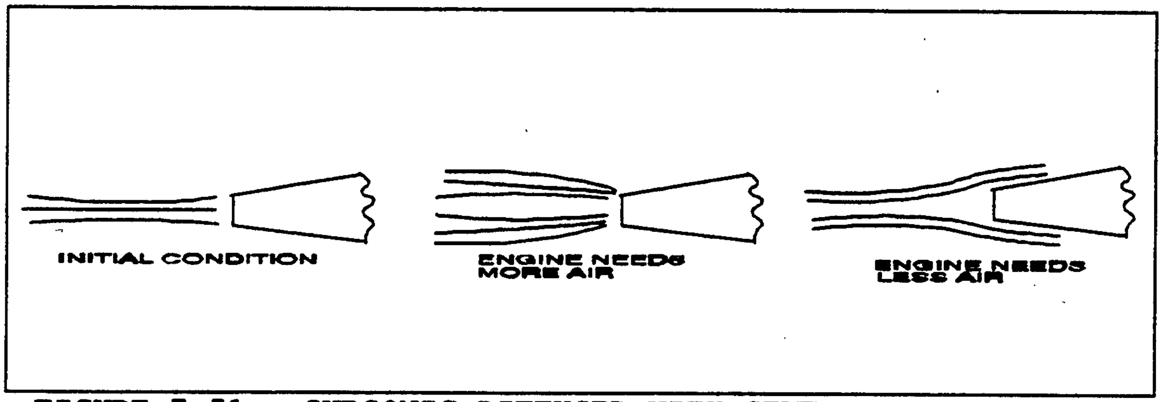


FIGURE 7.51. SUBSONIC DIFFUSER WITH SEVERAL DEMANDS FOR INLET AIR

In supersonic flow, when oblique shocks are formed, the condition is more serious because the "pressure signals" from the engine, which are sent to advise the free stream to give more or less air, cannot get to the free stream, or even to the inlet section, since supersonic velocities exist within the inlet. Consider an operating condition as shown in Figure 7.52.

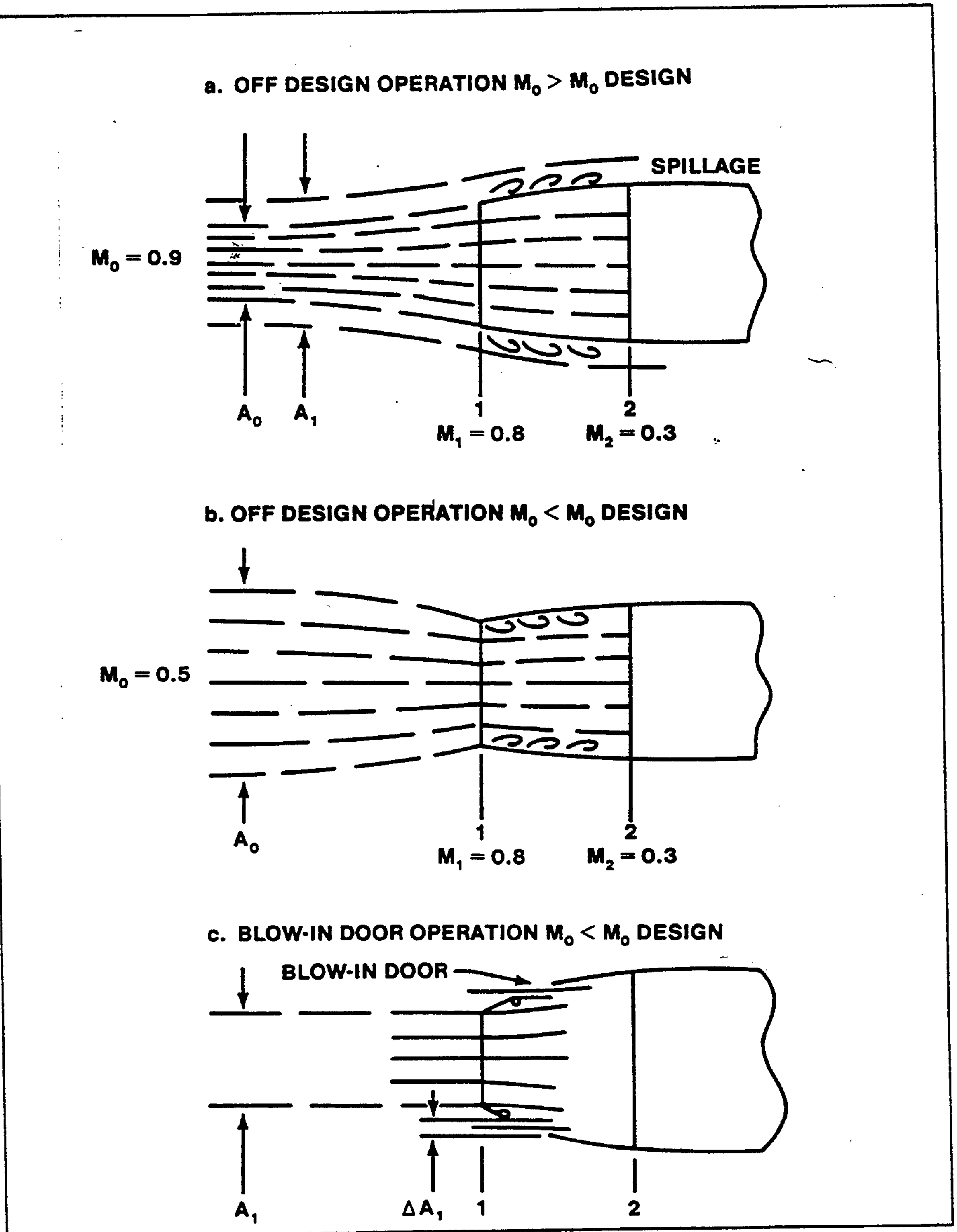


FIGURE 7.50.

Let us examine qualitatively what happens to the inlet flow characteristics when the engine demands a change in mass flow rate. If the engine demands more air than shown in the stabilized condition of Figure 7.52, it will decrease the pressure behind the normal shock and actually make that portion of the diffuser behind the normal shock act as an expanding supersonic nozzle. More shocks will occur with a consequent loss of total and static pressure. If the engine demands less air, the pressure behind the normal shock will increase and become greater than the normal

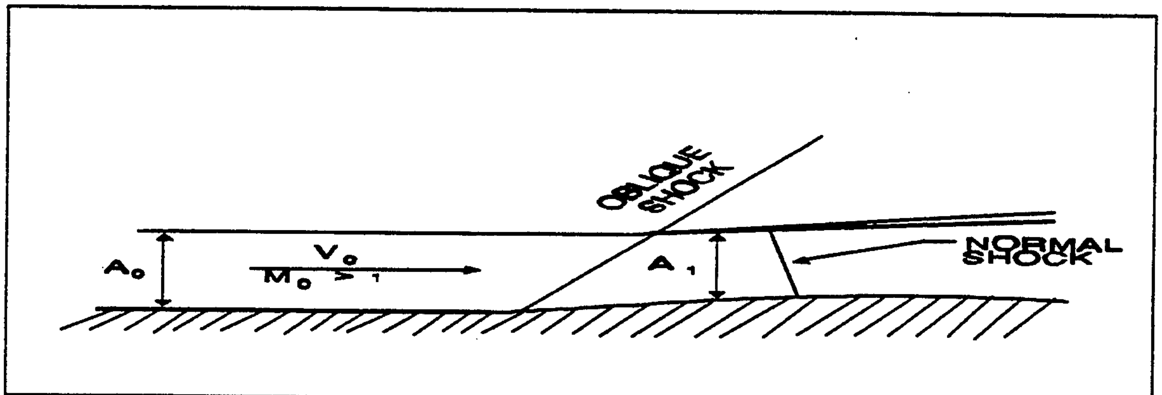


FIGURE 7.52. SUPERSONIC RAMP-TYPE DIFFUSER

shock can support for the given M_1. Therefore, the normal shock will move forward ahead of the inlet. The problem with the inlet operating under this condition is the attendant spillage and the possibility of distortion and inlet flow unsteadiness (buzz). This results because a portion of the air entering the engine has gone through an oblique shock, and a portion through a normal shock. This produces a shear layer (distortion) as shown in Figure 7.53.

Because of the significant increase in drag due to spillage and adverse engine and inlet operation due to the distortion, the shock should be attached to the inlet lip or slightly swallowed to provide efficient and stable inlet and engine operation. The normal shock can be moved to the inlet lip by one or a combination of means. If bleed doors are opened downstream of the inlet entrance but prior to the compressor, the total airflow through A_1 will increase. The bleed doors have to be scheduled as a function of both free stream and compressor Mach to maintain ideal operating conditions. Another means used to attach the shock is to increase M_1 by decreasing A1 in order to match the area to subsonic Mach after the normal shock. This could be accomplished by increasing the ramp angle, decreasing A_1 as shown by the dashed line on Figure 7.54. Caution must be exercised when increasing the ramp angle such that the oblique shock does not detach. If this occurs, a second ramp angle may be

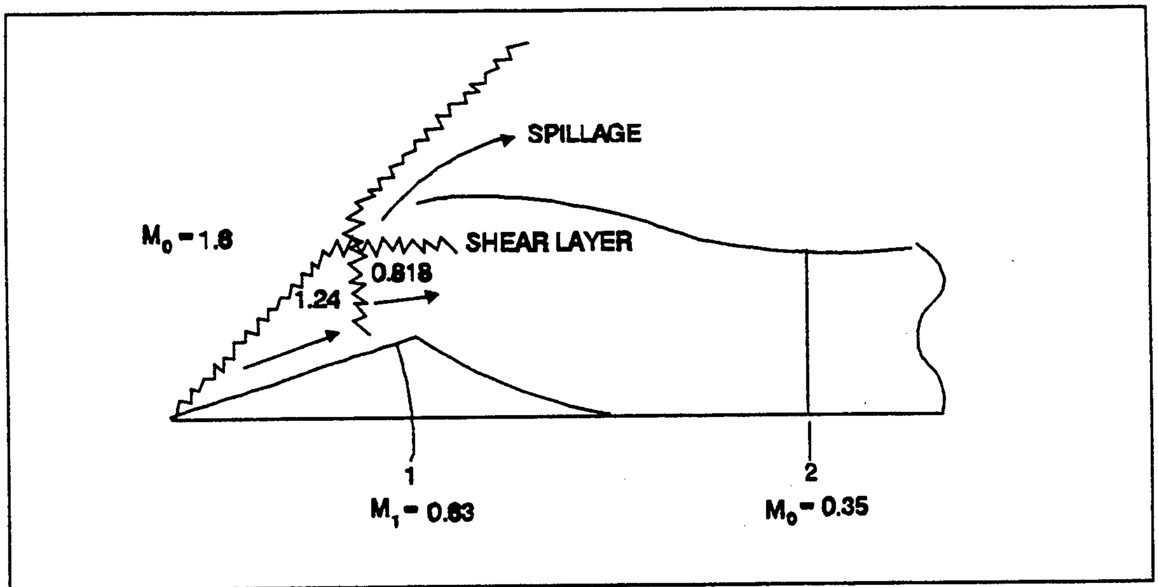


FIGURE 7.53. MULTIPLE SHOCK INLET - OFF-DESIGN

added to decrease A_1. The inlet area could also be decreased by deflecting the external lip inboard. For optimum inlet performance, the oblique shocks should intersect the lip to reduce spillage.

From the previous discussion, it becomes clear that any off-design operation produces problems. The ultimate goal would be to have a fully variable inlet such that all flight/engine conditions are on-design. The variable geometry inlets described attempt to achieve this goal but may result in other problems. Each configuration must be weighed, comparing the gains of increased thrust, reduced drag, and stable inlet/engine operation to the penalties of cost, complexity, weight, and reliability.

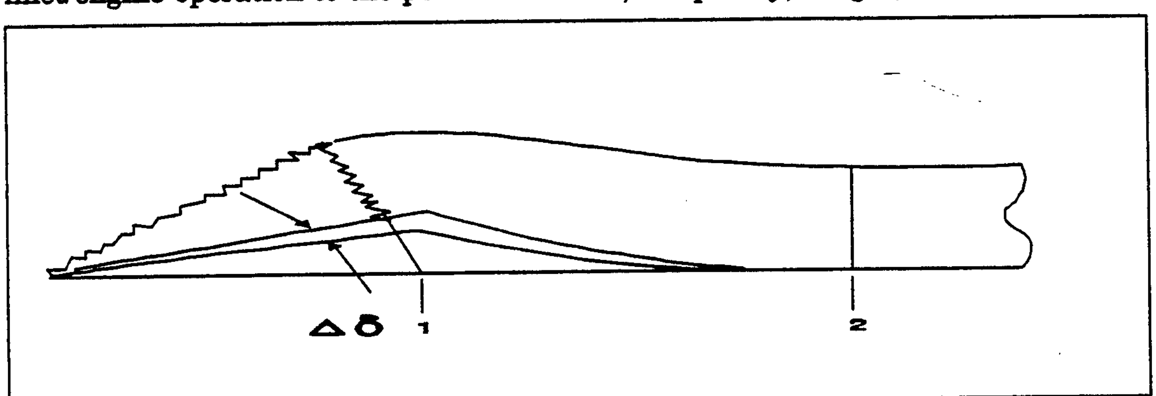


FIGURE 7.54. ADJUSTING INLET AREA WITH VARIABLE RAMP

7.8.2.5 MODES OF SUPERSONIC DIFFUSER OPERATION

It has been pointed out that the performance of a supersonic diffuser is as much a function of the mode of operation as it is of other factors; therefore, it is appropriate to examine some typical supersonic inlet operating modes. The three basic modes of operation frequently referred to are subcritical operation, critical operation, and super critical operation. These three types of operation are shown for a conical inlet in Figure 7.55.

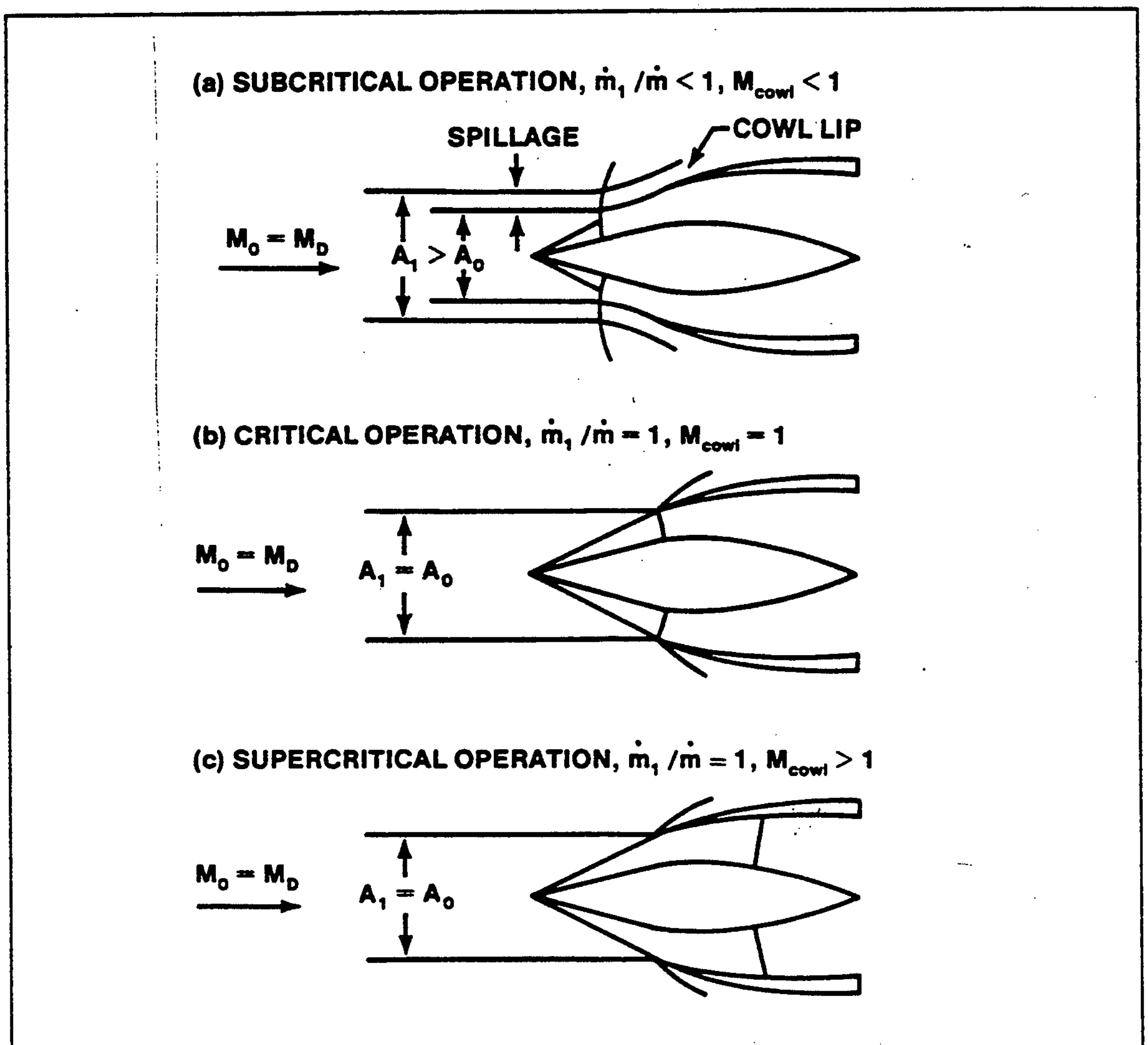


FIGURE 7.55. A CONICAL INLET AT ZERO ANGLE OF ATTACK AND DESIGN MACH SHOWING THREE MODES OF OPERATION

All three inlets are operated at the design Mach which means that the conical shock or conical shock extended will intersect the cowl lip. Figure 7.55a illustrates subcritical operation where the normal shock is external and subsonic velocities exist at the cowl. For this condition spillage exists, and the inlet is not swallowing air at maximum capacity. Pressure recovery is low since some of the air goes through a single, near normal shock. Operation is generally unstable and conducive to a condition called "buzz" (normal shock moves in and out of the inlet at relatively high frequencies). In general, subcritical operation is unsatisfactory and should be avoided. As the flow resistance downstream of the diffuser is increased, the mass flow can be reduced to its limit value of zero at which point no flow exists. When, for the same design Mach, the downstream flow resistance is decreased, perhaps by increasing the RPM and pumping action of turbojet engine compressor, the mass flow will increase and the normal shock will move downstream. At one unique condition, it will be located at the cowl lip, Figure 7.55b. This condition illustrates critical operation. As the normal shock moves downstream from its location in Figure 7.55a to that in Figure 7.55b, the ram recovery also increases because more of the shock through which the entering air passes becomes oblique. For critical operation, both maximum mass flow and ram recovery are attained for the design Mach; thus, this condition represents the optimum performance condition. It has the disadvantage, however, of being marginally unstable in actual applications because small changes in angles of attack, yaw, or boundary layer separation ca induce the critical mode of operation across the threshold into the subcritical regime. Consequently, for actual operation it is usually better to operate the inlet in a more stable condition, the super critical operation, there are various degrees of super critical operation, the better operation in this regime being where the normal shock is far enough downstream form the throat to produce stable operation but not excessively far downstream to lower the ram recovery factor to unacceptable values.

The three modes of inlet operation discussed above can and do occur for other than design Mach operation. In accelerating to the design Mach, for example, the conical shock will lie outside the cowl lip, and depending on the mass flow required by the engine for a given Mach, operation can be either subcritical or super critical.

7.8.2.6 OTHER SUPERSONIC DIFFUSER PERFORMANCE PARAMETERS

As mentioned previously, the design of induction systems for aircraft involves a number of compromises in order to obtain the optimum arrangement. Usually the compromises are directed toward maximizing the thrust minus drag and minimizing the weight of the aircraft insofar as the induction system influences it. In addition to the factors already mentioned (ram pressure) recovery, additive drag, and inlet engine airflow matching), there are a number of other inlet parameters which also influence this optimization. Included are the following: (1) the effect of boundary layer; (2) inlet flow stability; (3) inlet flow distortion; and (4) the static and low speed losses of the sharp lips which are required for low drag at high speed.

The boundary layer influences the performance of a supersonic inlet in several ways. These include friction losses, the displacement effect of the boundary layer on the compression field, and shock-boundary layer interaction. Friction losses are similar to the losses in subsonic diffusers. For inlets located adjacent to the fuselage, the boundary layer buildup on the fuselage presents a potential additional loss in that the low energy air of the fuselage may be inducted, significantly reducing the diffuser pressure recovery. For this reason, it is normal practice to move the inlet from the fuselage and to provide a boundary layer removal duct which prevents the low energy air from entering the main induction system. Even though this increases the frontal area of the aircraft and the total aircraft drag, the improvement in inlet recovery more than balances the extra drag. Figure 7.56 shows the improvement in total pressure recovery with increasing depth of the boundary layer removal diverter for a double conical side fuselage inlet.

From Figure 7.56, it may be seen that increasing the depth of the boundary removal diverter increases the inlet pressure recovery to the point that the depth of the diverter equals the boundary layer thickness. Converting the relation in Figure 7.56 into thrust minus drag shows that the overall aircraft performance also increases to $h/\delta = 1$.

Probably the greatest effect of the boundary layer on supersonic diffusers is in the area of shock-boundary layer interaction. Both the shock wave and the boundary layer can be significantly modified by this interaction, depending upon the strength of the shock and the boundary layer type (laminar or turbulent).

In extreme cases, the interaction can result in a separation of the boundary layer. Obviously, such extremes should be avoided in supersonic inlets when high pressure recovery is desired. Two methods can be used to control the interaction and prevent separation. The first is to bleed off the boundary layer. A boundary layer removal duct similar to that shown in Figure 7.56 can thus serve two purposes: to prevent low energy air from entering the inlet and to minimize shock boundary layer interaction. Where shocks occur internally, the boundary layer can be bled off though flush scoops. Where such boundary layer removal in not possible, separation can still be prevented by maintaining the strength of the intersecting shock below the critical value.

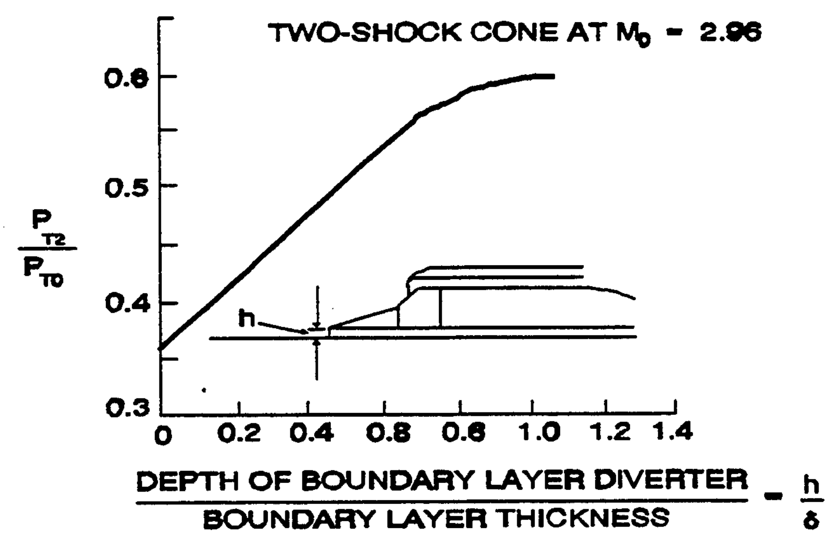


FIGURE 7.56. EFFECT OF BOUNDARY LAYER REMOVAL DEPTH ON TOTAL PRESSURE RECOVERY OF FUSELAGE SIDE INLET

Supersonic diffuser flow stability is another important inlet parameter. Flow instability, often called inlet buzz, is a very complex phenomenon associated with the subcritical region of inlet operation. Buss manifests itself a large and often violent flow pulsations or fluctuations which can occur randomly or regularly. There are a number of apparent causes of the instability, but no final explanation has yet been developed. Inlets which are at all prone to buzz are normally operated super critically even at the expense of pressure recovery in order to avoid any possibility of encountering the buzz phenomenon.

Another important inlet parameter is the flow uniformity at the exit of the diffuser (at the engine inlet). Nonuniform flows at this station can significantly affect the engine performance and hence the aircraft. Such nonuniformity or distortion causes a significant reduction in compressor stall margin, decreases the internal performance of the engine, and increases the vibratory stresses on the compressor blades. Although distortion can be either radial or circumferential, the latter appears to be the more important with respect to compressor stall margin reduction.

There are a number of ways in which flow distortion can be minimized with respect to diffuser design. These include minimizing bends in the duct, low diffusion angles, and adequate boundary layer removal. Aircraft attitudes, such as very high angles of attack or yaw, will also tend to increase the distortion. Probably the best way of reducing the distortion, regardless of other effects, is to incorporate as long a straight cylindrical section as possible ahead of the engine.

In the interests of low drag at high Mach, the lips of supersonic inlets must be as thin and sharp as possible. For low speed or static operation, however, high lip angles of attack occur and flow separation often occurs on the inside of the lips, resulting in high pressure losses. These losses can be appreciable for a choked inlet (21% statically). Additional losses will occur in the diffuser downstream of the inlet.

7.8.3 COMPRESSORS

The combustion of fuel and air at normal barometric pressure will not produce sufficient energy to enable enough power to be extracted from the expanding gases to produce useful work at reasonable efficiencies. The compressor provides increased air pressure as is needed to increase the efficiency of the combustion cycle.

Finding a satisfactory manner in which to accomplish this necessary compression phase of the gas turbine cycle constituted the main stumbling block during the early years of turbojet engine development. Great Britain's Sir Frank Whittle and Germany's Hans Von Ohain solved the problem by using a compressor of the centrifugal type. This form of compressor is still being used successfully in many of the smaller gas turbine engines today. However, the engine efficiency levels with single-stage centrifugal compressors are somewhat better, but still do not compare with those of axial flow compressors. A compression ratio of 8 to 1 is about the maximum capability of single-stage compressors. Axial compressors were first introduced by Dr. Anselm Franz, an Austrian working for Germany in 1939. With multiple stages, axial compressors produce much higher pressure ratios. A high efficiency dual axial compressor, for instance, can attain a ratio 23 to 1, or better.

Axial compressors have the added advantages of being more compact and presenting a relatively small frontal area, which are important features in a high speed aircraft engine. Therefore, most large fan and turbojet engines employ this type of compressor.

7.8.3.1 GENERAL THERMODYNAMIC ENERGY ANALYSIS

Before examining details of specific compressors, consider a general energy analysis of the compression process. Figure 7.57 depicts a mechanical compressor which takes in its fluid at Section 2, does work on it, and delivers the fluid at a high pressure level at Section 3.

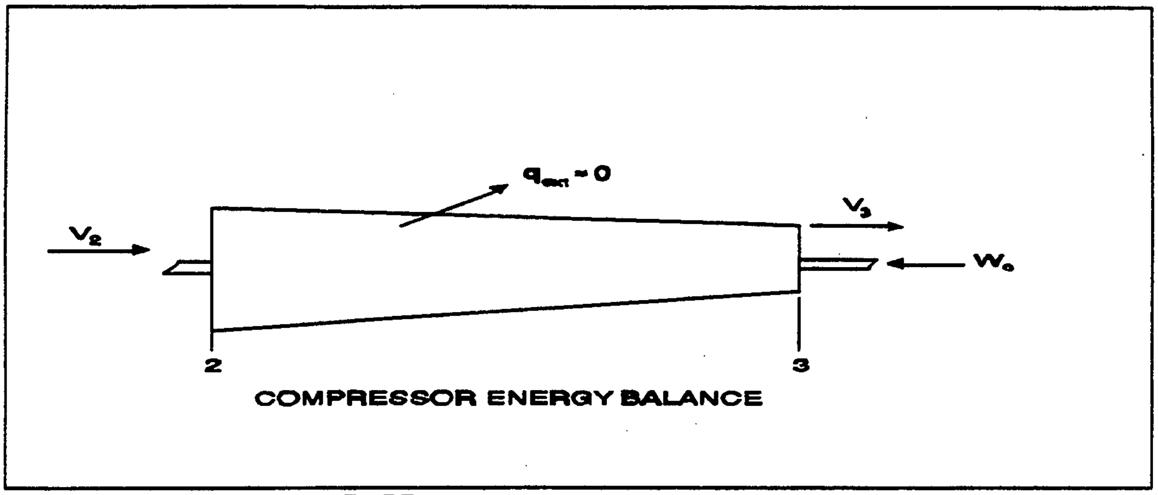


FIGURE 7.57. COMPRESSOR ENERGY BALANCE

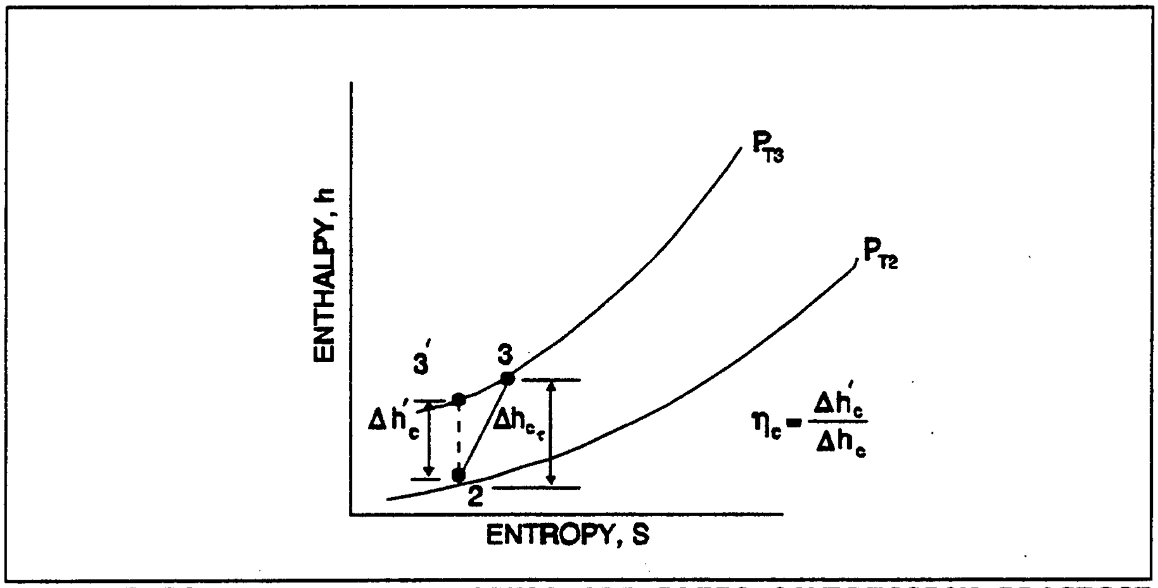


FIGURE 7.58. IDEAL AND ACTUAL ADIABATIC COMPRESSION PROCESSES

An energy balance on this machine considers the various energy terms which are applicable between entrance and exit. The general energy equation, when written between these sections, is as follows

$$z_2+\frac{V_2^2}{2g}+h_2-q_{ext}+W_c = z_3+\frac{V_3^2}{2g}+h_3 \tag{7.49}$$

Since in most compressors the difference in potential energy is negligible and the amount of heat gained or lost from the machine is very small in comparison to the amount of work which is delivered to the machine, a logical assumption is that the process is without change in potential energy, and that it is adiabatic; thus

$$\Delta z = q_{ext} = 0 \tag{7.50}$$

By these assumptions, the work required by a compressor can be written

$$W_c = \frac{V_3{}^2 - V_2{}^2}{2g} + h_3 - h_2 \tag{7.51}$$

Note: V_3 will usually be less than V_2 in practice.

It is usually more convenient to consider the total enthalpy at exit and entrance in order that one need not measure the velocity at these sections. When Equation 7.57 is written in terms of total conditions, it takes the following form:

$$W_c = h_{T_3} - h_{T_2} = \Delta h_c \tag{7.52}$$

For constant specific heats, Equation 7.46 can also be written as

$$W_c = C_p[T_{t_3} - T_{T_2}] = C_p \Delta T_c \tag{7.53}$$

Equations 7.52 and 7.53 merely state that the work required by the compressor is equal to the total energy delivered to the fluid.

The energy absorbed by the fluid is reflected in a change of velocity, temperature, and pressure, and since the function of a compressor is primarily to increase the pressure of its fluid, all efforts are made to make the major portion of the energy transfer reflect as a change in pressure. Since pressure increase is the function of a compressor, it is desirable to establish an efficiency factor that is based on the pressure ratio of the machine. Figure 7.58 shows two adiabatic compression processes on an h-s diagram between Pressures P_{T2} and P_{T3}.

For the assumption of the adiabatic flow process, we can state that there are in general two types of compression between Pressures 2 and 3 namely, the isentropic process and the general adiabatic process with friction. Since the pressure lines diverge with increasing entropy on an h-s diagram, it is readily apparent that the most efficient adiabatic process of compression is an isentropic path which is depicted as Line 2-3'. The adiabatic process with friction involves an increase in entropy and is shown as Line 2-3. The amount of compression work for either process is given by Equation 7.52, and the amount of work is equal to the actual enthalpy change of the fluid. An actual machine requires more work (because of frictional losses) to accomplish the same pressure rise than an ideal machine. The ratio of enthalpy change of an ideal adiabatic process to the enthalpy change of an actual adiabatic process is defined as the compressor adiabatic efficiency; thus, we can write

$$\eta_c = \frac{\Delta h'_{c_t}}{\Delta h_{c_t}} \tag{7.54}$$

It should be noted that this efficiency is a comparison between an ideal machine operating between the same total pressure limits.

Another useful form in which the compressor work can be expressed is as the shaft horsepower absorbed by the compressor.

$$HP_c = \frac{\Delta h c t \dot{w} \text{ x } J}{550} \tag{7.55}$$

7.8.3.2 CENTRIFUGAL COMPRESSORS

The centrifugal compressor was the type utilized in our first turbo-jet-powered airplane, and it has long had other applications. Its early engineering uses lay in pumping water and other liquids. Turbo-supercharged aircraft engines utilize a centrifugal compressor as the supercharger; many of the earlier turbojet engines and even some current ones use the centrifugal compressor. The more important advantages of the centrifugal compressor are that it produces a large pressure ratio for a single stage of compression, and it is easily manufactured.

Centrifugal compressors operate by taking in outside air near their hub and rotating it by means of an impeller. The impeller, which is usually an aluminum alloy forging, guides the air toward the outer circumference of the compressor, building up the velocity of the air by means of the high rotational speed of the impeller. The compressor consists of three main parts; an impeller, a diffuser and a compressor manifold (Figure 7.59). Air leaves the impeller at high speed and flows through the diffuser, which converts high-velocity kinetic energy to a low-velocity, high-pressure energy. The diffuser also serves to straighten the airflow and to turn the air so that it may be picked up by the compressor manifold which acts as a collector ring. The diffuser blades direct the flow of air into the manifold at an angle designed to retain a maximum of the energy imparted by the impeller. They also deliver air to the manifold at a velocity and pressure which will be satisfactory for use in the burner section of the engine.

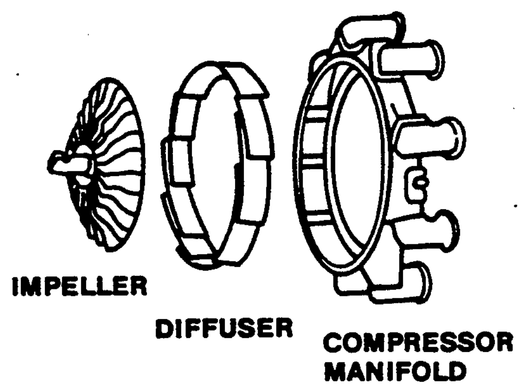


FIGURE 7.59. COMPONENTS OF A CENTRIFUGAL COMPRESSOR

The compressor shown in Figure 7.59 is known as a single-face or single-entry compressor. A variation of this is the double-face or double-entry compressor in which the impeller is constructed as shown in Figure 7.60. The double-face compressor can handle the same amount of airflow and has a smaller diameter than a single-face compressor. This advantage is partially offset by the complications involved

in delivering air from the engine inlet duct to the rear face of the impeller. Double-entry centrifugal compressors must have a plenum chamber to enable the incoming air to be collected and fed to the rear impeller. Plenum chambers are, in essence, air chambers in which the compressor-inlet air is brought to low velocity after having passed through the inlet duct of the aircraft. This air is brought in at ambient

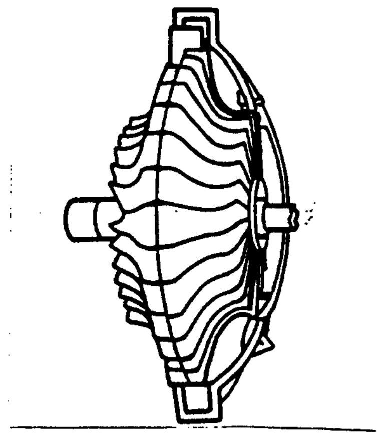


FIGURE 7.60. DOUBLE-ENTRY COMPRESSOR IMPELLER

pressure plus ram pressure. The pressure in the plenum chamber is, therefore, greater than that of the outside atmosphere. The plenum chamber is actually a diffuser that acts as a tool which the rear impeller is able to receive its air supply.

Multistage centrifugal compressors consist of two or more single compressors mounted in tandem on the same shaft (Figure 7.61). The air compressed by the first stage is passed on to the second stage at its point of entry near the hub. This stage further compresses the air before passing it on to still another stage, if there is one. In compressors of this type, the greatest difficulty is encountered in turning the air as it is passed from one stage to the next.

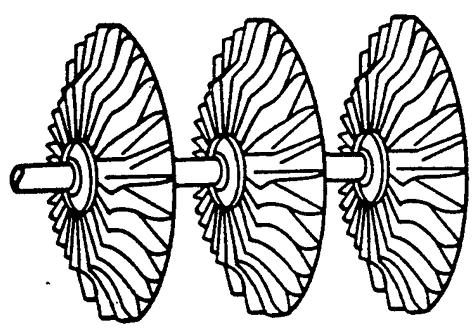


FIGURE 7.61. MULTISTAGE CENTRIFUGAL COMPRESSOR

7.8.3.3 AXIAL FLOW COMPRESSORS

During the early development of turbojet engines, it was realized that the centrifugal flow compressors would impose certain performance limitations upon the high-thrust engines of the future. Consequently, the axial flow compressor development program was initiated early in turbojet engine history. This is borne out by the fact that the first all American turbojet engine was the 19A engine, an axial flow compressor engine designed and constructed by Westinghouse. As mentioned previously, the pressure ratios attainable in centrifugal flow compressors is about 8:1 (unless multistaging is employed, resulting in multiple air turning problems) at an efficiency of about 70 to 85%. The axial flow compressor, however, can achieve a much higher pressure ratio at a high level of efficiency; thus, where high pressure ratios are required, it is this type of compressor that must be used. Perhaps the greatest advantage of the axial flow compressor is its high thrust per unit frontal area.

In today's engines, the average axial flow type attains a static thrust per unit area of about 1500 lb/ft^2, which is about four times the amount of the average centrifugal flow type engine developing about 400 lb/ft^2. These two characteristics of the axial flow compressor - high pressure ratios at good efficiency and high thrust per unit frontal area - indicate the realm of its best application in high-thrust engines for high-speed aircraft. Briefly, the axial flow compressor provides large air-handling abilities with a small frontal area, a straight-through flow system, and high pressure ratios with relatively high efficiencies. Its chief disadvantage is its complexity and cost. Hundreds of blades are needed to achieve the pressure ratios required by turbojet engines. Figure 7.62 shows the stator and rotor blades of a typical axial flow type compressor.

The complexity can be visualized from Figure 7.62. In general, each row of rotor and stator blades is a different size and design. Compressor blades are usually made of steel, magnesium alloy, aluminum alloy, or titanium, and it is not uncommon for one compressor to have some steel blades and some alloy blades. The rotor blades must be secured properly to the rotor disk to withstand the stresses imposed by high rotational speeds.

7.8.3.4 PRINCIPLE OF OPERATION AND BASIC TERMS

The basic principle of operation of the axial flow compressor is the same as that of the centrifugal compressor, namely, imparting kinetic energy to the air by means of the rotating blades, and thence converting the kinetic energy to a pressure rise. The air

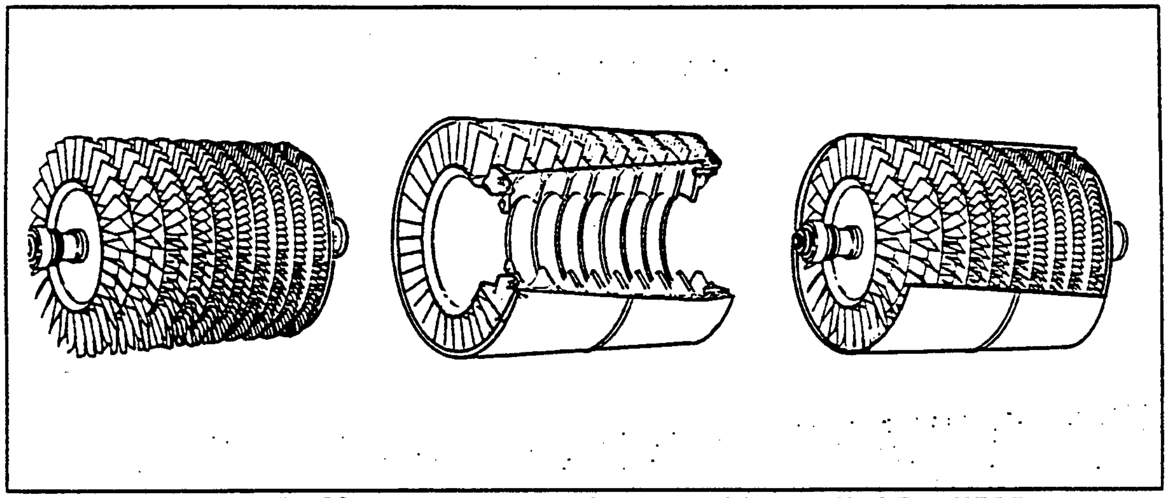


FIGURE 7.62. COMPONENTS AND ASSEMBLY OF AXIAL FLOW COMPRESSOR

enters axially from the left and into the inlet guide vanes where it is turned through a certain angle to impinge on the first row of rotating blades with the proper angle of attack. The rotating vanes add kinetic energy to the air and increase the pressure slightly, then discharge it with the proper angle to the first row of stator blades where the pressure is further increased by diffusion. The air is then directed to the second row of rotating blades, and the process is repeated through the reaming stages of the compressor. A compressor stage consists of a row of rotating blades followed by a row of stator blades. Most compressors have one to three rows of "straightener" or "diffuser" blades installed after the last stage to straighten and slow the air prior to its entry into the combustion chamber. If the purpose of these latter stator vanes is to provide additional air turbulence (as is sometimes necessary to alleviate combustion problems), they are called "mixer" blades.

The pressure ratio accomplished per stage of compression for subsonic stages is very modest when compared to one stage of a centrifugal compressor. For a typical axial flow compressor, the average pressure ratio per stage is about 1.20. Modern technology demonstrations have produced stage pressure ratios of 1.4. The over-all pressure ratio of an n stage compressor can be calculated by the relationship

$$SPR^n = TPR \tag{7.56}$$

where

SPR = stage pressure ratio

n = number of stages

TPR = total compressor pressure ratio

For a 10-stage compressor with an average SPR of 1.14, $1.14^{10} = 3.7$. The Allison/Rolls Royce TF-41 engine used in the A-7D and A-7E has SPR's up to 1.36 in the low pressure compressor (Reference 2). These relatively low stage pressure ratios are the reason for the large number of stages to achieve an overall pressure ratio up to 21 (as in the TF-41).

7.8.3.5 VELOCITY VECTOR ANALYSIS

The velocity vectors entering the compressor and through the first stage of the compressor are defined in Table 7.10. (See also Figure 7.64).

TABLE 7.10

Axial Flow Compressor Velocities

Symbol	Definition
$co = c_a$	absolute velocity of air entering the inlet guide vanes
c_1	absolute velocity of air leaving the inlet guide vanes
u_1	absolute linear velocity of a point on Rotor Stage 1 ($u_1 = 2\pi\, r_2\, N/60$)
w_1	velocity of air entering the rotor, <u>relative to Rotor Stage 1</u>
c_2	absolute velocity of air leaving Rotor Stage 1
u_2	absolute linear velocity of a point on Rotor Stage 1 ($u_2 = 2\pi\, r_2\, N/60$)
w_2	velocity of air leaving the rotor, <u>relative to Rotor Stage 1</u>
c_3	absolute velocity of air leaving Stator Stage 1
u_3	absolute linear velocity of a point on Rotor Stage 2
w_3	velocity of air entering Rotor Stage 2, <u>relative to Rotor Stage 2</u>

Figure 7.64 is a schematic diagram of the inlet guide-vanes and two stages of compressor blades that shows the air flow path through these blades, the velocity diagrams for each row of blades, and the static and total pressure variation of the air as it passes through the blades. This figure provides a vivid illustration of the principle of operation of an axial flow compressor.

The inlet guide vanes direct the air to give a proper angle of attack for the first row of rotating blades. During this process, the absolute air velocity, c, increases and the static pressure decreases. The first row of rotating blades imparts kinetic energy to the air proportional to $(w_1^2 - w_2^2 + c_2^2 - c_1^2)$, which brings about a total pressure increase proportional to the same term and a static pressure increase proportional to $(w_1^2 - w_2^2)$.

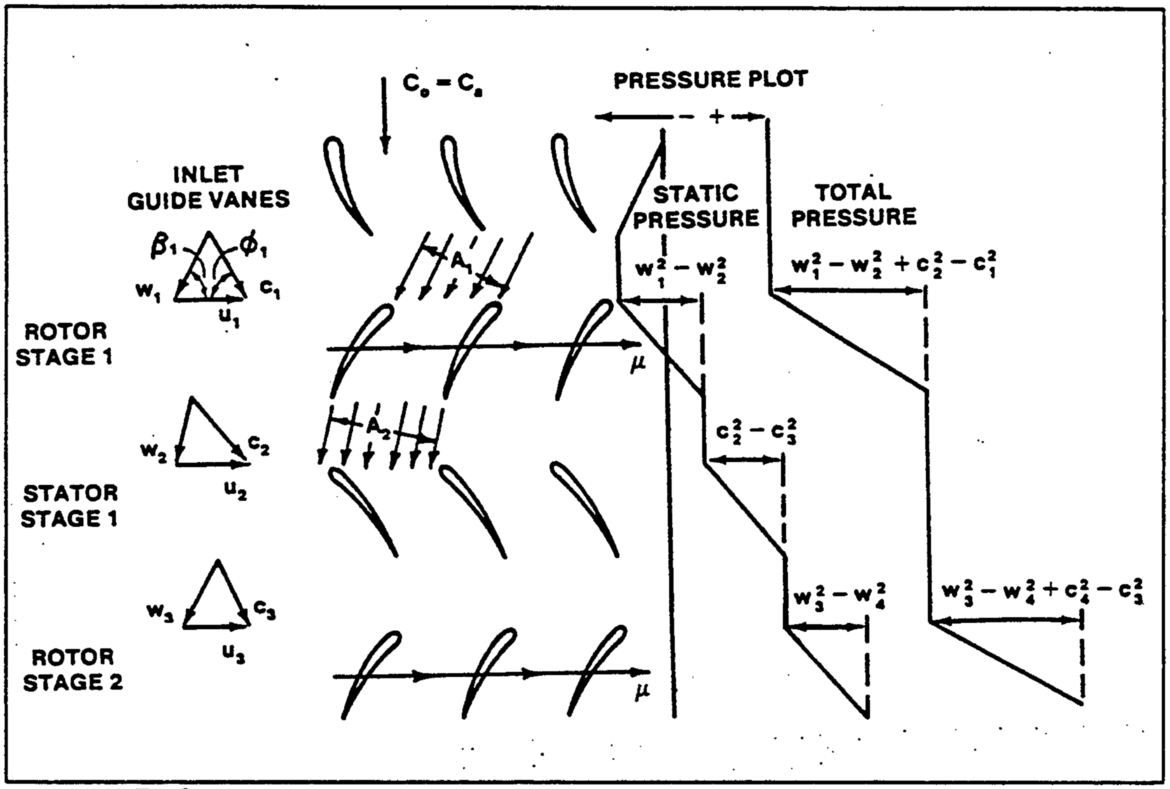


FIGURE 7.64. SCHEMATIC DIAGRAM OF COMPRESSOR BLADING EFFECTS

The air velocity relative to the rotating blades decreases ($w_2 < w_1$) because the flow area increases ($A_2 > A_1$). The stator blades of the first stage decrease the absolute air velocity ($c_3 < c_2$) to bring about a static pressure rise and turn the air to achieve the proper angle of attack for the second stage of rotating blades. Table 7.11 shows the pertinent variation that occurs in a typical axial flow compressor stage.

TABLE 7.11
VARIATION ACROSS A TYPICAL AXIAL FLOW COMPRESSOR STAGE

	Absolute Velocity c	Relative Velocity w	Flow Width	Static Pressure P	Total Pressure PT
Rotor	increases	decreases	increases	increases	increases
Stator	decreases	increases	increases	increases	about constant

Normally, the temperature change caused by diffusion alone, is not significant. The temperature rise which causes the air to get hotter and hotter as it continues through the compressor is the result of the work being done on the air by the compressor rotors.

Because the airflow process in an axial compressor diffusion, i.e., and adverse pressure gradient exists; it is very unstable. High efficiencies can be maintained only at very small rates of diffusion. When compared with a turbine, quite a number of compressor stages are necessary in order to keep the diffusion rate small through each individual stage. Also, the permissible turning angles of the blades are considerable smaller than those which can be used in turbines. These are the reasons why an axial compressor has such a small pressure ratio per stage and must have many more stages than does the turbine which drives it.

7.8.3.6 DUAL AXIAL COMPRESSORS

A single axial compressor might theoretically be built to consist of as many stages as would be necessary to produce any required compression ratio. If such were the case, at low off-design speeds the rear most stages of the compressor would operate inefficiently, and the foremost stages would be overloaded. Such a condition would produce "compressor stall". This condition can be corrected by bleeding interstage compressor air overboard or varying the airflow and pressure ratio of the stages by use of variable vanes during part-throttle operation. Excessive air bleeding, however, is wasteful. Greater flexibility for part-throttle conditions and for starting can be attained more efficiently by splitting the compressor into two mechanically independent rotor systems. Each is driven by its own separate turbine, at its own best speed (Figure 7.65). The high pressure compressor has shorter blades than the low pressure compressor, and is lighter in weight. Since the work of compression by the high pressure compressor heats the air within the compressor to higher temperatures than occur within the low pressure compressor, higher tip speeds are possible before the blade tips attain their limiting Mach, because the speed of sound increases as the air temperature increases. Hence, the high pressure compressor can run at a higher speed than the low pressure compressor.

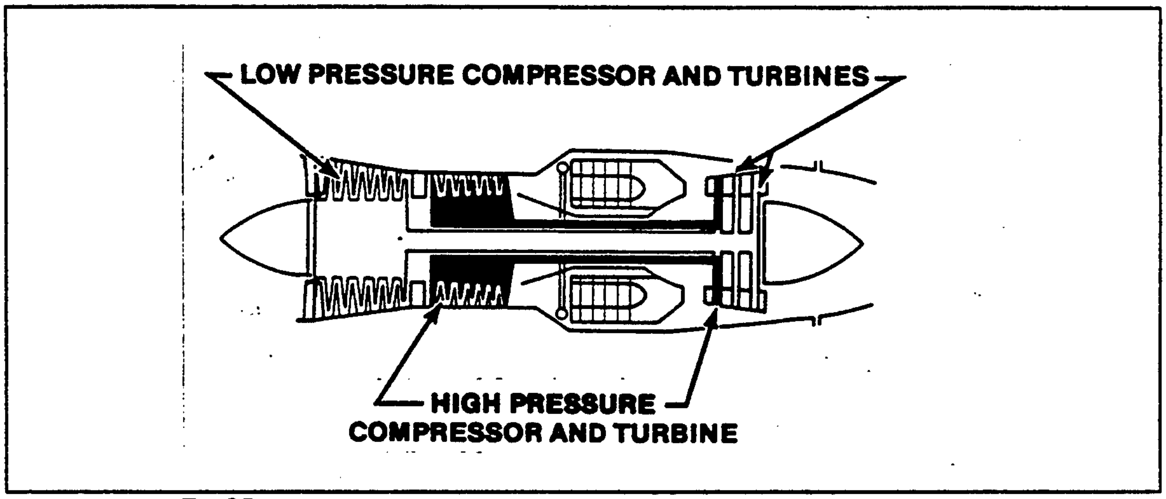


FIGURE 7.65. DUAL AXIAL COMPRESSOR OR TWIN-SPOOL SYSTEM

When dual, or, as sometimes called, split or twin-spool, compressors are used, high compression ratios can be attained with minimum total compressor weight and frontal area. Usually when dual compressors are used, the high pressure compressor rotor is the rotor to which the engine starter drive is connected. Only the lighter part of the compressor is cranked, which considerably reduces the torque required to start the engine. The size and weight of the starting system may therefore be appreciable less. The speeds of the tow rotors are matched for best efficiency and stall margin.

7.8.3.7 COMPRESSOR PERFORMANCE CHARTS

Compressor performance charts map the various regions of operation of the compressor. Their use gives a quick visual presentation of operational characteristics that could not be matched by scores of equations or tables.

From the dimensional analysis techniques discussed in Chapter 2, several dimensionless ratios used to analyze engine compressor performance can be derived. These ratios are listed in Table 7.12.

TABLE 7.12
COMPRESSOR DIMENSIONLESS PERFORMANCE RATIOS

Actual Value		Corrected Value
Thrust F_n		F_n/δ
Fuel Flow	$\dot{w}_f$	$\frac{\dot{w}_f}{\delta\sqrt{\theta}}$
Air Flow	$\dot{w}_a$	$\frac{\dot{w}_a\sqrt{\theta}}{\delta}$
RPM's n		$\frac{N}{\sqrt{\theta}}$

Figure 7.66 shows a typical compressor performance chart, which presents pressure ratio plotted against the corrected weight-flow rate for various corrected RPM's.

Note that all RPM operating lines reach the limiting surge, or stall, line. Operation to the left of this line is unstable. Thus, the slope of the RPM line in a stable region must always be negative. A positive slope indicates unstable operation; therefore, do not operate to the left of the surge line. One dotted line in Figure 7.66 shows the RPM line as it appears in unstable operation.

Normally, a cross plot of compressor adiabatic efficiency is also made on the performance chart as illustrated in Figure 7.66. These various efficiencies show that at one optimum design flow rate, RPM, and pressure ratio, maximum efficiency is achieved. This point is properly called the design point. It should be noted, therefore, that a compressor should be operated as much as possible near the design point in order to maintain reasonable values of efficiency. For example, if the RPM is decreased from the design condition, the efficiency drops off. Furthermore, if the pressure of the system is decreased or increased, the efficiency also drops off.

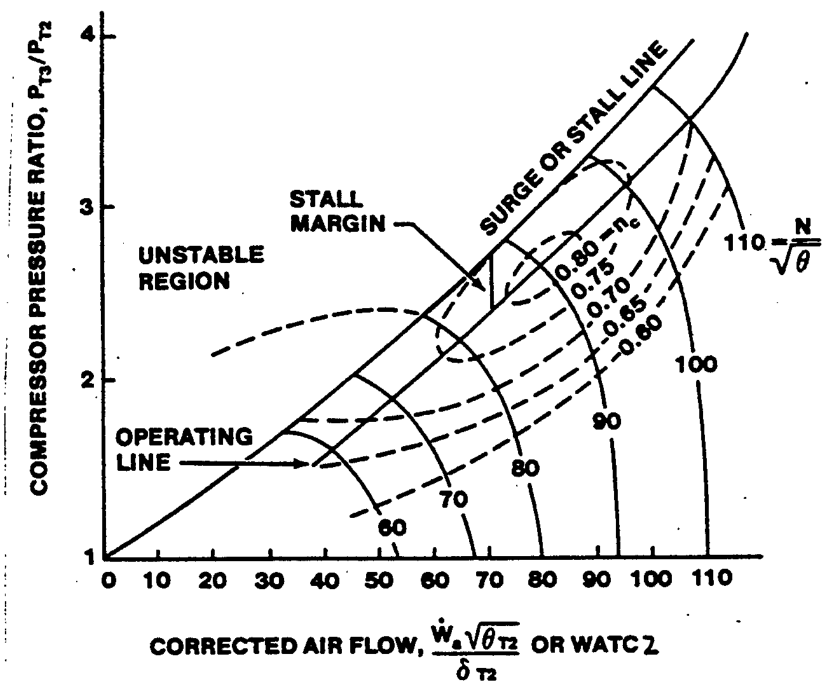


FIGURE 7.66. TYPICAL COMPRESSOR PERFORMANCE CHART

The compressor performance chart can be used to explain many turbojet engine operating characteristics. In Figure 7.66 the curve labeled "Operating Line" is that line on which the compressor operates when installed in a given fixed geometry engine. This line corresponds to the flow resistance of a given assembly of combustion chamber, turbine, tailpipe, and exhaust nozzle. Any change of these components would produce a new operating line. If the exhaust nozzle area were reduced, flow resistance would be added and the operating line would move upward nearer the surge line.

Compressor Stall Margin is defined as the difference between the compressor pressure ratio at the stall line and the CPR at the operating line divided by the CPR at the operating line for a constant value of WATC2.

Mathematically

$$SM=\frac{CPR_{STALL}-CPR_{op}}{CPR_{op}} \quad FOR\ WATC2\ CONSTANT$$

(7.57)

7.8.3.8 COMPRESSOR STALL

It is a characteristic common to gas turbine compressors of all types to stall under certain operating conditions. Some call this surge. Others endeavor to differentiate between stall and surge, but usually the two terms may be considered synonymous and may be treated as one and the same thing. Compressor stall occurs in many different forms and under many different conditions. Stall is neither easy to describe nor to understand, particularly because the stall characteristics of no two engine designs will be the same. In general, stall results when the compressor attempts to supply pressure ratios higher than its capability.

In its milder form, compressor stall can be recognized by the condition known as "chugging" which is occasionally encountered during ground engine operation at low thrust. In flight, under severe conditions of "slam" acceleration, or when slipping or skidding during evasive action, or when flying in very turbulent air, stall may become sufficiently pronounced to cause loud bangs and engine vibration. In most cases, this condition is of short duration and will either correct itself or can be corrected by retarding the throttle to IDLE and advancing it again, slowly.

If a physical occurrence took place which greatly increased the blade angle of attack, the blade would stall. For example, if the air flow rate were reduced, the blade angle of attack would increase, and stall might occur. This same phenomenon can occur during rapid rotor acceleration if the fuel scheduling to the combustor is improper. If the fuel flow rate during acceleration is too high, the high temperature and pressure resulting in the combustor will produce excessive back pressure, causing an increase in blade angle of attack, which if great enough, will produce stall. Compressor stall due to engine acceleration and afterburner initiation are the most common types of stall. Another type compressor stall is "rotating stall." This type of stall is characterized by the stall region progressing from one blade to the adjacent blade, and the resulting stall cell rotates in the direction of the rotor at .4-.5 rotor speed. The flow separation on the stalled blade causes the angle of attack to increase on the adjacent blade. This stalls it, causing stall on the next blade, and so on.

During rotating stall, although the pressure ratio and flow rate of the engine system may be in equilibrium, the compressor and turbine torques are not equal. The turbine torque has fallen off because the turbine pressure ratio and flow rate are at lower values, and it generates less torque. The compressor torque is reduced, but not as much as the turbine due to large energy dissipation in the compressor. The compressor torque will usually exceed the turbine torque significantly, and the rotor will tend to decelerate.

The occurrence of rotating stall also causes combustor gas temperature to rise rapidly because the airflow rate drops quickly and the fuel-to-air ratio in the combustor is increased. The danger of rotating stall operation in a gas turbine engine is that the combustor gas temperature will exceed allowable limits for the turbine and/or that the rotor speed will fall below the self-sustaining level. Generally, it will then be necessary to shut down the engine and allow it to cool somewhat before restarting. All but the very minor degrees of compressor stall are to be avoided by both design and operation.

There are a number of other factors that tend to induce compressor stalls -- high altitude operation, with its consequent reduction in compressor inlet Reynolds number, causes a slight reduction in the compressor stall pressure ratio. The pressure gradients which may exist over the "face" (entering section) of the compressor may reduce the stall margin by decreasing the stall line sufficiently to cause stall. These pressure distortions can result from poor inlet duct design, inadequate removal of inlet duct boundary layer, operation of aircraft at high angles of attack or side slip, expulsion of gun and rocket gases into the inlets, and so on. Most of these items can be controlled by good airframe design.

There are many degrees of compressor stall. It may range from one or several blades of single stage to complete flow breakdown, and the flow will momentarily reverse to cause a loss of combustor flame. Flameouts can occur even in less severe cases.

7.8.3.9 METHODS OF INCREASING STALL MARGIN

Current engines employ several techniques for either lowering the operating line or increasing the stall line for increased stall margin. Paragraph 7.8.3.6 discusses the use of compressor bleed and variable compressor vanes as techniques for improving stall margin at the lower rotor speeds. New engines also vary the fan operating line based on the level of distortion the inlet is generating and/or if a throttle transient is being requested. These techniques for maximizing either performance or stall margin, depending on current conditions, are becoming more viable with the use of digital flight and propulsion control systems.

7.8.4 COMBUSTION CHAMBERS

The combustion chamber of an air-breathing gas turbine jet propulsion engine is required to deliver large amounts of heat energy to the airstream and direct it with proper temperature level and temperature profile to the turbine. A good combustion chamber must provide complete burning of the fuel with a minimum of pressure loss, operate without accumulating deposits, ignite the fuel easily, and give reliable service over an extended period of time. The above requirements must be met over the complete range of jet engine operation - engine RPM, flight speed, and altitude.

In order to meet these requirements, the combustion chambers of turbojet and turboprop engines have evolved into two basic types: (1) the can or tubular type and (2) and the annular type.

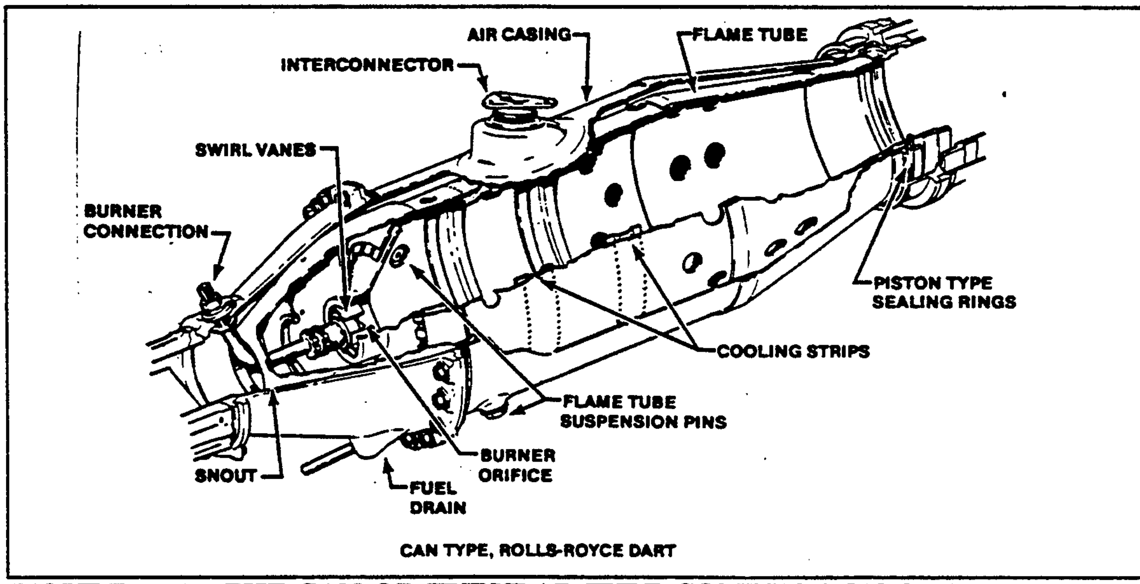


FIGURE 7.67. THE CAN OR TUBULAR-TYPE COMBUSTION CHAMBER

Both types contain the same basic elements: an outer casing or shell, a perforated inner liner or flame tube, a primary combustion zone, a liquid fuel-injection system, and provisions for initial ignition. Figures 7.67 and 7.68 respectively, present a typical can or tubular-type combustion chamber and typical annular combustion chamber.

The can-type combustor illustrated in Figure 7.68 is one of several (7-14) such units that make up the overall combustion chamber for a given engine. These individual "cans" are interconnected by means of tubes located between the cans in order to provide uniform combustion characteristics in each tube and to allow flame travel between "cans" for ignition since, in the normal installation, only two cans will be equipped with spark plugs or igniters. The cans are located around the main rotor shaft and are connected to the compressor and turbine sections. The annular type combustion chamber illustrated in Figure 7.68 similarly has only two igniters. The relative merits of the two types of burners are about equal.

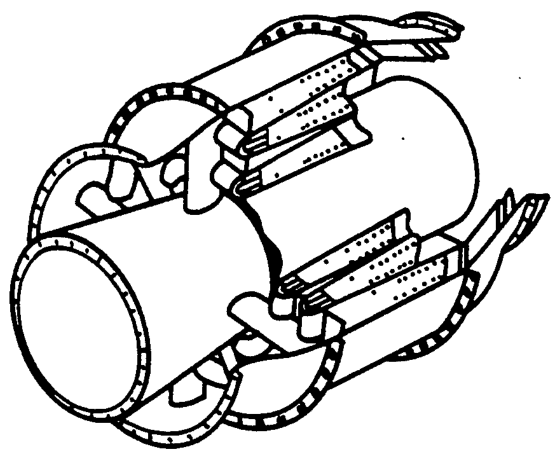


FIGURE 7.68. TYPICAL ANNULAR COMBUSTION CHAMBER

7.8.4.1 COMBUSTOR OPERATION

Combustion chamber design is dictated mainly by the general characteristics found in any type of combustion process. The requirements deriving from these characteristics are basic, whether considering combustion in a fireplace or in a turbojet engine. They are: (1) proper mixture ratio, (2) temperature of reactants, (3) turbulence for good mixing, and (4) time for burning. In addition, for aircraft turbine engines, the combustion process should be accomplished with a minimum possible pressure loss.

The first requirement in any combustion process is mixture ratio because fuel-air rations have lean and rich limits of inflammability beyond which burning is impossible. For heat engines, these limits in terms of fuel-air ratio are about 0.04 for the lean limit, and about 0.15 for the rich limit.

For most hydrocarbon liquid fuels, the stoichiometric fuel-air ratio is about 0.066. Thus, it is apparent that any combustion chamber must maintain a mixture ratio which is within allowable limits if burning is to occur at all, and within much more stringent limits if good burning is to occur. In gas turbine engines, operation with overall mixture rations which even approach the stoichiometric value are not feasible, because such mixture ratios produce exhaust gas temperatures of about 4000°R, well in excess of the maximum allowable turbine blade inlet temperature. In order to reduce the temperature of the gases leaving the combustion chamber to an allowable value, it is necessary to operate the combustion chambers with a large quantity of excess air to provide adequate cooling. The large amount of excess air required reduces the overall fuel-air ratio to a value which is generally below 0.02. This fuel-air ratio is, of course, too lean for burning; hence, the burner design must provide a method of bypassing about 60 to 75% of the air around the actual combustion zone. The bypassed air is known as secondary air because it does not enter into the combustion process, whereas the remainder of the air, that which actually takes part in the combustion process, is known as primary air. The amount of primary air is dictated by the fuel-air ratio in the actual combustion zone, or primary zone, of about 0.08. Figure 7.69 presents a schematic diagram of the cross section of a typical burner.

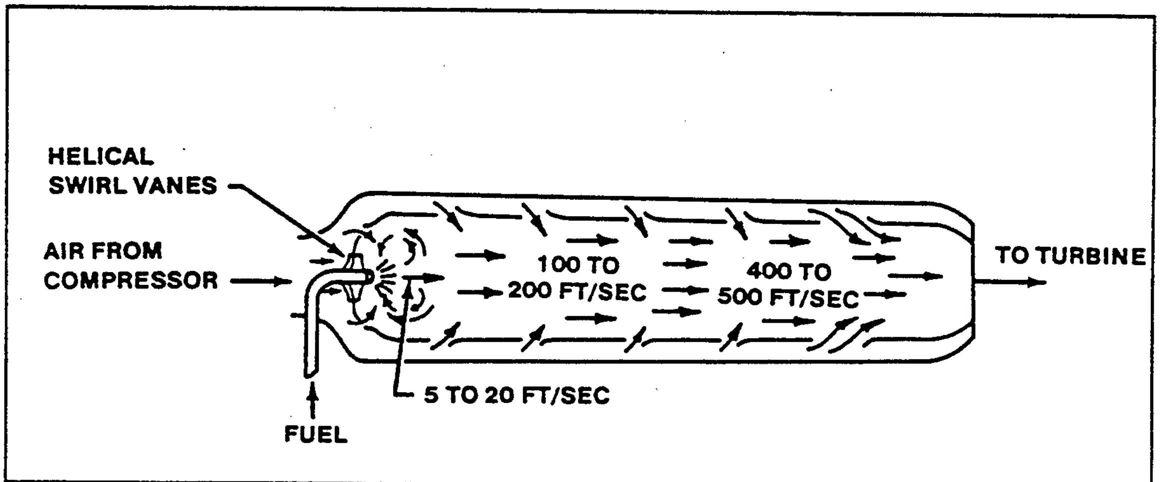


FIGURE 7.69. SCHEMATIC DIAGRAM OF BURNER CROSS SECTION

This diagram shows the flow path of the primary and secondary air. The secondary air is progressively mixed with the combustion gases as they flow within the inner liner in order to cool the overall mixture to its proper temperature where it enters the turbine section, and to provide a film of cool air to protect the inner liner. It is evident that the combustion chamber design is largely dictated by the mixture ratio requirements, and the proportioning of the primary and secondary airflows must be properly maintained during the whole process. This mixing of the secondary air must also be accomplished to provide the proper temperature profile to the turbine.

After proper mixture ratio is attained, combustion will not be maintained unless the three T's of combustion - temperature, turbulence, and time - are also present. The temperature of the reactants must be above the ignition temperature. For initial combustion, this temperature is usually provided by means of electrical energy in the form of a spark. Normally, two spark igniters are located in the combustion chamber to initiate combustion by providing localized regions where the reactant temperature will be considerably above the ignition temperature. After ignition occurs and burning begins, reactants are kept at a high temperature by the heat released from the burning fuel, and the spark energy is no longer required.

Sufficient turbulence must be created and maintained for combustion to be complete, because each fuel molecule requires an exact number of oxygen molecules before complete combustion can occur. Adequate turbulence insures that each fuel molecule will intimately mix with the air and find its proper number of oxygen molecules. Since turbulence is accompanied by a fluid pressure drop, it is essential that only enough turbulence be created to achieve proper mixing, otherwise excess pressure drop will occur with a consequent reduction in overall engine thrust.

Sufficient time must be allocated for the fuel to burn if the combustion process is to be complete. If the air flow velocities are greater than the flame speeds, approximately 60-100 ft/sec, the flame will be blown down the combustion chamber and out of the engine, causing flameout.

7.8.4.2 COMBUSTION PROCESS AND EFFICIENCY

The ideal combustion process occurs at constant pressure with complete release of the fuel heating value. In an actual combustion process, the total pressure drops slightly due to friction and the momentum pressure drop due to heat addition (static pressure drops because of friction as well as gas acceleration), and the combustion is not complete because some of the fuel molecules are not burned.

As shown in Figure 7.70, the ideal process occurs at the constant total pressure P_{T3} from the burner inlet Section 3 to the burner outlet Section 3 to the burner outlet Section 4'. The actual process occurs from Sections 3 to 4, which, compared to the ideal process, represents a loss in total pressure (about 5% in modern combustors) and a loss in end temperature because of incomplete combustion. Figure 7.70 shows the combustor process for the maximum full throttle position where T_{T4} is equal to the maximum allowable 4 turbine inlet temperature. Operation at less than full throttle will produce T_{T4} values, which are less than that shown on the figure.

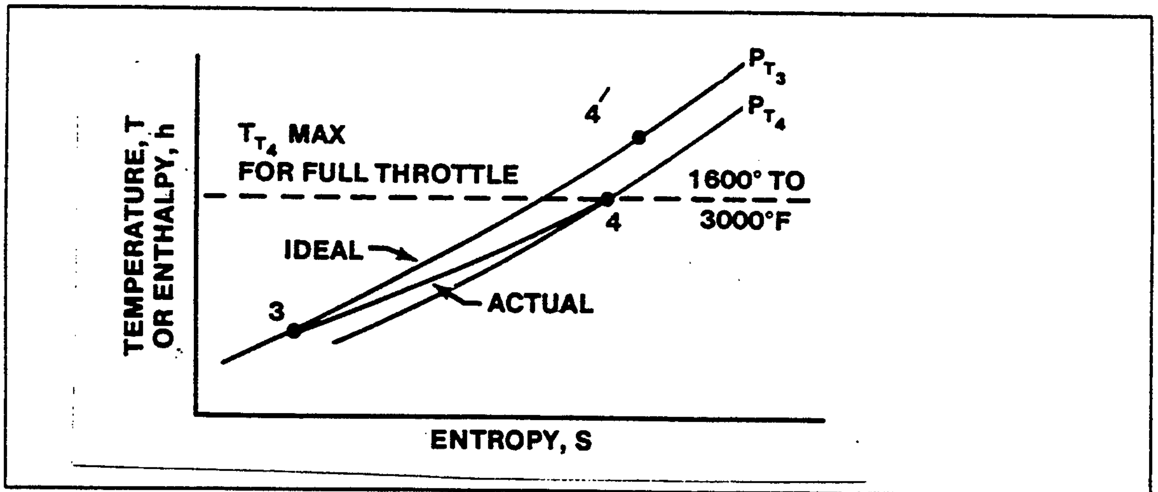


FIGURE 7.70. IDEAL AND ACTUAL COMBUSTION PROCESS ON h-S PLANE

The combustion chamber efficiency is defined as the ratio of the actual heat released to that which ideally could be released from a given quantity of fuel, or

$$\eta_b = \frac{\dot{W}_a}{\dot{W}_f} \quad \frac{c_p(T_{T4} - T_{T3})}{H.V.}$$

For constant specific heat this becomes

$$\eta_b = \frac{T_{T4} - T_{T3}}{T_{T_4'} - T_{T_3}}$$

In most turbojet and turbofan engine operations, combustor efficiencies vary between 98% and 100%.

7.8.4.3 FUEL CONTROL UNITS

The amount of fuel supplied to the combustion chamber must be closely controlled and adjusted for different engine operating conditions; altitude, temperature, engine RPM, and forward flight speed. This job is performed by the fuel control unit. In basic fuel control units, the unit senses throttle position, engine RPM, engine air inlet pressure, and engine air inlet temperature. Many aircraft turbine engines have a much more complicated fuel control system than the basic unit. These are the variable geometry engines, those equipped with variable area exhaust nozzles, variable angle inlet guide vanes, or variable angle compressor stator vanes. On engines of this type, the fuel control unit, in addition to sensing these variables, must also sense turbine outlet temperature and control the exhaust nozzle area or inlet guide vane area.

7.8.4.3.1 Digital Electronic Engine Control. Engine Control (DEEC) is a much more advanced system than the current generation of ordinary fuel control systems. Although it is a completely electronic system, a hydro-mechanical Back-Up Control (BUC) may be selected either by the pilot or automatically by the DEEC if the need arises.

The primary engine variables that are sensed and controlled by the DEEC are illustrated in Figure 7.71.

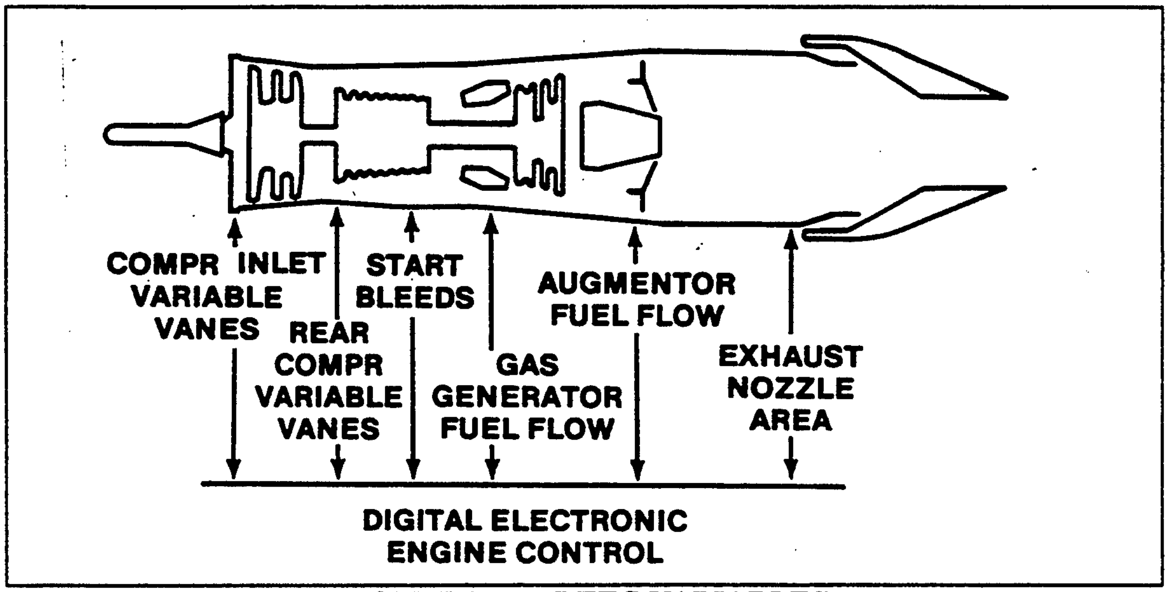


FIGURE 7.71. DEEC VARIABLES

Sequencing and control of these variables allows the DEEC to provide:

1) Reliable engine starts (including airstarts)
2) Safe throttle transients without stall, over temperature or blowout
3) Consistent idle thrust
4) Stable intermediate thrust without exceeding speed or temperature limits
5) Smooth augmentor transients without blowout or stall
6) Backup control capability
7) No required engine trim

Pilot workload is considerably reduced with incorporation of DEEC.

7.8.5 GAS TURBINES

The primary purpose of the gas turbine in a turbojet or turbofan engine is to extract mechanical energy from the hot gases delivered to it by the combustion chamber and to supply shaft power to drive the compressor. The turbine must also supply power to the auxiliary equipment, such as fuel pumps, oil pumps, and electrical generators. In turboprop and turboshaft engines, the turbine must also supply power to drive the propeller or helicopter rotor. Typically, three-fourths of all the energy available for the products of combustion is necessary to drive the compressor. If the engine is a turboprop or turboshaft, the turbine is designed to extract all the energy possible from the gases passing through the engine. So efficient is the turbine, in this case, that in a turboprop aircraft the propeller provides approximately 90% of the propulsive force, leaving but 10% to be supplied by jet thrust.

The axial flow turbine is comprised of two main elements: a turbine wheel, or rotor, and a set of stationary vanes (Figure 7.72).

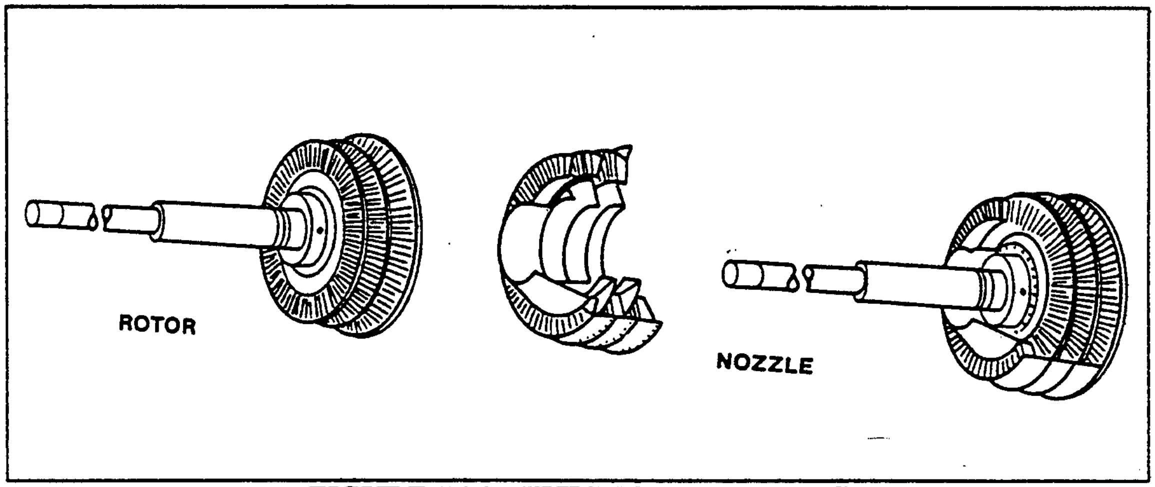


FIGURE 7.72. TURBINE ELEMENTS

The stationary section consists of a plane of contoured vanes, concentric with the axis of the turbine, and set at an angle to form a series of small nozzles which discharge the turbine gases onto the blades of the turbine wheel. For this reason, the stationary vane assembly is usually referred to as the turbine nozzle, and the vanes, themselves, are called nozzle guide vanes. Aerodynamic design of turbine blades is less critical than compressor blades because they operate in a regime of favorable pressure gradient rather than an adverse pressure gradient. Turbine-nozzle area is a critical part of turbine design because it establishes the engine (compressor operating line). The jets of escaping gases which are formed by the nozzle discharge are directed against the rotating turbine blades in a direction which enables the kinetic energy of the gases to be transformed into mechanical energy which is generated by the rotating turbine wheel.

Turbines may be either single or multiple-stage. When the turbine has more than one stage, stationary vanes are inserted between each rotor wheel and the rotor wheel downstream, as well as at the entrance and exit of the turbine unit. Each set of stationary vanes forms a nozzle vane assembly for the turbine wheel that follows. The exit set of vanes serves to straighten the gas flow before passage through the jet nozzle.

7.8.5.1 TURBINE DESIGN CONSIDERATIONS

Turbines are subjected to both high rotor speeds and high temperatures. High rotor speeds result in high centrifugal forces, and because of high temperatures, turbines must operate close to temperature limits which, if exceeded, will lower the strength of the construction materials used in them. Turbine blades with continued use undergo distortion of the blade, which is know as "creep." Creep means that the blade stretches or elongates. This condition is cumulative, the rate of creep being determined by the load imposed on the turbine and the strength of the blade, which is determined by the temperature within the turbine.

From Figure 7.73 it may be seen that the highest stress plus fatigue considerations near the blade root require a lower temperature to maximize the blade material strength.

Creep is the dominant factor in the middle of the blade. Higher temperatures are allowable further out because of the lower stresses. Near the blade tip, erosion or stator blade stresses again reduce the temperature level. With these limits established, the combustion chamber design objective is to match the desired gas temperature profile as closely as possible.

The turbine wheel is a dynamically balanced unit consisting of steel alloy blades, or buckets, as they are sometimes called, attached to a rotating disc. The base of the blade is usually of a so-called "fir tree" design to enable it to be firmly attached to the disc and still allow room for expansion. In some turbines, the rotating blades are open at their outer perimeter. In others, the blade is shrouded at the tip, as shown in Figure 7.74. The shrouded blades form a band around the perimeter of the turbine

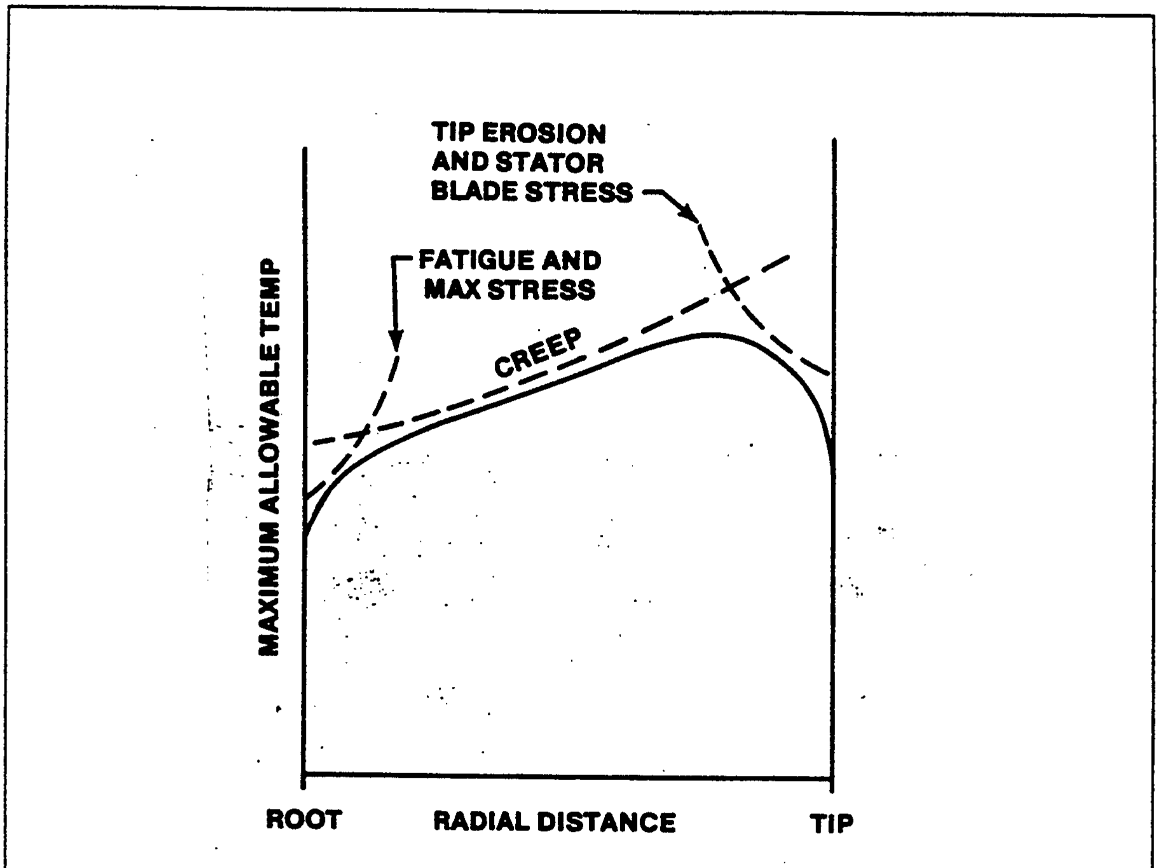


FIGURE 7.73. TURBINE INLET BLADE TEMPERATURE PROFILE LIMITATIONS

wheel, which serves to reduce blade vibrations. The weight of the shrouded tips is offset because the shrouds permit thinner, more efficient blade sections than are otherwise possible because of vibration limitations. Also, by acting in the same manner as aircraft wingtip fences, the shrouds improve the air flow characteristics and increase the efficiency of the turbine. The shrouds also serve to cut down gas leakage around the tips of the turbine blades.

7.8.5.2 GENERAL THERMODYNAMIC ANALYSIS

Figure 7.75 depicts the energy balance of a gas turbine. It receives high-temperature, high-pressure gas at Section 4, extracts energy from it in the form of shaft work, and discharges the gas at a lower level of pressure and temperature.

The energy equation for this machine, when written for no change in potential energy and an adiabatic process, is

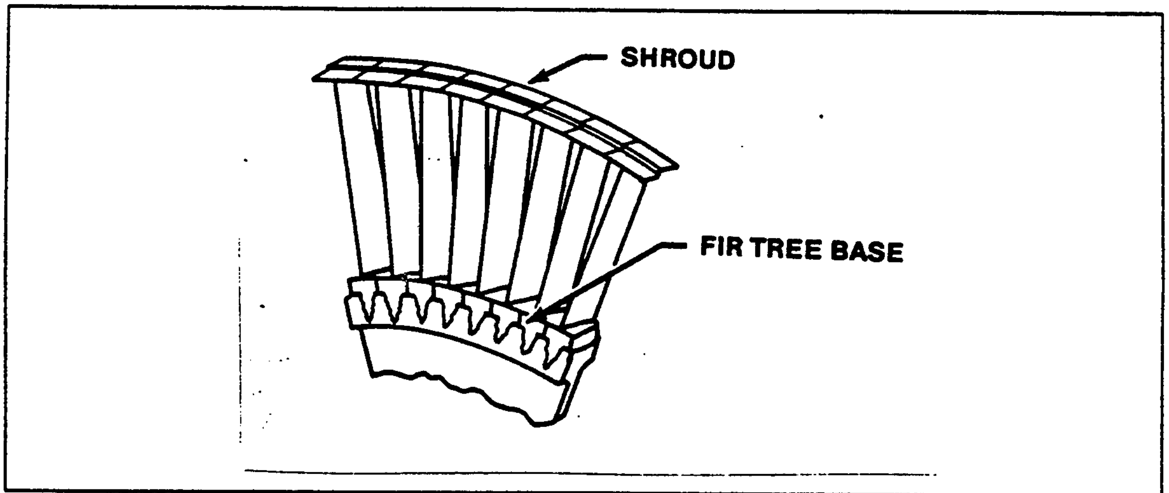


FIGURE 7.74. SHROUDED TURBINE-ROTOR BLADES

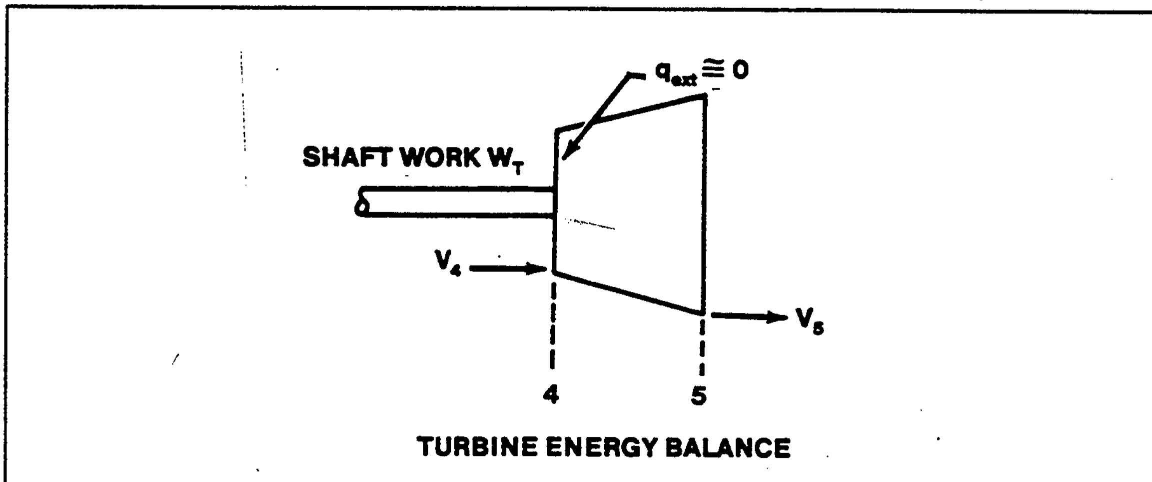


FIGURE 7.75. TURBINE ENERGY BALANCE

$$\frac{V_4^2}{2g} + h_4 = W_T + \frac{V_5^2}{2g} + h_5$$

Solving Equation 7.60 for the turbine work and expressing the gas energy content at Sections 4 and 5 in terms of total conditions gives

$$W_T = h_{T_4} - h_{T_5} = C_P\left[T_{T_4} - T_{T_5}\right]$$

Total conditions are used in preference to the sum of static enthalpy and kinetic energy, because it is far easier to measure and evaluate two total enthalpies than it is to measure two static enthalpies and two velocities. Equation 7.61 merely states that the turbine work is equal to the change in energy content of the gases passing through the turbine.

Figure 7.76 shows the actual and ideal turbine process on an h-S diagram.

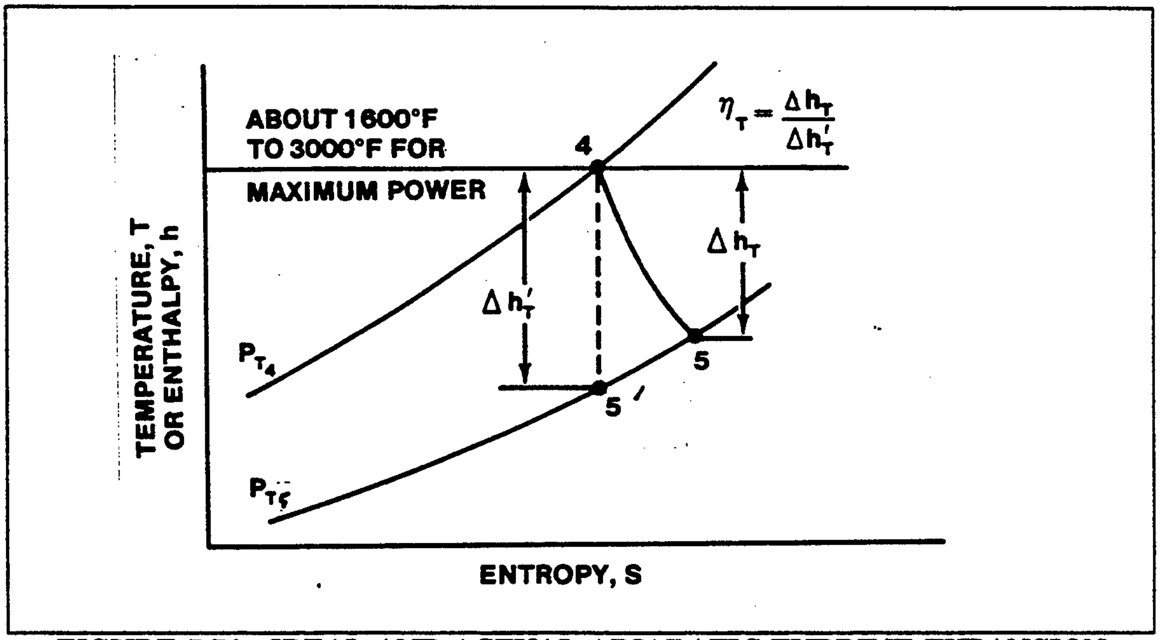


FIGURE 7.76. IDEAL AND ACTUAL ADIABATIC TURBINE EXPANSION PROCESS

Note that the ideal and actual processes for the turbine are defined in the same manner as they were for the compressor, namely, between the same two total pressure lines. For the assumption of an adiabatic flow process between pressures P_{T4} and P_{T5}, we can have either the ideal isentropic process 4-5' or a general adiabatic process with friction 4-5. It is evident that for an adiabatic process between two pressure lines, maximum turbine work will occur along a constant entropy path; therefore, great effort is expended to minimize the friction in turbines. The turbine adiabatic efficiency is defined as

$$\eta_T = \frac{h_{T_4} - h_{T_5}}{h_{t_4} - h_{T_5}} = \frac{\Delta h_T}{\Delta h_T'} \qquad (7.62)$$

The work produced per pound of fluid can be expressed in terms of the ideal path 4-5' as

$$W_T = \eta_T \, \Delta \acute{h}_T = \eta_T \, C_P \left[T_{T_4} - \acute{T}_{T_5} \right] \tag{7.63}$$

Since, by definition, the total pressure at points 5 and 5' are equal, the turbine work can be expressed in terms of pressure ratio and inlet temperature as

$$W_T = \eta_T \, \Delta \acute{h}_T = \eta_T \, C_P \, T_{T_4} \left[1 - \left[\frac{P_{T_5}}{P_{T_4}} \right]^{\frac{\gamma-1}{\gamma}} \right] \tag{7.64}$$

Considering Equation 7.64, the important factors which affect turbine work, namely, turbine efficiency h_T, turbine inlet temperature T_{T4}, and turbine pressure ratio P_{T4}/P_{T5} may be seen. An increase in any of these three factors will allow the turbine to develop more work per pound of fluid. Only small gains can be expected to accrue from improvements of turbine efficiency, since present efficiencies are up near the peak of development, 85 to 93%. Because of gas friction over the many turbine blades and the leakage losses over the blade tips, turbines inherently have about 10% overall loss. However, the prospect of operating turbines at higher inlet temperatures is indeed an attractive one to achieve more work per pound of fluid because, as shown in Equation 7.64, the turbine work is directly proportional to the absolute temperature of the entering gases.

$$\text{Turbine Shaft Horsepower} = \dot{w} \Delta h_T \times \frac{J}{550} \tag{7.65}$$

7.8.5.3 VELOCITY VECTOR ANALYSIS

Recalling the definitions of the velocity vectors c, w and u from the compressor section, Figure 7.77 illustrates the velocity, static pressure, and total pressure changes through a two-stage turbine.

7.8.5.4 IMPROVEMENT OF TURBINE INLET TEMPERATURE

The most attractive method of increasing thrust and turbine work per pound is to increase the turbine inlet temperature. Increases in turbine inlet temperature are

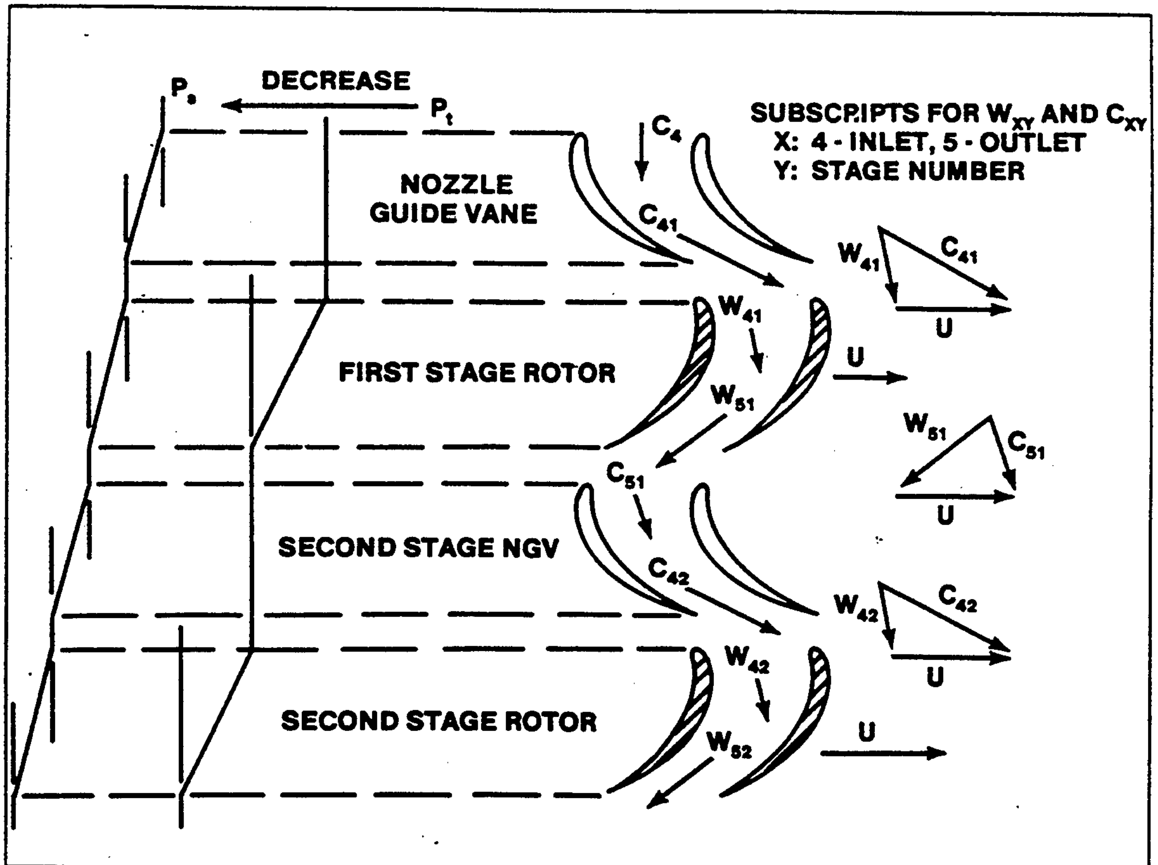


FIGURE 7.77. TURBINE ENERGY TRANSFER

directly tied to the need for better materials in construction of turbine blades and efficient methods of cooling them.

7.8.5.4.1 Materials Considerations. Much work has been done in recent years toward improving the high-temperature strength characteristics of metals and alloys. From this effort has come a series of cobalt and nickel-based alloys that offer significant high temperature strength improvements over iron-based alloys. Newer, more exotic materials hold still greater promise. Improved metallurgical techniques have allowed blades made of materials which have been directionally solidified, subjected to rapid solidification rates, or even constructed of a single crystal to be manufactured. Gains due to these metallurgical techniques are illustrated in Figure 7.78.

Some manufacturers are presently investigating the use of ceramic materials for use in turbine blades. While these blades hold a significant advantage over metal blades

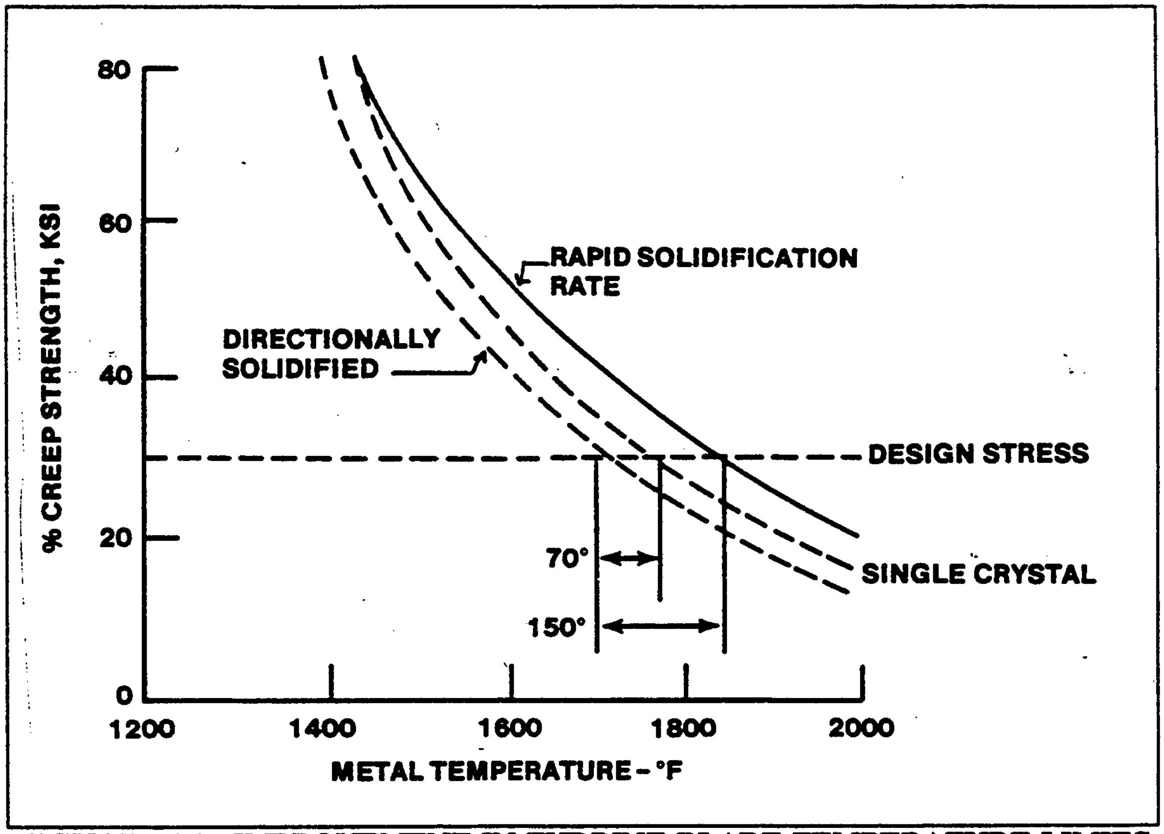


FIGURE 7.78. IMPROVEMENT IN TURBINE BLADE TEMPERATURE LIMITS DUE TO IMPROVED METALLURGICAL TECHNIQUES

with regard to their ability to withstand high temperatures, problems associated with stresses due to centrifugal loads associated with high rotation rates open new avenues of difficulties to overcome.

7.8.5.4.2 Turbine Blade Cooling. Turbine blades can be cooled by several different methods, but basically, each method uses a cooling fluid that passes through the blade so as to keep the blade metal within safe operating limits. The fact that air-cooled blades can produce appreciable power gains makes utilization of compressor bleed air appear to the best overall system for blade cooling. The criteria for achieving good cooling effectiveness come directly from the principles of heat transfer of a fluid in a closed duct. To attain high heat transfer rates in such a system, it is necessary to meet two basic requirements, namely, (1) flow the cooling fluid with a high Reynolds number, and (2) provide a large surface area for the heat flow path. With these points in mind, it is obvious why a finned blade is many times better than an open hollow blade. The open hollow blade does very little cooling, because a boundary layer which acts as an excellent insulator to heat transfer forms over the inner surface of the blade. The insertion of fins or tubes in the blade causes the cooling air to pass over greater surface area with high turbulence or a rubbing action, which produces a turbulent boundary layer that readily passes heat.

Another disadvantage of the open hollow blade is its structural limitation. Without fins or supporting members, the open hollow blade vibrates readily, and with large magnitude at its resonant frequency to produce a "breathing action" with consequent fatigue failure.

There are three general methods employed for blade cooling. These are the convection, impingement, and film cooling methods. A fourth method called transpiration cooling may be found in the literature, but the differences between film cooling and transpiration cooling are difficult to distinguish. The three methods discussed here are illustrated in Figure 7.79.

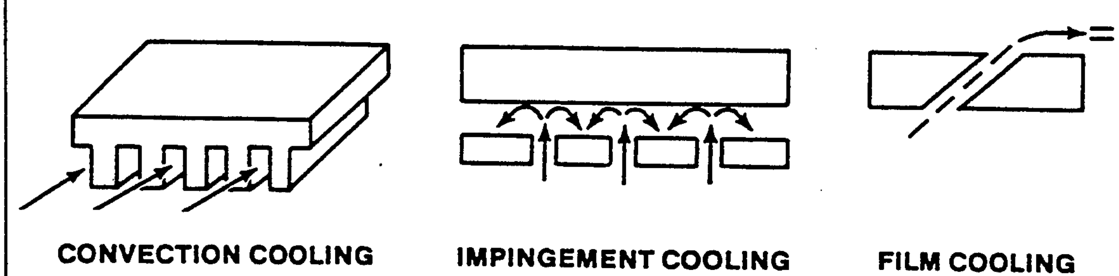


FIGURE 7.79. GENERAL TURBINE BLADE COOLING METHODS

Convection cooling is the simplest and was the first turbine blade cooling method used. With the convection cooling method, the coolant air flows outward from the base of the turbine blade to the end through internal passages within the blade. The effectiveness of convection cooling is limited by the size of the internal passages within the blade and the restriction on the quantity of cooling air available.

Impingement cooling is a form of convection cooling, but instead of the air flowing radially through one or more sections of the blade, the air is turned normal to the radial direction and passed through a series of holes so that it impinges on the inside of the blade at the area where cooling is desired. Impingement cooling is a very effective method in local areas and is easily adapted to stator blades. This method is usually employed at the leading edge of the blade where the highest temperatures are expected because of impingement of hot gases, but may be employed in any desired area.

Film cooling involves the injection of a secondary fluid, usually air, into the boundary layer of the primary fluid (hot gas). Injection of too much air into the boundary layer can defeat the purpose of increasing turbine inlet temperature. Film cooling is more effective than either convection cooling or impingement cooling. The air used for film cooling must be under high pressure because it is quickly dissipated by downstream mixing of the film air with mainstream hot gases.

The effectiveness of the three cooling methods is compared in Figure 7.80.

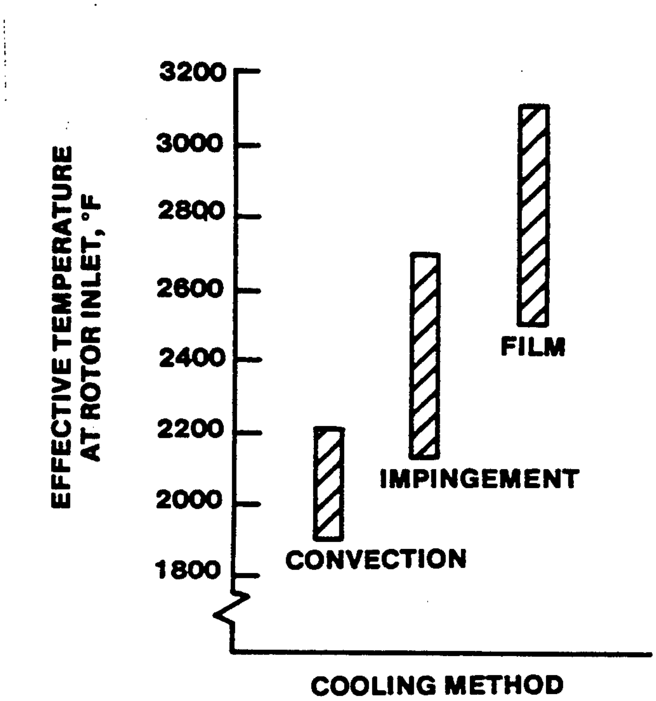


FIGURE 7.80. RELATIVE EFFECTIVENESS OF TURBINE BLADE COOLING METHODS

Like any other system which presents advantages to engine performance, there are also disadvantages incurred by cooling turbine blades. A cursory examination of the turbine blade-cooling problem leads one to think that the solution is relatively simple, that is, merely pass some compressor bleed air through hollow turbine blades, and the job is done. A more detailed study of the subject, however, will show that the overall turbine blade cooling problem is very complex. The basic problems of heat transfer in a duct are made more difficult and more complicated because the cooling air within the blades is accelerated by centrifugal forces while it absorbs large quantities of heat and the tendency for internal gas choking is present. At a given turbine inlet temperature, an engine with cooled blades suffers a definite performance loss relative to one with uncooled blades because the coolant air bled from the compressor does not take part in the combustion process, nor can it develop power in the turbine. It also requires pumping work to force it through the cooling system. Perhaps the greatest disadvantage of turbine blade cooling is cost due to complexity in fabrication. It has already been pointed out that the simple, open, hollow blades do not cool well enough; the ones with fins, inserts, and bundles of tubes, are difficult to manufacture but do provide adequate cooling. These complex cooled blades must be manufactured properly. In addition to providing adequate cooling, they must withstand the high stresses imposed on them by centrifugal loads. The turbine rotor required for cooled blades is also difficult to manufacture. In addition to the fabrication problems, a rotor supplying cooling air is further complicated by the air-sealing problem at the section where the coolant is brought into the rotor hub.

Some advantages and disadvantages of turbine blade cooling have been discussed, and in spite of the many complexities added to the engine by a blade cooling system, the performance attractiveness is still great, especially for turboprop engines, turbojet engines for high Mach flight, and high bypass ratio turbofan engines.

Historical and expected gains in turbine inlet temperature from combined materials improvement and cooling effectiveness are shown in Figure 7.81.

7.8.5.5 ENGINE INTERNAL TEMPERATURE CONTROL

In the event of a malfunction or under extreme flight conditions, regulation of engine internal temperatures can be marginal or even above the desired limits. Over temperatures can't be treated lightly. Just because the turbine does not melt away, there is no reason to assume that the engine cannot be or has not been damaged. Several momentarily high over temperatures will have as profound an effect on the engine as a single prolonged one of a lesser degree. Excessive internal temperatures aggravate such conditions as creep, deformation of sheet metal parts, and drooping. Operating the engine within the specified limits of temperature, RPM, and turbine

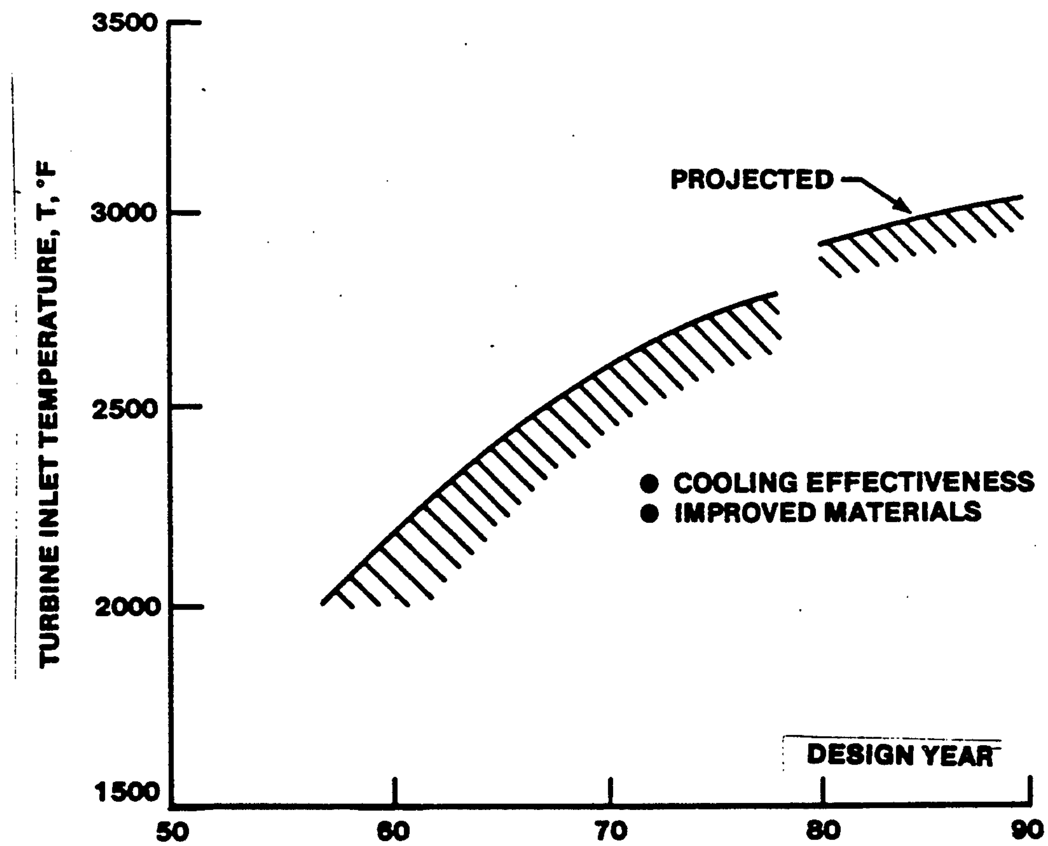


FIGURE 7.81. CHRONOLOGICAL IMPROVEMENT IN TURBINE INLET TEMPERATURE AS A FUNCTION OF TURBINE BLADE COOLING

discharge pressure or engine pressure ratio should become an instinctive technique to the turbojet, turbofan, and turboprop pilot. Modern digital electronic controls will automatically eliminate over temperatures without pilot attention.

7.8.6 EXHAUST DUCT/NOZZLE

The term, "exhaust duct," applies to the engine exhaust pipe or tailpipe connecting the turbine outlet and the jet nozzle of a nonafterburning engine. Although an afterburner might also be considered a type of exhaust duct, afterburning is a subject in itself and is dealt with subsequently.

If the engine exhaust gases could be discharged directly to the outside air in an exact axial direction at the turbine exit, an exhaust duct would not be necessary. This, however, is not practical. The largest total thrust can be obtained from the engine if the gases are discharged from the aircraft at the velocity obtained when nozzle exhaust static pressure is equal to ambient pressure. This was discussed in Chapter Six. An exhaust duct is therefore added, both to collect and straighten the gas flow as it comes from the turbine, and to increase the velocity of the gases before they are discharged from the exhaust nozzle at the rear of the duct. Increasing the velocity of the gases increases their momentum and increases the thrust produced.

7.8.6.1 CONVERGENT EXHAUST NOZZLE

The velocity of the gases within a convergent exhaust duct (Figure 7.82) are held to Mach 1 or less. These nozzles are used on subsonic aircraft where small performance penalties are incurred due to non-optimum expansion, but the weight and cost of a divergent nozzle is not cost effective.

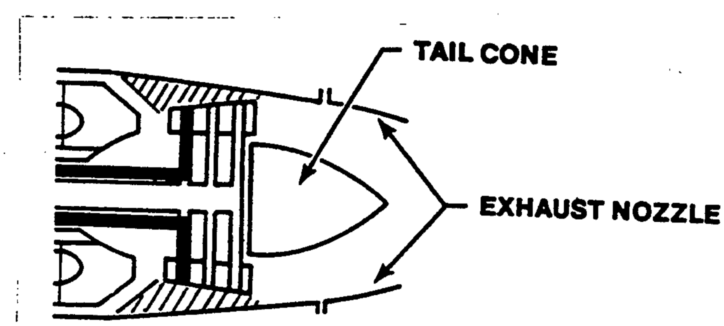


FIGURE 7.82. CONVENTIONAL CONVERGENT EXHAUST DUCT

7.8.6.2 CONVERGENT - DIVERGENT EXHAUST NOZZLE

Whenever the pressure ratio across an exhaust nozzle is high enough to produce gas velocities greater than Mach 1, more thrust can be gained by using a convergent-divergent type of nozzle (Figure 7.83). The advantage of a convergent-divergent nozzle is greatest at high Mach because it allows maximum thrust to be obtained.

In the discussion on thrust, it was pointed out that all of the pressure generated within an engine cannot be converted to velocity.

7.8.6.3 VARIABLE AREA NOZZLES

In order to obtain optimum performance, the pressure at the nozzle exhaust plane must match the ambient pressure. Recall that for a fixed area ratio, only one

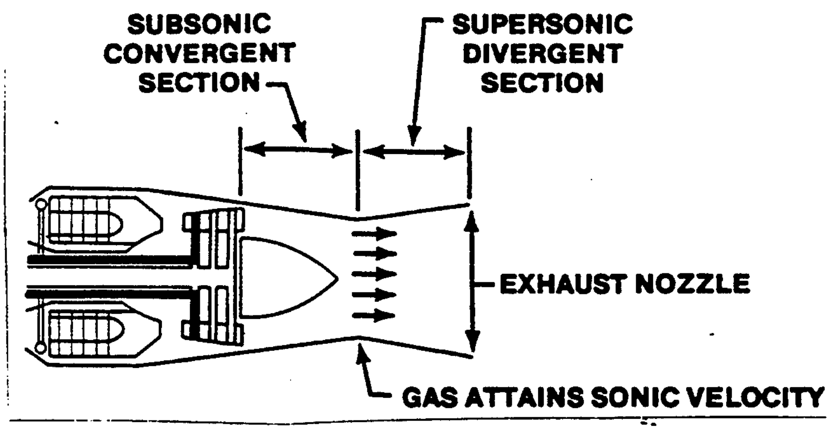


FIGURE 7.83. CONVERGENT-DIVERGENT EXHAUST DUCT (NOZZLE)

pressure ratio will give optimum performance. Therefore, in order to ensure a continuous pressure match over the entire flight spectrum, some method of varying the area ratio must be employed. This can be accomplished by varying either the nozzle throat area or the exit plane area.

For an engine with an afterburner, the governing flow parameter which is a constant for choked flow is

$$\dot{w}\frac{\sqrt{T_{T_8}}}{P_{T_8}A*}$$

Looking at this relationship, we can analyze the changes that occur when the afterburner is ignited at supersonic speeds. First, mass flow rate ($\dot{w}$) is almost constant, increasing only by the fuel added to the afterburner which is a small fraction of the total mass flow. On the other hand, T_{T8} goes up dramatically when the afterburner is lit. Since P_{T8} changes little in the afterburner nozzle, A* must increase to keep the flow parameter a constant. Thus for supersonic flight, A_e/A^* must be variable to maintain optimum pressure balance for peak performance.

7.8.6.4 TWO-DIMENSIONAL NOZZLES

Research is presently being conducted on the use of two-dimensional exhaust nozzles. Tests have shown that there is no degradation in thrust with a 2-d nozzle from that of the axisymmetric nozzle for a given engine.

An added benefit with a 2-d nozzle is the capability for thrust vectoring. Thrust vectoring capability would be beneficial in helping to control glidepath angle during approaches. Vectored thrust might also be useful during spin recoveries.

Thrust reverses are more easily incorporated in a 2-d system than an axisymmetric one. The 2-d nozzle system can be installed with aerodynamically clean contours and may have a more efficient cooling system than its axisymmetric counterpart.

7.8.6.5 JET NOZZLE VELOCITY

At the higher throttle settings and airspeeds encountered during normal aircraft operation, the nozzle throat will probably be "choked" most of the time, which means that the gases passing through the convergent section of the nozzle will be at or near the speed of sound. When the nozzle is choked, the only variation in the exit velocity of the gases will be due to changes in the engine exhaust gas temperature. Whenever the nozzle is not choked, varied atmospheric conditions will cause some change in jet nozzle velocity. As can be seen by the thrust equations, changes in the exhaust gas or nozzle velocity (V_{10}) will affect thrust.

7.8.6.6 NOZZLE EFFICIENCY

Since losses are present in an actual nozzle flow process, it is desirable to examine nozzle flow with friction. Let us consider a nozzle which operates between a pressure at inlet, P_{T5}, and a lower pressure at exit, P_{10}. Figure 7.84 illustrates such a nozzle with its expansion process on a T-s or h-s plane.

As pointed out previously, the function of the nozzle is to transform the high pressure-temperature energy (enthalpy) of the gases at this entrance position (Point 5) into kinetic energy. This is done by decreasing the pressure and temperature of the gases in the nozzle. Referring to Figure 7.84, it is evident that the maximum amount of transformation will result with an isentropic process between the pressures at entrance and exit. Such a process is illustrated as the Path 5-10. Now, when nozzle flow is accompanied with friction, an increase in entropy results, and the path is curved as illustrated by Line 5-10. It is noted that the actual enthalpy change is somewhat less than the enthalpy change for an isentropic process. The difference in the enthalpy change between the actual process and ideal process is due to friction.

Since an actual nozzle does not have as great an enthalpy drop as an ideal nozzle, the relative merits of the ideal nozzle and the actual nozzle can be compared using the ratio of the two enthalpy drops between the same pressure limits. This ratio is

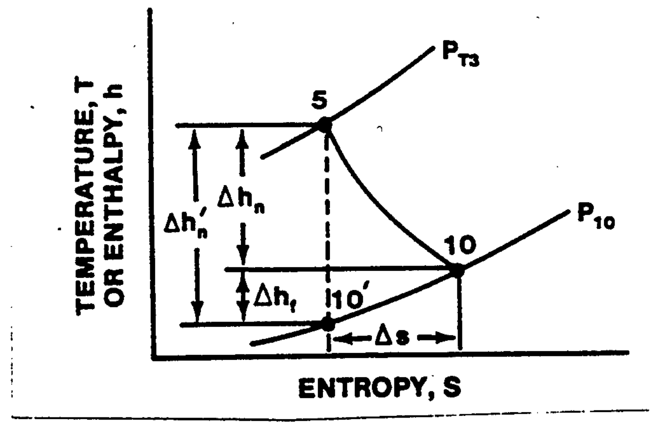


FIGURE 7.84. COMPARISON OF IDEAL AND ACTUAL NOZZLE EXPANSION ON A T-s OR H-s PLANE

defined as the nozzle adiabatic efficiency and is

$$\eta_n = \frac{h_{T5}-h_{10}}{h_T-h'_{10}} = \frac{\Delta h_n}{\Delta h'_n} = \frac{C_p(T_{T5}-T_{10})}{C_p(T_{T_5}-T'_{10})}$$

The value of ηn for a good nozzle should be somewhere in the range of .9 to .96.

7.8.7 THRUST AUGMENTATION

To achieve better takeoff performance, higher rates of climb, and increased performance at altitude during combat maneuvers, there has always been a demand for increasing the thrust output of aircraft powerplants for short intervals of time. In addition, a recent desire for sustained operation at supersonic speeds requires a significant increase in the thrust/frontal area output of aircraft engines. Two basic methods of providing thrust augmentation for turbojet and turbofan engines will be discussed: (1) afterburning or tailpipe burning (by far the most popular current type of thrust augmentation); and (2) water injection (either at the compressor inlet or in the combustion chamber). Each method provides a substantial thrust increase over the normal engine thrust but also requires a considerable increase in liquid consumption. Because each method produces overall engine efficiencies which are less than that of the normal engine, thrust augmentation devices should normally be operated for only short time intervals. The exception to this is for sustained supersonic operation where the only method of achieving the condition is with thrust augmentation. Each method of augmentation also makes the basic engine more complex--additional controls are required, engine geometry is changed, and special liquids with suitable lines and controls are required. Despite these disadvantages, the requirements for thrust augmentation have provided sufficient stimulus for much development effort. This effort, which continues, has resulted in reliabilities for the thrust augmentation system essentially equivalent to the basic engine.

7.8.7.1 THE AFTERBURNER

Turbine temperature limits the basic engine fuel/air ratio to about 0.025. As a result, the gases which are exhausted form the turbine section are primarily air; thus, if a suitable burner is installed between the turbine and exhaust nozzle, a considerable amount of fuel can be burned in this section to produce temperatures entering the nozzle as high as 3500°F. The increased temperature greatly augments the exhaust gas velocity and hence provides a thrust increase.

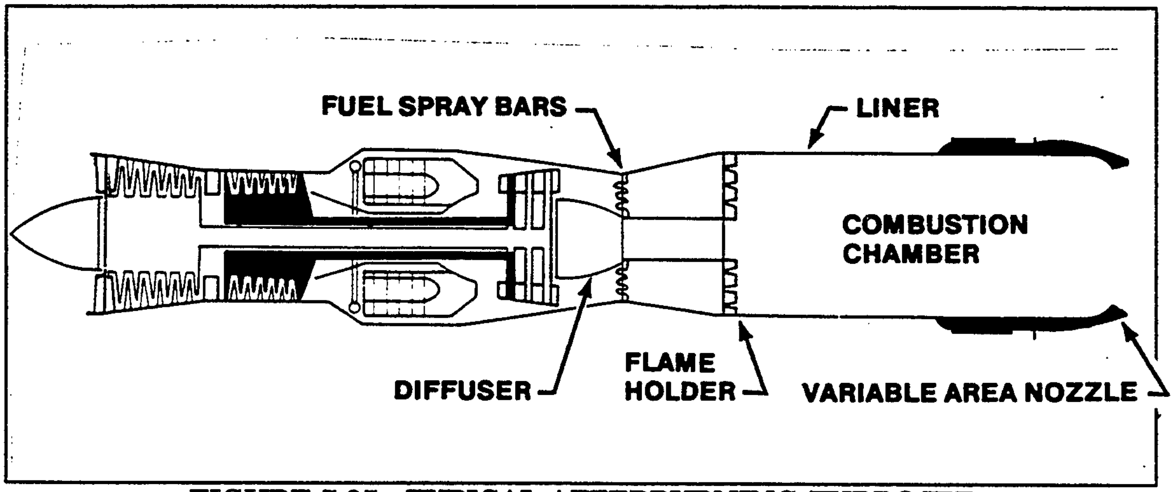


FIGURE 7.85. TYPICAL AFTERBURNING TURBOJET

Figure 7.85 shows a typical afterburning turbojet engine. The figure identifies the various afterburning components and reveals that the afterburner meets the basic requirements of the normal combustion chamber. First, the air discharging from the turbine must be slowed down to a low velocity so that combustion can be stabilized. To decrease the air velocity adequately, a diffuser section known as the turbine discharge diffuser, is placed between the turbine and the burner section (note the gradual area increase between the afterburner wall and the tail cone downstream of the turbine). Fuel is injected through a spray nozzle system which produces a mist that will thoroughly mix with the air.

Provisions must also be made for igniting the fuel-air mixture. Ignition is accomplished either by spark igniters which function in the same manner as in the normal burner or by the so-called "hot streak" method. The latter scheme requires that a small-diameter, high-velocity fuel stream be squirted from the main combustion chamber through the turbine blades into the afterburner. This small fuel stream, literally a hot streak, is ignited in the main burner and its flame volume increases progressively as it flows into the afterburner where it ignites the fuel-air mixture. The turbine blades are not overheated by the hot streak because of its relatively low energy content, and since a portion of the fuel vaporizes in the fuel stream, some cooling is provided; furthermore, the hot streak is operated only briefly.

To maintain a flame after ignition is accomplished, the afterburner requires the equivalent of the primary combustion zone in the normal burner. This is accomplished by a series of so-called "flame holders", which are usually V-shaped gutters mounted concentrically about the longitudinal axis of the burner.

To allow time and space for good combustion, the afterburner must contain a volume which is considerably larger than the normal combustion chamber. This extra volume requirement is necessary, because the after-burner consumes up to three times as much fuel as the normal burner. In general, when an afterburner is added to an engine, the overall engine length is about doubled. External cooling can be accomplished by producing airflow between the afterburner and aircraft structure. Insulating blankets may also be wrapped around the outer shell to provide an additional restriction to heat flow.

7.8.7.1.1 Afterburner Performance. It was mentioned that it is necessary to increase the nozzle area when the afterburner is operating. It is desirable to examine the reasons for the necessity of increasing the nozzle exit area. Assume that an afterburner engine is initially operated nonafterburning with full-throttle conditions (military thrust) so that the nozzle flow is choked. When the afterburner is turned on, each pound of air passing through the afterburner grows in volume by a factor of about two because of the increase in temperature. Therefore, in order not to reduce the mass flow rate through the nozzle, the nozzle must be opened to compensate for the increased gas volume.

A good afterburner installation is one which produces no uncalled for changes in the operating conditions of the basic engine components when the afterburner is turned on and off; that is, the basic engine should not be able to feel the difference between afterburner and nonafterburner operation.

Figure 7.86 shows the cycle diagram of a turbojet engine equipped with afterburner on the h-s plane. The process lines up to the turbine section are the same as for a basic engine. The afterburner process 5-6 is ideally a constant pressure process, but the internal drag losses and momentum pressure loss produce a total pressure drop such that P_{T6} is about 5% less than P_{T5}. It is apparent that more thrust can be realized per pound of air by examining the relative magnitude of the enthalpy drops across the normal nozzle 5-ef (see dotted line) and across the afterburner nozzle 6-ef.

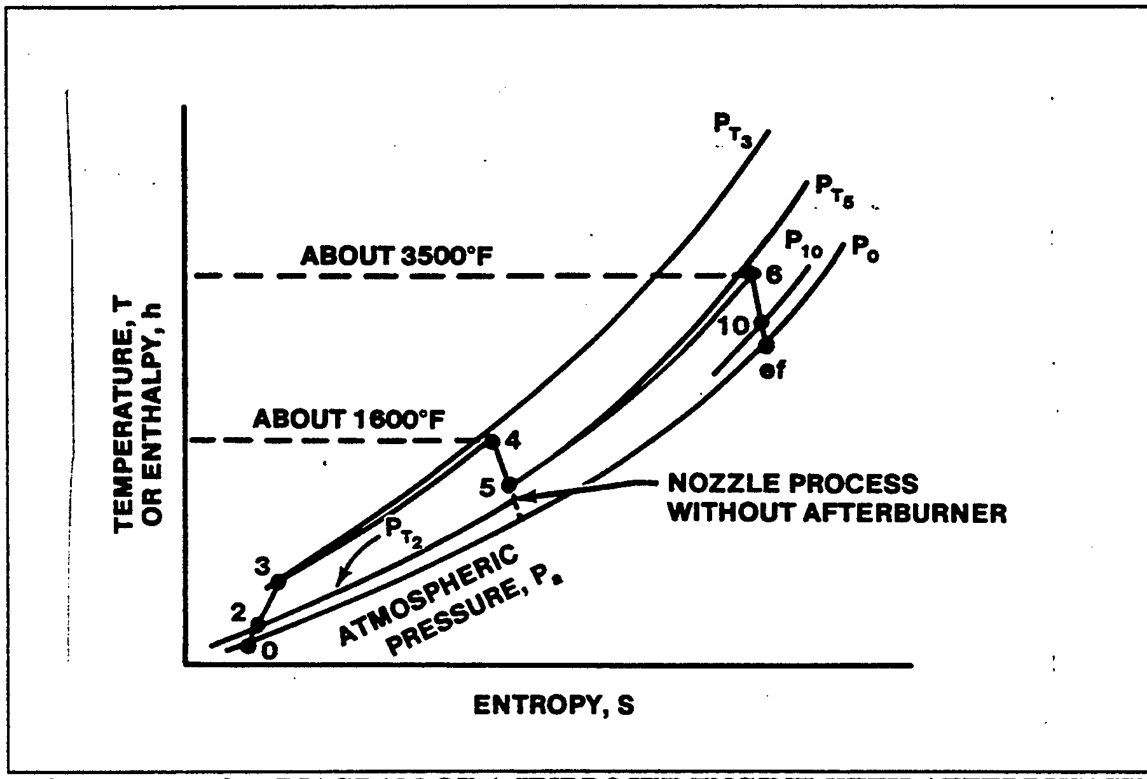


FIGURE 7.86. h-s DIAGRAM OF A TURBOJET ENGINE WITH AFTERBURNER

Afterburner performance is normally expressed in terms of the ratio of augmented thrust (thrust available with afterburner on) to military thrust without afterburner. Expressing this ratio in terms of gross thrust

$$\frac{(F_G)_a}{F_G} = \left[\frac{(W_{10})_a}{W_{10}} \quad \frac{(V_{10})_a}{V_{10}}\right] \tag{7.67}$$

where subscript a refers to the augmented condition.

Table 7.13 summarizes the characteristics of some current U.S. and British afterburning turbojet engines.

TABLE 7.13
CHARACTERISTICS OF SOME CURRENT U.S. AND BRITISH AFTERBURNING TURBOJET ENGINES

Engine Designation	Manufacturer*	After-burner Take-off Thrust (lb)	On Takeoff SFC (lb/ hr/ lb)	After burner off Take-off Thrust (lb)	Weight (lb)	Dia-meter (in.)	Leng th (in.)
J85-GE-5	G.E.	3850	2.20	2500	538	20	104
J79-GE-8	G.E.	17000	2.00	12000	3630	32	208
J57-P-20	P.& W.	18000	2.35	10700	4750	40.5	267
J-P-19W	P.& W.	26500	2.20	16100	5960	43	259
J58-P-2	P.& W.	32000	----	25500	7000+	45	---
YJ93-GE-3	G.E.	30000	----	----	5800+	52.5	237
Olympus 201R	B.S.	24000	----	17000	4550	42	354
Orpheus 12SR	B.S.	8170	1.62	6740	1560	32	181
Gyron Jr.	D.H.	14000	1.80	10000	3100	32	191
Avon (300)	R.R.	16600	2.0	12500	3800	42	256

*G.E. - General Electric; P.& W. - Pratt & Whitney; B.S. - Briston-Siddeley; D.H. - DeHavilland; R.R. - Rolls-Royce.

7.8.7.1.2 Afterburner Screech Liners. Afterburners are occasionally subject to a type of combustion instability known as "screech". Screech is a condition of periodic, violent pressure fluctuations in the afterburner duct, resulting from cyclic vibration due to unsteady release of combustion energy. Cyclic vibration is a pressure variation of high frequency, 400 - 600Hz, and intensity, which can sometimes attain destructive proportions. Screech is characterized by intense noise. When screech occurs, heat transfer rates and temperatures of the afterburner parts increase greatly. Moderate to severe screech can cause rapid deterioration or failure of the flameholders or the afterburner duct. Screech is controlled by placing so-called "screech liners" in the duct. These are inner steel sleeves which are literally perforated by thousands and thousands of small holes. The special design of the sleeves tends to absorb the periodic, combustion-energy fluctuations, and to prevent random pressure fluctuations from developing into cyclic vibrations of large amplitude.

7.8.7.1.3 Rumble. With the introduction of the mixed-flow augmentor in turbofan engines, a type of low frequency instability known as rumble or chugging became a serious problem. Rumble is a periodic afterburning combustion instability (pressure oscillations fed by the combustion process) usually occurring at high fuel-air ratios at flight Mach and altitude when low duct inlet air temperatures and pressure exist. This instability usually leads to afterburner blowout and/or fan surge and engine stall. The frequency of oscillation usually lies between 30 and 200 Hz.

Even subtle changes in flameholder designs have altered the rumble characteristics of a turbofan engine. With some experience at hand, the design engineer has successfully produced "fixes" for unstable conditions. Redistribution of the fuel-to-air mixture ratio has worked, and deriching the fan duct has lessened rumble problems in the past.

As a result of analytical and experimental efforts, the following major conclusions have been reached:

> Rumble was identified as a system problem in which the airflow dynamics couple with the combustion process.
>
> Experimental rig tests identified fuel distribution as a rumble contributor.
>
> The most significant driver of rumble is the falloff in combustion efficiency as fuel-air ratio is increased.
>
> The variations of augmentor efficiency caused by pressure, velocity, and temperature were identified as minor rumble drivers.

7.8.7.2 WATER INJECTION

The sensitivity of gas turbine engines to compressor inlet temperature results in appreciable loss of thrust available for takeoff on a hot day. It is frequently necessary, therefore, to provide some means of thrust augmentation for non-afterburning engines during takeoff on warm or hot days. Ten to thirty percent additional thrust (power) can be gained by injecting water into the engine, either at the compressor air inlet or the combustor inlet.

When water is added, thrust or power augmentation is obtained principally by cooling the air entering the engine by means of vaporization of the water introduced into the airstream. Cooling the air has the effect of reducing the compressor inlet temperature. The reduction in temperature increases the air density and the mass airflow. More and cooler air to the combustors permits more fuel to be burned before limiting turbine inlet temperatures are reached, which, in turn, means more thrust.

7.8.7.3 SUMMARY OF THRUST AUGMENTATION DEVICES

Table 7.14 presents a summary of the augmented thrust ratios and specific liquid consumptions obtainable from typical turbojet engines. Remarks are included in the table to summarize the application and limitations of the various thrust augmentation devices.

TABLE 7.14

SUMMARY OF PERFORMANCE DATA OF TYPICAL THRUST AUGMENTATION DEVICES

Method	Thrust Ratio and Liquid Consumption	Sea Level M=0	Sea Level M=2.0	35,000 FEET M=0	35,000 FEET M=2.0	Remarks
Afterburner	F_{n_a}/F_n	1.5	3.0	1.5	2.5	Limited by Stoichiometric mixture or thermal choking.

	slc	2.4	2.4	2.0	2.2	In service use on an extensive basis
Water injection at compressor	F_{n_a}/F_n	1.4	2.6	1.2	2.0	Requires separated liquid. Limited by air saturation at
inlet	slc	3.2	9.0	2.4	6.0	compressor outlet. In service use on limited basis, primarily for thrust restoration at takeoff.
Water injection into combustor	F_{n_a}/F_n	1.3	2.4			Not practical on operating near stall line.
	slc	8.0	15.0			Limited by compressor stall. In service use on limited basis.

7.9 OVERALL ENGINE ANALYSIS

The individual components of gas turbine engines have been discussed in detail. To summarize, it might be beneficial to examine the variation of gas properties throughout the overall engine. Figure 7.87 is a sketch of a typical axial flow turbojet engine showing the variation of T, T_T, P, P_T, V, and thrust force through each engine component.

The figure applies to an in-flight condition where the diffuser develops a positive pressure rise. Thrust force variation is shown below the engine-positive slopes indicate that forward thrust forces act on the engine, and conversely, negative slopes indicate rearward thrust forces. For example, the axial flow compressor receives a forward thrust force which increases in magnitude as the flow progresses through the stages. The unbalanced engine force, labeled F_n, is the net force that is delivered to the airframe for propulsion.

Figure 7.88 shows an enthalpyentropy diagram for a real engine with reasonable irreversible effects and typical temperatures, for a compressor pressure ratio of ten. Afterburning and non-afterburning processes are shown, with the exhaust pressure equal to ambient pressure in both cases.

The process beings with atmospheric air at h_0, P_0 INF. By virtue of the relative (flight) velocity between the air and the engine, this air has a stagnation enthalpy h_{T0}, higher than h_0. Further, since no work or heat transfer occurs between ∞ and 2, the stagnation enthalpy is constant through Station 2. The air is externally decelerated from 0 to 1. For all practical purposes this external deceleration is an isentropic process (unless an external shock occurs), hence State 1 is on an isentrope with State 0 and $P_{T1} = P_{T0}$. From 1 to 2 the air is further decelerated, accompanied by an increase 0 in entropy though frictional effects. Note that this results in a decrease in stagnation pressure. From 2 to 3 the air is compressed, again with an increase of entropy due to irreversibilities in the compression process. State 3' is defined as that state which would exist if the air could be compressed isentropically to the actual outlet stagnation pressure. State 3 is the actual outlet stagnation state.

From Station 3 to Station 4, some fuel is mixed with the air and combustion occurs. Strictly speaking, the fluid composition changes between these stations, and a continuous path between them should not be shown. However, since the fluid characteristics do not change markedly, there is no difficulty in showing the two substances on different portions of the same diagram. The stagnation pressure at 4

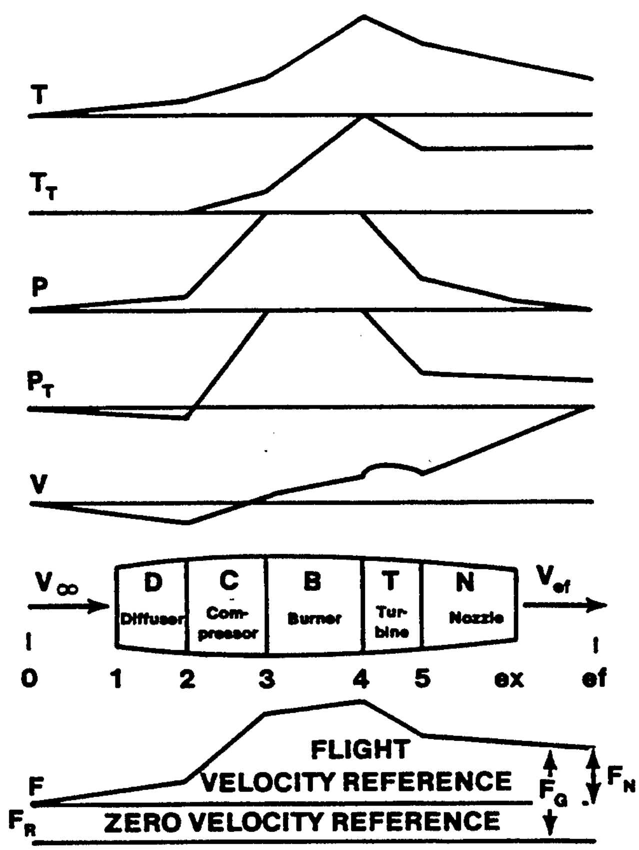


FIGURE 7.87. VARIATION OF GAS PROPERTIES THROUGH A TURBOJET ENGINE DURING FLIGHT

must be less than at 3 because of fluid friction, and also because of the drop in stagnation pressure due to heat addition at finite velocity. As we shall see later, it is advantageous to make T_{T4} as high as material limitations will allow.

From 4 to 5, the fluid expands through the turbine, providing shaft power equal to the shaft power input to the compressor (plus any mechanical losses or accessory power). Since no work or heat transfer occurs downstream of Station 5, the stagnation enthalpy remains constant throughout the rest of the machine. State 6 depends on the geometry involved, but P_{T6} must be less than P_{T5}. The exhaust pressure P_{10} generally equals the atmospheric pressure P_0, but it may be different if the exhaust flow is supersonic. If the afterburner is operative, the fluid is raised in temperature to State 6A, after which it expands in the nozzle to State 10A.

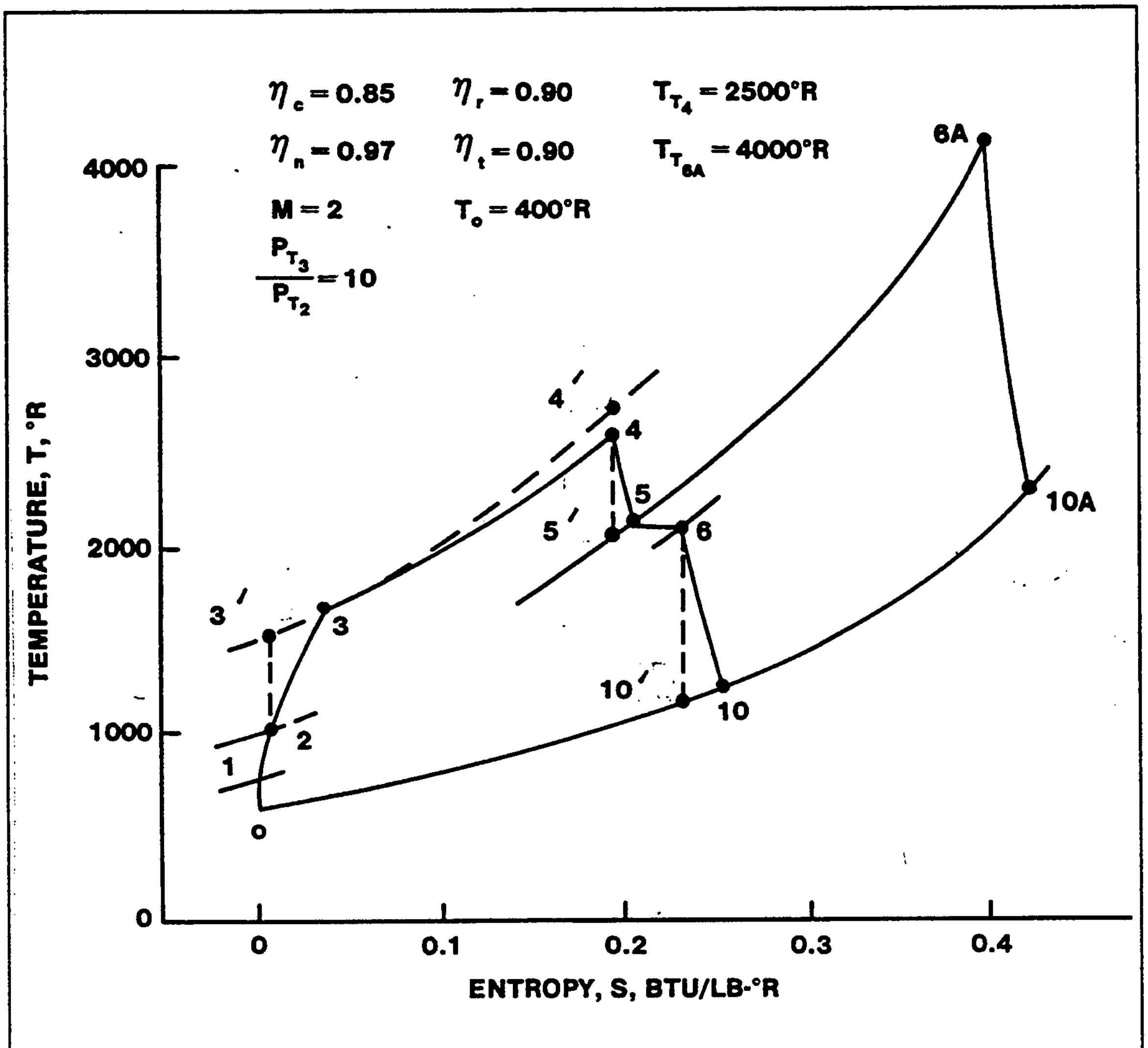


FIGURE 7.88. T-s DIAGRAM FOR TYPICAL TURBOJET ENGINES

Again it can be seen that the exhaust kinetic energy is the relatively small difference between the total available enthalpy drop from State 4 and the compressor work input. For a given compressor-pressure ratio, irreversibilities increase the compressor power requirement while at the same time increasing the necessary turbine pressure drop. Both effects decrease the exhaust kinetic energy, so that overall performance may be expected to be very sensitive to compressor and turbine performance.

7.9.1 EFFECT OF HUMIDITY ON ENGINE PERFORMANCE

Humidity affects turbojet and turbofan engine performance because the mixture of water vapor and air has gas properties which differ slightly from those of dry air. The primary reason for this difference is the fact that water vapor is lighter than air. This is evident from their relative molecular weights: H_2O = 18, and air = 29.0. However, even at 100% relative humidity, the effect on engine performance is only about 1%. Many Flight Manuals give the method for correcting for humidity in those cases where extreme accuracy is desired.

7.9.2 THRUST HORSEPOWER

Thrust horsepower is defined as the rate of doing work. Therefore thrust horsepower can be expressed as

$$THP = F_n V_0 \tag{7.68}$$

The units of the above equation are ft lb/sec. Care must be taken to convert horsepower units depending on the units in which velocity is given for a particular calculation.

For the propulsion of aircraft, thrust horsepower is used to overcome drag. When the maximum thrust horsepower is equal to the power required for steady level flight, the maximum velocity for a particular engine-aircraft combination is achieved.

7.9.3 SPECIFIC IMPULSE

Specific impulse is another term for measuring fuel and thrust efficiency. It is generally used for rockets, but occasionally used for turbine engines. Specific impulse is defined as the ratio of thrust to fuel consumed, i.e., $I_s = F_n / \dot{w}_f$, and is the reciprocal of thrust specific fuel consumption. High pressure cryogenic rocketry provides I_s in excess of 400 sec. The $\dot{w}_f$ for a rocket includes both fuel and oxidizer since there is no compressor. For comparison, a jet engine with a TSFC of 0.5 would have an I_s = 1/0.5 hr = 2 x 3600 sec/hr = 7200 sec. or 18 times better than a rocket. Therefore, vehicles operating in the atmosphere for any flight duration should obviously use the air for an oxidizer.

7.10 ENGINE OPERATIONAL CHARACTERISTICS

The foregoing sections have described the features of the three basic, gas turbine engine types: the turbojet, the turboprop and the turbofan. Particular attention has been given to the turbojet because this is the most common configuration. Less has been said about the operational characteristics of the other two engine types or the particular use to which each is best suited. Like engines of all types, each of the three engine configurations has limitations and advantages.

7.10.1 ADVANTAGES AND DISADVANTAGES OF THE TURBOJET

Because the efficiency of the straight turbojet is sustained at high altitude and high airspeed, engines of this type are ideal for high-flying, high-speed aircraft that operate over a sufficient range to make the climb to their best operating altitude worthwhile. Exceptionally high thrust at low airspeed is not a turbojet characteristic. Hence, aircraft powered with these engines require a relatively long takeoff roll or a low wing loading. Turbojet thrust specific fuel consumption (TSFC) is higher than that of a turboprop or turbofan; this disadvantage decreasing as the altitude and airspeed increase. In addition to their high-speed capabilities and the very high altitudes at which they can operate, axial compressor turbojet engines present a relatively small frontal area. The smaller diameter does mean that turbojet engine-nacelle ground clearance is less of a problem to the aircraft designer than in some aircraft, particularly when it is necessary that an engine be mounted in a pod beneath a wing.

7.10.2 TURBOPROP CHARACTERISTICS

A turboprop engine has some fundamental characteristics which make it quite different from a turbojet from the standpoint of the pilot.

A turboprop engine combines the advantages of a turbojet engine with the propulsive efficiency of a propeller. The turbojet engine derives its thrust by a rapid acceleration of a relatively small mass of air. The turboprop develops propulsive force by imparting less acceleration to a relatively small mass of air. The turbine of a turbojet engine extracts only the necessary shaft horsepower to drive the compressor and the accessories. The turbine of a turboprop is designed to absorb large amounts of energy from the expanding combustion gases in order to provide not only the power required to satisfy the compressor and other components of the engine, but to deliver the maximum torque possible to a propeller shaft, as well. Propulsion is produced through the combined action of a propeller at the front and the thrust produced by the unbalanced forces created with the engine that result in the discharge of high-velocity gases through a nozzle at the rear. The propeller of a typical turboprop engine is responsible for roughly 90% of the total thrust under sea level, static conditions on a standard day. This percentage varies with airspeed, exhaust-nozzle area and, to a lesser extent, temperature, barometric pressure and the power rating of the engine. The power supplied to the propeller is measured as shaft horsepower (shp), to which must be added the effect of jet thrust when the total power output or equivalent shaft horsepower (eshp) of a turboprop engine is calculated.

Although some turboprop engines employ a compressor of the centrifugal type, larger, high-performance models almost invariably require the greater efficiency and higher compression ratios attainable only with an axial flow compressor. The compressor may be either of a single or dual rotor design; the latter having both a low pressure compressor and a high pressure compressor. When a single compressor is used, the propeller reduction and drive gear is usually connected directly to the compressor shaft, and, when a dual, or so-called split, compressor is used, it is connected to the low pressure rotor. Sometimes, the propeller is driven independently of the compressor by a free turbine of its own.

In spite of the fact that it is more complicated and heavier than a turbojet engine of equivalent power, the turboprop will deliver more thrust up to moderately high subsonic speeds (Figure 7.89a). This advantage decreases as airspeed increases. In normal cruising speed ranges, the propulsive efficiency of a turboprop remains more or less constant, whereas the propulsive efficiency of a turbojet increases rapidly as airspeed increases. The spectacular performance of the turboprop during takeoff and climb is the result of the ability of the propeller to accelerate a large mass of air at relatively low flight speed.

If it is assumed that the fuel flow of a turbojet and a turboprop of approximately the same size will be substantially the same under similar conditions, it follows that the one delivering the most thrust will have the lower TSFC. Because of its propeller, this will be the turboprop version of a basic gas generator. The TSFC for a turbofan version of the same gas generator will fall between the TSFC for the turboprop and the TSFC for a turbojet (Figure 7.89b).

The turboprop attains its most economical operation at a somewhat lower airspeed than a turbojet of equivalent power. Usable power at a high efficiency is produced only when the engine is operating within a narrow range of high RPM. The efficiency of a vane-type compressor, whether centrifugal or axial, is dependent upon high RPM. Progressively larger amounts of turboprop power are obtained by increasing the propeller blade angle and fuel flow rather than by increasing RPM.

Figure 7.90, portraying typical turboprop performance as it relates to throttle setting, gives an indication of what to expect when this type of engine is operated. The curves, as presented, are more or less representative of the characteristics of a Pratt & Whitney Aircraft PT-2 or T-34 turboprop engine. The curves will change somewhat when different fuel controls are used. Typical curves for other turboprop engines (the

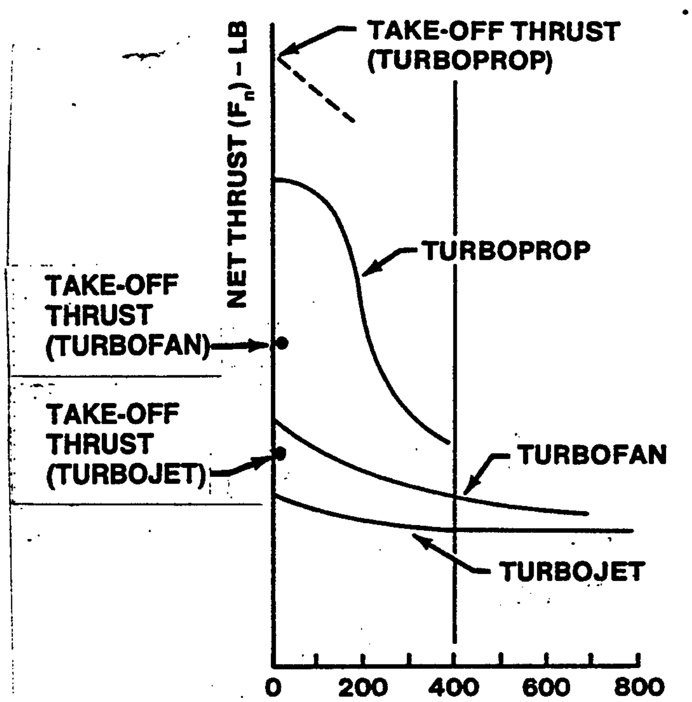
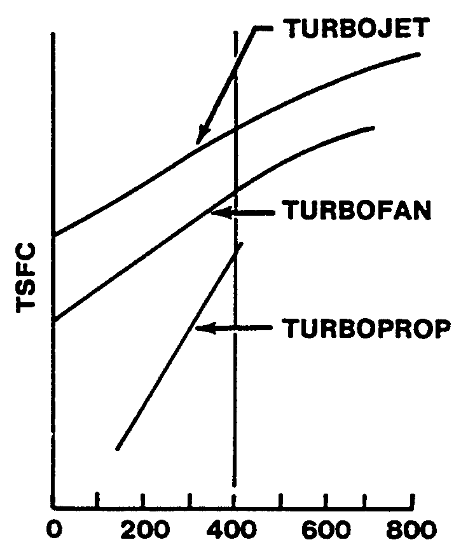


FIGURE 7.89a. COMPARATIVE NET THRUST AT SEA LEVEL



7.89b. COMPARATIVE THRUST SPECIFIC FUEL CONSUMPTION

RPM curve, in particular) will not be the same, although they, in all probability, will reflect the same general engine characteristics.

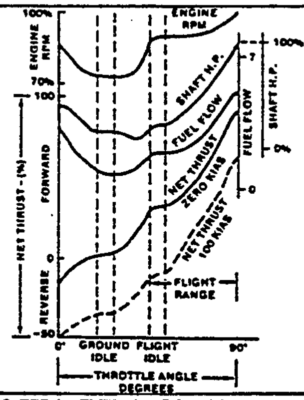


FIGURE 7.90. TYPICAL P&WA PT-2 OR T-34 TURBOPROP PERFORMANCE

7.10.2.1 THE TURBOPROP PROPELLER

The propeller of a turboprop engine retains most of the essential features common to those employed on large piston engine installations. Both hydromechanical and electrically controlled propellers are in current use. From an operational standpoint, the differences between the two types are minor.

The fuel control operates in conjunction with a propeller governor in a turboprop engine. The propeller and engine RPM are mechanically governed in the flight operating range. In the Beta or ground operating range, propeller pitch varies with throttle position. Propeller blade angles from full feather to full reverse pitch may be obtained throughout the entire range of operative RPM. Because of the high RPM of a gas turbine engine, a reduction gear arrangement, is usually used.

The blades of a turboprop propeller must have very rapid pitch-changing characteristics. The narrow RPM operating range of the engine requires that the propeller blades change angle at a much more rapid rate than is required in the case of the reciprocating engine. The blades of a turboprop propeller must change from about 5° to 45° in only 10% of the RPM range (Figure 7.91). Translated into flight-operating-technique, this means that the turboprop engine is very sensitive to throttle movement.

The turboprop propeller blade angle at Flight Idle is small (approximately 20) during a glide at minimum power. The turboprop aircraft, consequently, can have high aerodynamic drag, provided that the fuel control and propeller governor are adjusted to provide this characteristic during glide and approach. High drag will result in a rapid rate of descent.

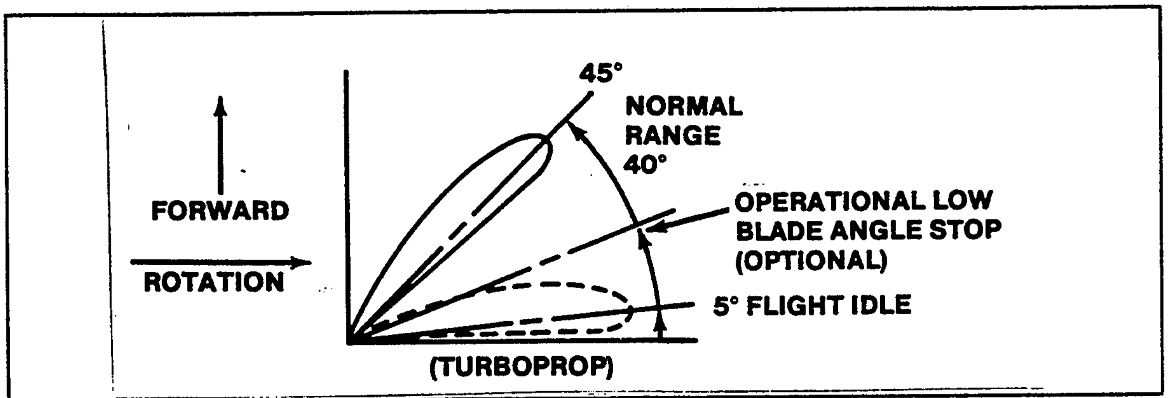


FIGURE 7.91. PROPELLER BLADE ANGLE VARIATION

7.10.3 THE TURBOFAN ENGINE

In principle, the turbofan version of an aircraft gas turbine is the same as the turboprop, the geared propeller being replaced by a duct-enclosed fan driven at engine speed. One fundamental, operational difference between the turbofan and the turboprop is that the airflow through the fan of the turbofan is controlled by design so that the air velocity relative to the fan blades is unaffected by the airspeed of the aircraft. This eliminates the loss in operational efficiency at high airspeeds, which limits the airspeed capability of a turboprop engine. Also, the total airflow through the fan is much less than that through the propeller of a turboprop.

Because of its greater inlet-area, the fan draws inconsiderable more air than the compressor of the turbojet. However, a great deal of this air, after being compressed by the fan, is released through the fan exit ducts, completely bypassing the burner and turbine sections. This bypass air is ducted outside the basic engine because the air has already been accelerated by the fan and has therefore served its purpose of providing additional thrust; the same kind of additional thrust that would be gained from air passing through the propeller of a turboprop or reciprocating engine.

One might ask, "Why not use a conventional propeller instead of a fan"? There are two reasons. First, the engine and propeller combination in a propeller-driven aircraft commences to lose efficiency rather rapidly at airspeeds above 400 knots at cruising altitude, while turbofan engines produce thrust efficiently at the airspeeds flown by present-day commercial aircraft (Figure 7.89a). Secondly, the complexity and weight of the propeller reduction gearing and the intricate propeller governing feature of a turboprop are completely eliminated in a turbofan. The turbofan is therefore not only lighter than a turboprop, but, of even more importance, the turbofan can never be plagued by any of the malfunctions to which the propeller and its associated systems in a turboprop are sometimes susceptible. Here, then, lie the main advantages of a turbofan over a turboprop version of the same gas generator.

Ducting the fan exhaust overboard instead of through the combustion chamber enables a turbofan engine to obtain low specific fuel consumption. From the time that the air enters a turbojet engine until the burned gases leave the combustion chamber, temperatures are progressively rising. First, the compressor works on the air, raising its temperature, and then the combustion process adds energy in great quantities. To do all this, of course, takes energy which must come from the fuel burned. If the fan exhaust were passed all the way through the engine, its temperature would be greatly increased, only later to be wasted as heat energy thrown out the engine exhaust nozzle. Not having to heat the air that passes only through the fan serves to add to the efficiency of a turbofan engine.

Like the turboprop, a turbofan accelerates a relatively large mass of air to a relatively low velocity. When large air masses are accelerated at reduced velocity, the propulsive efficiency of an engine is vastly improved. Therefore, a turbofan engine operates more efficiently and thus operates at a lower TSFC than a turbojet engine of similar size (Figure 7.92).

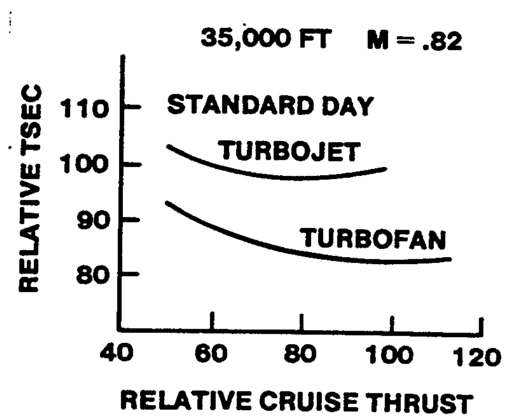


FIGURE 7.92. RELATIVE PERFORMANCE AT MAXIMUM CRUISE

A fan engine accelerates much larger quantities of air than a turbojet. This enables the turbofan to produce more thrust than a turbojet at low airspeeds, such as during climb (Figure 7.93), or even when an aircraft is standing still on the ground. This thrust increase for a turbofan in comparison to a similar basic turbojet varies as m_{TF}/m_{TJ} where m_{TF} is the mass flow rate through the turbofan engine and m_{TJ} is the mass flow rate through the basic turbojet (Reference 7.1). For this reason, an aircraft powered by turbofan engines will have more available thrust for takeoff (Figure 7.94) and therefore will not need as much distance for takeoff as will the same aircraft powered by turbojets of the same approximate size. By the same virtue, the aircraft with the turbofans can take off at a much higher gross weight than can the aircraft powered by turbojets. This feature, combined with the much higher speed characteristics of the turbofan when compared with a turboprop, makes the engine a very attractive powerplant for passenger and cargo type aircraft, whether short of long range.

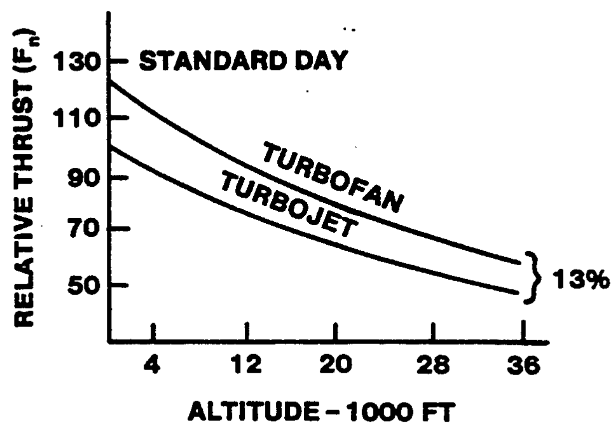


FIGURE 7.93. RELATIVE MAXIMUM CONTINUOUS-THRUST COMPARISON DURING CLIMB

Still another advantage of the turbofan is a lower engine noise level. This is because the velocity of the gases as they leave the engine tailpipe is lower than that for a turbojet engine of comparable size. The decrease in velocity is due to the fact that a turbofan engine has an additional turbine stage which extracts power from the exhaust gases to drive the fan. Less velocity results in less noise.

7.11 PROPELLER THEORY

A propeller is a device which absorbs the horsepower from the engine and generates a thrust which propels the aircraft. The propeller can be considered as a device which

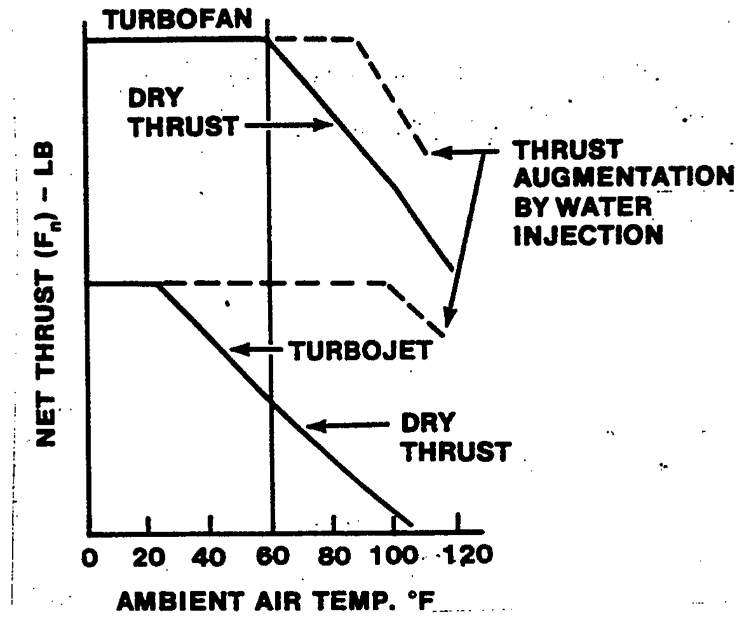


FIGURE 7.94. SEA LEVEL STATIC TAKEOFF THRUST

accelerates the air passing by it, thereby generating thrust by the rate of change of momentum, or as a rotating wing that generates a lift which is a thrust.

It is obvious from Figure 7.95 that aircraft operating at vehicle speeds less than 300 knots will be using propellers as the most efficient means of propulsion.

The efficiency of the various types of propulsion systems depends primarily on the disk loading of the system, which essentially means that a propulsive system is most efficient when it accelerates a large mass of air through a small velocity increment. As the mass of air decreases and the velocity increment increases, the overall efficiency decreases as is obvious from he curves in Figure 7.95. The reason that the rotor, which is very efficient at low speeds, peaks out at approximately 150 knots airspeed is because of retreating blade stall and advancing tip compressibility

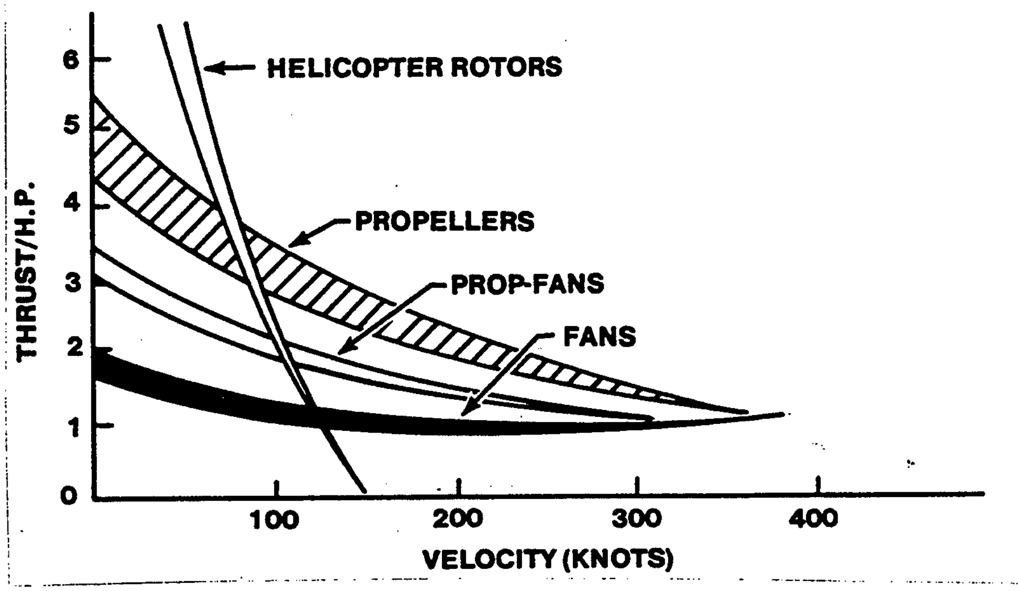


FIGURE 7.95. THRUST PER HORSEPOWER VERSUS VEHICLE SPEED

problems. Low disc loaded propellers of large diameter, and therefore high tip velocities, have high speed limitations due to tip Mach compressibility problems Not only are propellers much more efficient than jets at low air speeds, but with proper design and low tip speeds, they can be much quieter.

Regardless of the type of propulsive method, the thrust is produced as a consequence of Newton's Second Law:

$$Force = Mass\ (Acceleration) = \frac{dv}{dt}$$

which states that the force or thrust produced is equal to the rate of change of momentum.

7.11.1 MOMENTUM THEORY

The simplest theory describing propeller action is the momentum theory originally used by Froude in the last century in his study of screw propellers for ships. This theory assumes that the propeller disc is replaced by an actuator disc that has an infinite number of blades and is capable of producing a uniform change in velocity through the disc. It also assumes that the flow has no rotational components, that there are no hub or tip losses, and that the actuator disc has no profile drag.

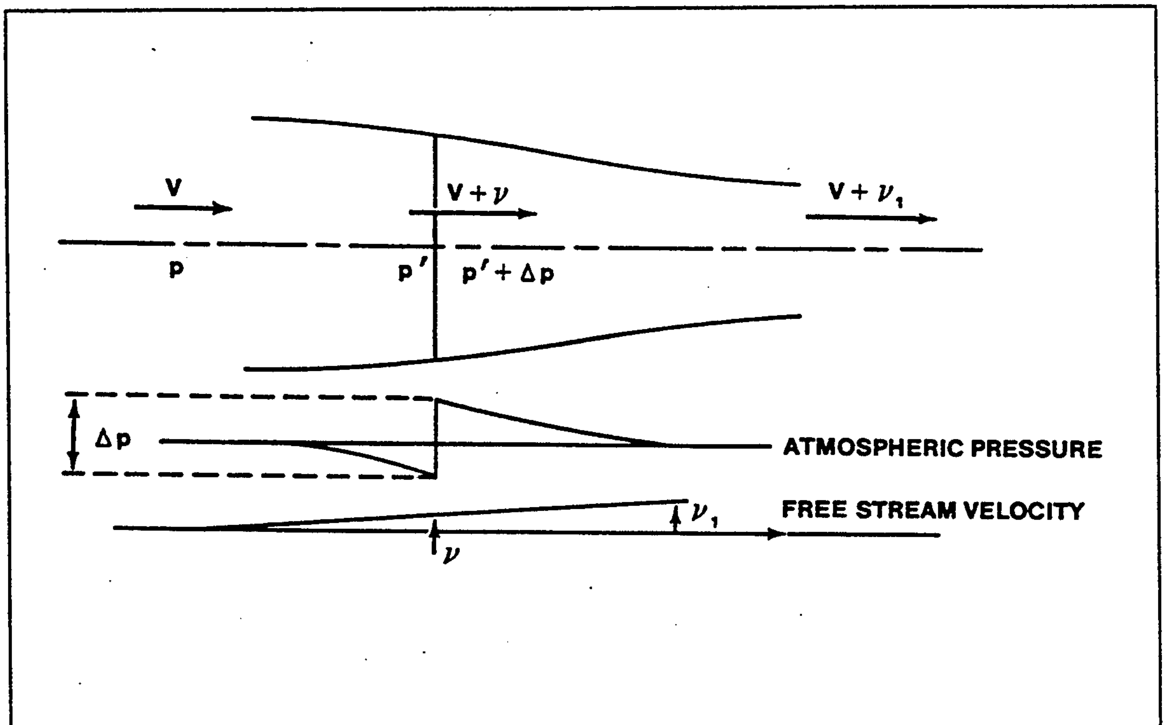


FIGURE 7.96. PROPELLER MOMENTUM THEORY

Bernoulli's equation applies in front of and behind the disc, but due to the discontinuity of the disc, not through it. Therefore:

Total pressure in front of disc = $$p+\frac{\rho_{v}2}{2}=p'+\frac{\rho\,(V+v)^{2}}{2}$$

Total pressure behind the disc = $$p'+\Delta p\frac{v^{2}}{2}=p+\frac{\rho\,(V+v)^{2}}{2}$$

where p = Atmospheric pressure
p' = Pressure just ahead of the actuator disc
ρ = Density
V = Free stream velocity
υ = Velocity increment at the actuator disc
υ1 = Velocity increment in the wake

The change in pressure across the disc must be equal to the change in total pressure.

The thrust acting on the disc is T = Aδp
where A = Disc area

$$T = A\rho\, V + \left[\frac{v_i}{2}\right] v^i \qquad (7.69)$$

Newton's Second Law also applies

$$T = ma = m\frac{dv}{dt}$$

The mass per unit time = $Q = A\,(V+v)$

and $dv = v_i$

$$T = m\frac{dV}{db} = A\rho\,(V+v)\,v_i \qquad (7.70)$$

Equating 7.69 and 7.70

$$A\rho\left[V+\frac{v_i}{2}\right]v_i = A\rho\,(V+v)\,v_i$$

or

$$v_1 = 2v \tag{7.71}$$

which states that the change in velocity at the exit of the control volume is exactly twice the increase in velocity at the actuator disc. Substituting Equation 7.70 yields

$$Thrust\ (T) = 2\rho A(V+v)v$$

The ideal or theoretical efficiency is defined as the ratio of the power output of the actuator disc to the power input.

$$\eta = \frac{TV}{T(V+v)} = \frac{V}{V+v} \tag{7.72}$$

Another useful expression will result if the wake velocity V_w is substituted for $V + v$.

$$\eta = \frac{2}{V} = \frac{V}{V+v} \tag{7.73}$$

$$1 + \frac{w}{V}$$

where

$$V_w = V + 2v$$

This ideal efficiency cannot be obtained in practice due to the initial assumption; however, the momentum theory can give a simple iteration on propeller operation. The actual design of a propeller or rotor requires a much more detailed analysis than the simplified momentum theory, one of the big shortcomings being that the actual shape of the propeller is not defined.

7.11.2 BLADE ELEMENT THEORY

The momentum theory is useful in determining theoretical maximum efficiencies but tells nothing about blade geometry and the effects of a finite number of blades with profile drag characteristics. Therefore, the blade element theory was developed; it gives more realistic results in the prediction of propeller and rotor operating characteristics. The blade element theory consists of determining the force acting on an element of the propeller blade, then integrating over the entire blade to obtain thrust and torque characteristics. The primary assumptions in the blade element theory are that uniform in flow exists, the flow is irrotational, and the blade is twisted such that each element of the blade is operating at its maximum L/D angle of attack.

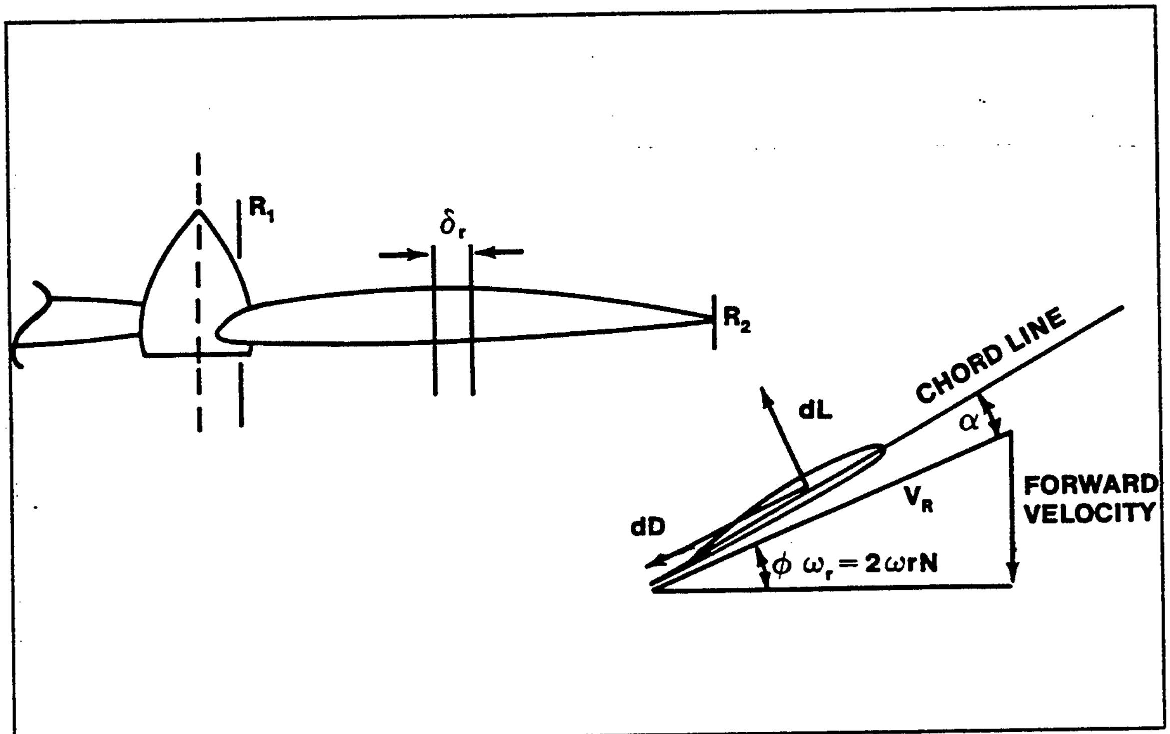


FIGURE 7.97. PROPELLER BLADE ELEMENT THEORY

The resultant velocity (V_r) that each blade element sees is the geometric sum of the aircraft forward velocity (V) and the local tangential velocity of the element (δr) where ω = angular velocity in rad/sec and r is the radius of the element from the center of rotation (Figure 7.97).

$$V_r=\sqrt{V^2+(\omega r)^2} \tag{7.74}$$

The blade element lift = $dl = 1/2\rho V^2{}_R (C\delta r)\, C_L$

where c is the average chord and C_L average lift coefficient and

the blade element drag = $dD = 1/2 V^2{}_R (C\delta r)\, C_D$

where C_D is the average drag coefficient.

Note that the blade element lift and drag are perpendicular and parallel to the relative local free stream respectively and to convert to a thrust and a torque requires summing the component of lift and drag perpendicular and in the plane of rotation of the propeller.

The thrust component = dT = dL cos ϕ - dD sin ϕ

Therefore by substituting and integrating, the total thrust (T) can be determined

Thrust (T) = $$N \int_{R1}^{R1} 1/2\rho V_R^2 (C_L \cos\phi - C_D \sin\phi)\, dr \qquad (7.75)$$

Similarly, the total torque required to drive the propeller at the angular velocity necessary to develop the torque (Q) is

Torque (Q) = $$N \int_{R1} 1/2\rho V_R^2 (C_L \cos\phi - C_D \sin\phi)\, dr \qquad (7.76)$$

The above blade element theory gives results with the accuracy compromised by the basic assumption of irrotational, uniform flow and optimized twist distribution.

7.11.3 VORTEX THEORY

The vortex theory using the technique of finite wings computes the induced flow velocities at each radial station rather than assuming uniform inflow. More exact theories have been developed by Goldstein and Theodorsen which account for tip loss, nonuniform blade twist distribution, and interference losses. The computer time required to perform the above vortex theory computations is extensive in comparison to the relatively small increase in predicted propeller performance accuracy.

Figure 7.98 gives a comparison between the calculated and the actual thrust distribution on a propeller blade. It can be seen that the difference primarily occurs on the inner one-third of the blade where it is difficult to achieve the necessary blade twist distribution to generate lift. Figure 7.87 shows that only the outer two-thirds of a propeller blade generates thrust and the inner one-third is there to provide attachment to the shaft. Designers try to minimize the drag of the inner one-third to prevent reverse flow, improve engine cooling, and reduce the torque required. Due to the fact that propellers often operate in proximity to engine nacelles and other parts of the airframe, etc., as well as problems with simple accurate prediction methods, light aircraft propeller manufacturers operate from past experience, NACA propeller charts, and black art.

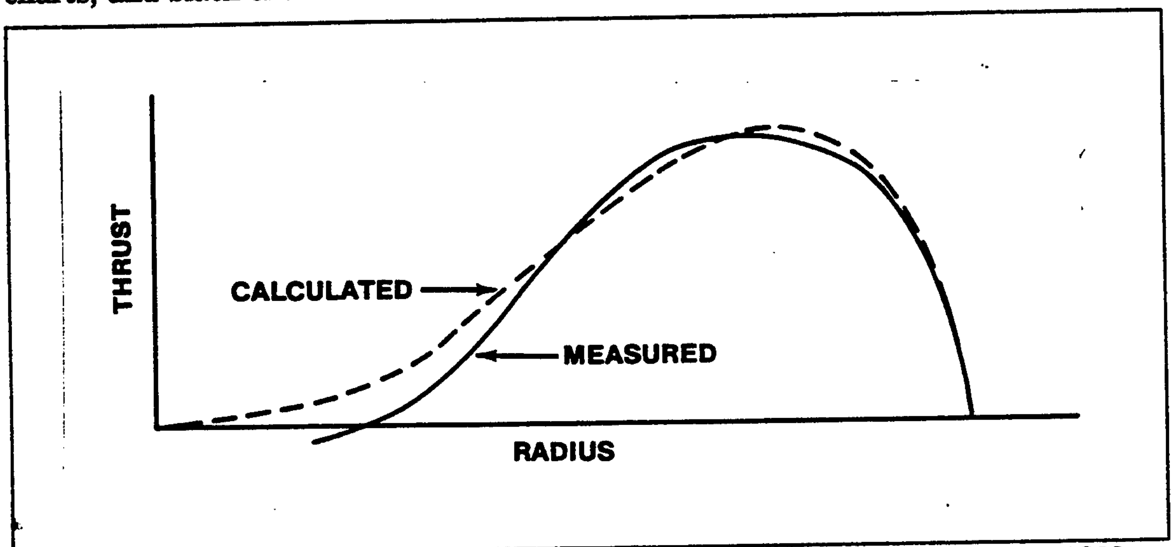


FIGURE 7.98. COMPARISON AND MEASURED THRUST DISTRIBUTION ON A PROPELLER BLADE

The above thrust distribution is good for one combination of free stream velocity and propeller RPM and assumes that each blade element is twisted to achieve its optimum L/D ratio. It is obvious that if either the RPM or free stream velocity changes, the propeller advance ratio, (J), where changes.

$$J \equiv \frac{V}{ND} = \frac{Free\ Stream\ Velocity}{(RPM)\ (Diameter)} = Advance\ Ratio, \tag{7.77}$$

Therefore, for a propeller with a fixed pitch, the thrust to torque ratio will be non-optimum, and the propeller efficiency will decrease. Similarly if a propeller is designed for a certain advance ratio and horsepower, increasing the horsepower driving the propeller will required either increasing the propeller RPM, which may cause tip compressibility problems, or increasing the blade pitch angle to absorb the horsepower. Increasing the blade pitch angle means that each blade element is not at the optimum L/D angle of attack and the propeller efficiency will decrease.

Another method of absorbing more horse power with a propeller is to increase the propeller diameter. Besides running into tip compressibility problems, the diameter is limited by ground and airframe clearance problems. Increasing the number of blades increases the problem of hub design, enlarges the hub diameter, and creates large aerodynamic interference problems between blades. Five blades seem to be the maximum operationally feasible number. Counter-rotating propellers are capable of absorbing large amounts of horsepower at reasonable diameters as there is more blade area. An additional benefit from the use of counter-rotating propellers is the lack of rotating slip stream and the lack of an out of balance torque on the aircraft. To have an equal amount of horsepower absorbed by both counter-rotating propellers, the rear propeller must be set at a slightly finer blade angle (approximately 1-1/2°) to allow for the slip stream of the front propeller. The disadvantages of counter-rotating propellers are weight, complexity and cost, and their non-availability for general aviation use.

7.11.4 PROPELLER PERFORMANCE

To obtain an understanding of the factors that influence propeller performance, the laws of similitude will be applied, i.e, since a propeller is a rotating airfoil, the laws of airfoils will be applied:

Airfoils: Force = (Dynamic Pressure) (Area) (Coefficient of Lift)

(At constant angle) $(1/2\rho_0 V_e^2)$ (S) (C_L)

Propellers: Force = (Dynamics Pressure) (Area) (Coefficient of Thrust)

$(1/2\rho\,(ND)_2)$ $(D)_2$ C_T

(thrust)

where D = Reference Length

D_2 = Reference Area

ND = Reference Velocity

Note that all of the constants have been dropped in the above referenced dimensions.

Therefore propeller thrust (T) = $\rho\,(ND)^2D^2C_T$

$$T = \rho\, N^2D^4C_T \tag{7.78}$$

The factor 1/2 is included in the thrust coefficient C_T and is not carried around as it is in airfoil aerodynamics. From the above equation it is clear that the thrust generated by the propeller is a very powerful function of propeller diameter.

Similarly, propeller torque is a thrust multiplied by an additional reference length D.

$$\text{Propeller torque } (Q) = \rho N^2D^4C_Q\ (D) = \rho\, N^2D^5C_Q \tag{7.79}$$

where C_Q is the propeller torque coefficient

Propeller power is the propeller torque multiplied by the angular velocity, which is a linear function of the RPM (N).

$$\text{Propeller power } (P) = \rho N^3D^5C_P \tag{7.80}$$

where C_p is the propeller power coefficient

The coefficients C_T, C_Q, and C_p are functions of angle, Reynolds number, Mach and propeller planform shape, similar to the airfoil coefficients C_L and $C_D = f\,(\omega, R_e, M, \text{Shape})$.

The resultant inflow velocity is a function of both the aircraft forward velocity and the local angular velocity of the propeller, Figure 7.99.

The propeller advance ratio (J) can be considered a representative angle for a propeller in a manner similar to the angle of attack (a) being used for airfoils.

The propeller efficiency $(\eta) = \dfrac{\text{Power Output}}{\text{Power Input}}$

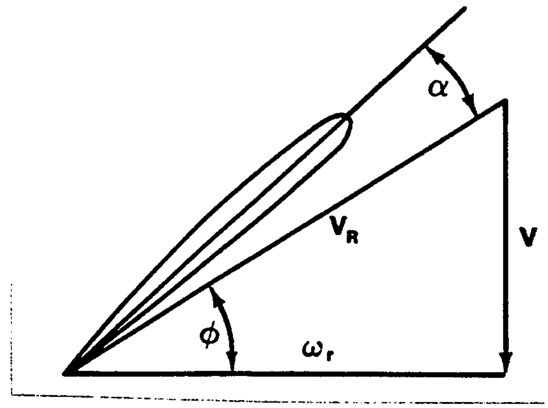


FIGURE 7.99. REPRESENTATIVE ANGLE FOR PROPELLERS

$$= \frac{(Thrust)\,(Velocity)}{(Torque)\,(AngularVelocity)}$$

$$= \frac{1}{2\pi}\;\frac{T}{Q}\;\frac{V}{N}$$

and since $T = \rho N^2 D^4 C_T$ and $Q = \rho N^2 D^5 C_Q$ then,

$$Propeller\ Efficiency = \frac{C_T}{C_Q}\;\frac{V}{ND} = \frac{C_T}{C_Q}\,J \qquad (7.81)$$

This indicates that the propeller efficiency is a function of thrust to torque ratio and propeller advance ratio which is similar to the efficiency of airfoils, i.e., the lift to drag ratio.

7.11.5 PROPELLER WIND TUNNEL TESTING

Wind tunnel testing of propellers is generally performed in special large test section wind tunnels that have reinforced test section enclosures and are equipped with large, variable speed and variable power electric motors capable of driving most propellers. The propeller thrust, torque, and power requirements are generally direct measurements from strain gauge instrumentation. The final data are plotted in the form shown in Figure 7.100.

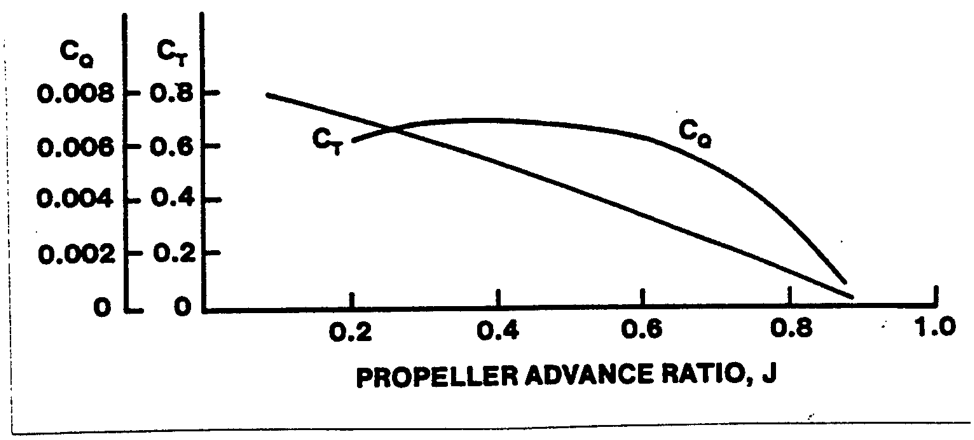


FIGURE 7.100. TYPICAL PROPELLER WIND TUNNEL RESULTS

The data presented in Figure 7.100 are not in the most useful form for aircraft designers, and often the propeller data are presented as shown in Figure 7.101. The propeller efficiency curve is used by the aircraft designers to ensure that the condition of the local free stream at the propeller or compressor in conjunction with the propulsive RPM are sufficient to achieve the peak efficiency. The three states of the propeller are also shown in Figure 7.101: the propulsive state in which the propeller absorbs power and delivers a thrust, the brake state in which the propeller absorbs power and delivers a drag, and the windmill state where the propeller absorbs power and delivers power from the free stream and produces a drag. Many propellers, especially those in turbo propeller installations, are placarded against extended operation in the windmill state because of propeller governor, bearing oil pressure, and bearing stress problems.

The data shown in Figure 7.101 apply to the propeller at one fixed blade setting, which is the blade angle of the propeller with respect to the plane of rotation. A fixed pitch propeller has the characteristic curves presented in Figure 7.101, and peak efficiency occurs only at one advance ratio. Therefore, most fixed pitch propellers are designed for use in optimum conditions such as cruise or climb, and non-optimum conditions are accepted in other flight regimes. The ground adjustable propeller used between 1925 and 1940 was the first attempt at using one propeller and having the capability of adjusting the pitch on the ground only for the most used flight conditions. The tow pitch position propeller controllable in flight is a simple tow position propeller in which one position is used for takeoff and climb and the other position for the cruise flight conditions. The controllable pitch propeller, similar to the early electrically operated propellers, had an infinite number of pitch settings between mechanical stops, that could be selected by the pilot for most efficient operation in every flight regime. To efficiently operate the controllable pitch propeller, the pilot must be given sufficient data by the airframe and propeller manufacturer regarding power settings, forward speed, and engine RPM.

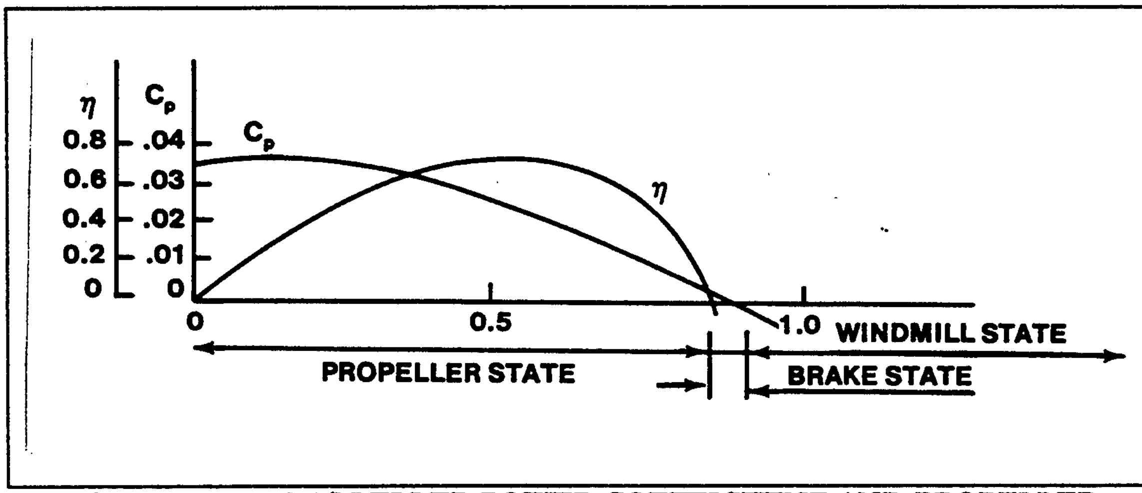


FIGURE 7.101. PROPELLER POWER COEFFICIIENT AND PROPELLER EFFICIENCY CURVES

The constant speed propeller operates under the principle that the pilot sets the desired RPM on the propeller governor and uses the throttle to command the power output of the engine. Once the power is sufficient to drive the propeller to the set RPM, additional power is absorbed by the propeller at the preset RPM by changing its pitch angle automatically. The use of a variable pitch or constant speed propeller modifies the propeller efficiency curves to those shown in Figure 7.102.

Obviously, from Figure 7.102, near peak efficiency is attainable on these variable pitch propellers throughout most operating conditions encountered in flight. A comparison of the characteristics of various types of propellers is shown in Figure 7.103 for a particular engine and airframe configuration. The variable pitch and constant speed propellers deliver the most thrust through out the complete speed range of the aircraft, and the selection of a fixed pitch propeller depends on the aircraft's mission such as cruise efficiency or glider towing.

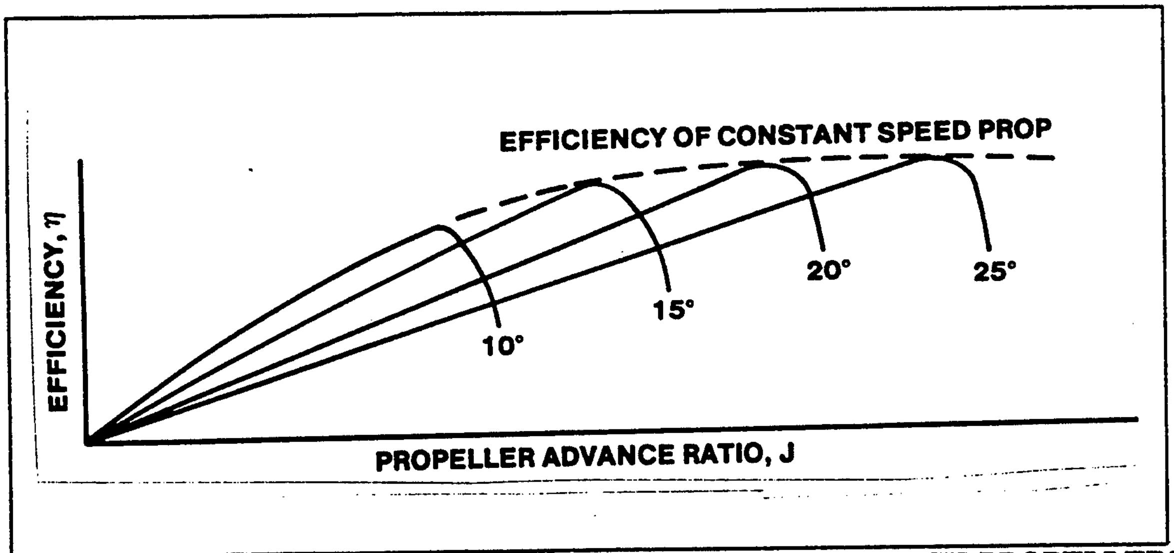


FIGURE 7.102. PROPELLER EFFICIENCY AND VARIABLE PITCH PROPELLERS

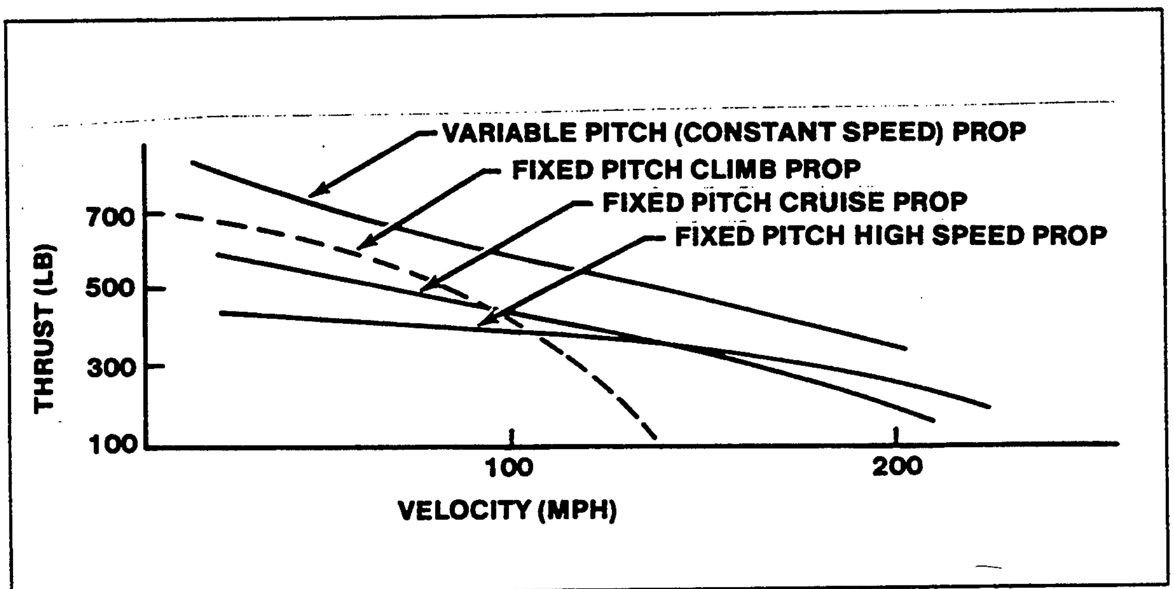


FIGURE 7.103. PROPELLER CHARACTERISTICS OF VARIOUS TYPES OF PROPELLERS

7.11.6 THE EFFECTS OF BLADE GEOMETRY ON PROPELLER CHARACTERISTICS

7.11.6.1 BLADE WIDTH

The propeller power coefficient increases with blade width; however, the propeller efficiency drops off with an increase in blade width. Both effects are due largely to increased slipstream velocity with the wider blades.

7.11.6.2 NUMBER OF BLADES

The number of blades affects the solidity ratio of the propeller disc where the solidity ratio,

$$\sigma = \frac{Bb}{\pi r} \tag{7.82}$$

B = Number of blades
b = Blade width
r = Length of blade

The factor Bb is very important in propeller work, and it has been found that a two-bladed propeller should have the same thrust, torque, and efficiency as a propeller having four blades, each half as wide, i.e., constant solidity. This is not completely true, due to an increase in interference drag, scale effects, and tip losses. However, an increase in the number of blades ensures a smoother operation.

7.11.6.3 BLADE THICKNESS

Blade thickness has little effect on propeller performance except in power requirements. Wooden propellers tend to be thicker due to structural requirements whereas aluminum blades can be thinner. Metal blades tend to be 4% to 7% more efficient than the equivalent wooden blades. Thinner metal blades would have a higher critical tip Mach.

7.11.6.4 BLADE SECTION

The airfoil sections of propellers are often made with a flat lower surface to facilitate measurement of the twist distribution. therefore, the shape of the profile is fairly well limited by the thickness ratio. With these factors fixed, any reasonably good airfoil section will give very nearly the same propeller characteristics as the best possible section. Promising results are being predicted for super critical airfoil sections in which relatively high lift to drag ratios can be achieved with a thick blade. The thick sections of the super critical blade look promising for hollow, blade type construction; this would be a big weight savings.

7.11.6.5 PLANFORM

The effect of planform on propeller performance is not great. The difference between a constant section planform and a tapered section is less than 1% efficiency. The tapered blades are advantageous form strength considerations.

Sweep back and rake of the blades have appreciable effects on the aerodynamic characteristics of a propeller; they affect the twist of the blades while operating and, therefore, the pitch. The blades on wooden propellers are often swept back in order to obtain smooth running qualities and to eliminate flutter.

7.11.6.6 BLADE TIPS

The geometry of the blade tips is very similar to the effect of wingtip geometry on wing characteristics and is difficult to measure in flight test because of the very small increments of thrust or drag involved. Due to the high centrifugal loads imposed on propeller tips, very little work has been performed on the effect of the geometry on propeller performance. Recently, the 'Q' tip propeller has been advanced by the propeller manufacturers as a method of improving propeller efficiency and decreasing propeller noise. Since the blade tip is bent backward (similar to a wingtip being bent downward) opposite to the current growth of winglets on aircraft, the performance improvements are questionable, and measurements have shown no perceptible noise decrease over regular propellers on the same aircraft.

7.11.7 SHROUDED PROPELLERS

Shrouded or ducted propellers have been used on tug boat propulsive systems since the early 1900's to increase the static thrust of the propeller at very low vehicle forward velocities. The use of the shrouded propeller for aircraft propulsion has been slow to catch on with the light aircraft manufacturers in the United States. However, a number of shrouded propeller military research vehicles were test flown in the mid 1960's with encouraging results, and a shrouded propeller is used on the German Fan Trainer.

A shrouded propeller, shown in Figure 7.104, consists of an airfoil sectional circular shroud around a propeller with the shroud cambered on the inside and the propeller located near the minimum diameter point in the shroud. The inherent circulation of the cambered shroud induces a velocity increment at the propeller plane. This increase in circulation produced by the thrust of the propeller promotes a low pressure region close to the leading edge of the shroud, which, when integrated circumferentially, results in a thrust of the perpendicular to the plane of the shroud and in the same direction as the thrust of the propeller. This thrust increment, together with the increase efficiency of the propeller blades due to the end plating effect of the shroud, gives a considerable increase in thrust over the open propeller at both static and slow forward velocities.

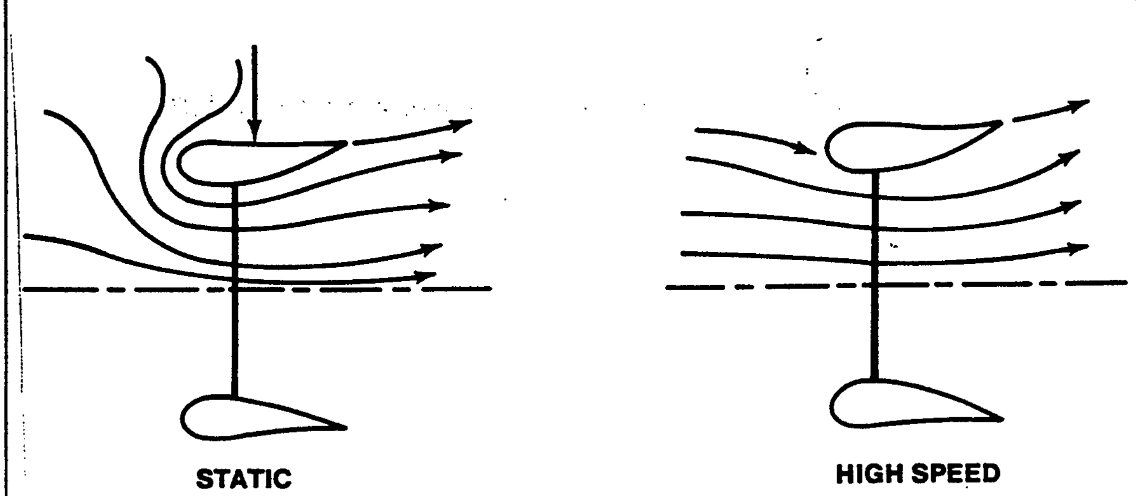


FIGURE 7.104. SKETCHES OF A SHROUDED PROPELLER

The thrust increase due to the presence of the shroud decreases with forward velocity. As the forward velocity increases, the front stagnation point on the shroud moves to the front of the shroud, thereby reducing the shroud circulation. Also, since the shroud has a parasitic drag which increases as the square of the forward velocity, it is obvious that here is some break even velocity where the thrust increment of the shroud equals the drag increment of the shroud (Figure 7.105).

The shrouded propeller, which is a combination of a propeller and a shroud, must be designed as a unit because the propeller and shroud are mutually interacting.

The general problem is to determine the flow field around a ring airfoil of known camber and thickness distribution inside of which exists a pressure discontinuity normal to the axis of symmetry and which is in the presence of a uniform free stream of arbitrary direction and magnitude. From the details of the flow field, the aerodynamic forces, moments, and overall efficiency can be calculated by integrating

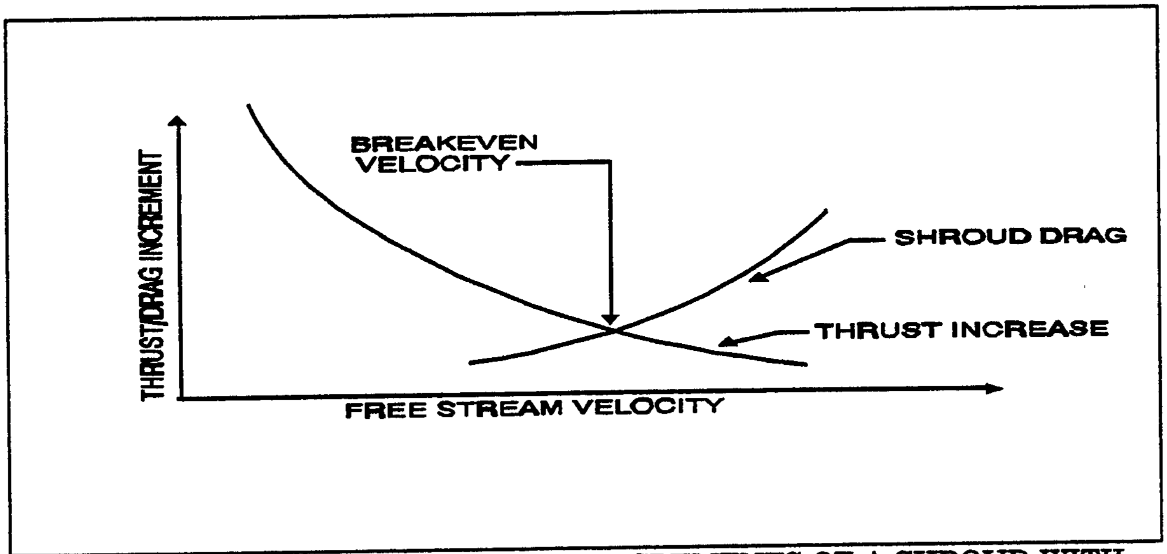


FIGURE 7.105. THRUST AND DRAG INCREMENTS OF A SHROUD WITH FORWARD VELOCITY

pressures over the various surfaces. Some of the methods of solving the above problem are briefly outlined as follows.

7.11.7.1 METHODS OF SINGULARITIES

Using this method, the shroud airfoil camber line is replaced by a distribution of vortices which produce the desired circulation equal to that of the shroud. Also, the effect of duct profile thickness and of centerbodies could be included by the use of additional distributed singularities. The mathematical expressions for determining the velocities induced throughout an inviscid, ideal, incompressible fluid due to an arbitrary distribution of potential vortices are well known and can be used in solving for the flow field. Holmbold, in advancing his method, assumed that the mathematical form of the shroud vorticity distribution had a number of unspecified coefficients for each term and that he had to satisfy the boundary condition at a number of points on the shroud equal to the number of unknown coefficients. Using this approach, Holmbold calculated the performance of a family of shrouds having assumed parabolic camber lines. These solutions of assumed vorticity distributions represent rather special cases and are not generally applicable unless a small chord-diameter ratio is used.

The mathematical difficulties encountered in the method of singularities make it very unpopular with designers; therefore, they use either the momentum methods or some modified method of their own.

7.11.7.2 MOMENTUM METHODS

The total thrust and power relationships of ducted propeller are quickly found by the application of Newton's Second Law to axial flow in front of and behind the duct. For example, the thrust can be expressed as a product of the mass flow per unit time through the duct and the change in velocity from infinity ahead to infinity behind the duct. This method is simple; however, certain assumptions must be made. The flow must be irrotational either by counterrotating propellers or by straightening vanes. Also, it generally is assumed that the jet area at infinity downstream equals the exit area of the duct. This overcomes the necessity of resorting to the method of singularities of linking the wake area and the wake velocity with the shroud design. However, assuming that the duct exit area is equal to the wake area implies that the velocity distribution in the wake is constant and also that the static pressure at the shroud exit is equal to ambient pressure. In other words, using this theory, the entire character of the wake is assumed.

Another wake area assumption suggested by Weining and developed by Treffitz is that the final wake area is related to the cross-sectional area and diffuser angle at the trailing edge of the duct, i.e.,

$$\frac{S_{Final}}{S_{Exit}} = \frac{1}{1 - 0.45\theta} \qquad (7.83)$$

where θ is the angle of inclination of the inside surface of the duct trailing edge with respect to the duct axis. The above equation, of course, is restricted to small values of θ unless some means of boundary layer control is applied to prevent flow separation.

7.11.7.3 OTHER METHODS

These methods are generally approximate methods that either place emphasis on the propeller, by using a blade element theory and modifying the blade element theory to take into account some of the influence of the shroud, or the emphasis is placed on the shroud, which usually consists of an approximation to the method of singularities for three-dimensional potential flow problems has been applied to the ducted propeller problem by Malavard. The boundary conditions are satisfied by the application of appropriate electrical potentials at the shroud and at the wake boundary, which is assumed to be of constant diameter.

In summary, it can be said that the mathematics are at our disposal for solving the problem of ducted propellers, provided that either the shroud camber line or the shroud vorticity distribution is specified. However, the mathematics are involved and complicated, and in practical applications where the inflow to the ducted propeller is not uniform, i.e., a ducted propeller at the rear of a fuselage, approximate methods using the momentum theory developed by Kuchmann and Weber are generally sufficiently accurate. Once the shroud shape is determined and the velocity distribution through the disc calculated, the required propeller twist can be determined using existing propeller design techniques.

7.11.8 SHROUDED FANS

The multi-bladed shrouded fan, driven by any type of engine including reciprocating, rotary or gas turbine engines, closely approximates the high bypass ratio turbofan in operational concept. High bypass ratio turbofans are much quieter than turbojets and therefore more environmentally acceptable. Shrouded propellers tend to be much quieter than open propellers, primarily due to the shroud end plating of the propeller blades and the fact that shrouded fans and propellers direct their noise output forward and rearward, thereby having low sideline acoustic signatures. The 1980 noise standards for general aviation aircraft in the United States and particularly in European countries have increased the possibility of using shrouded propellers on light aircraft for normal operational use. In 1978, Dowty of England installed and flight tested two shrouded fans on an Islander aircraft with an impressive reduction in propeller noise.

Design and trade-off studies of shrouded fans and propellers have been made by numerous companies, and a typical comparison is shown in Table 7.15, together with a drawing of the shrouded propeller/Q Fan. (Figure 7.106)

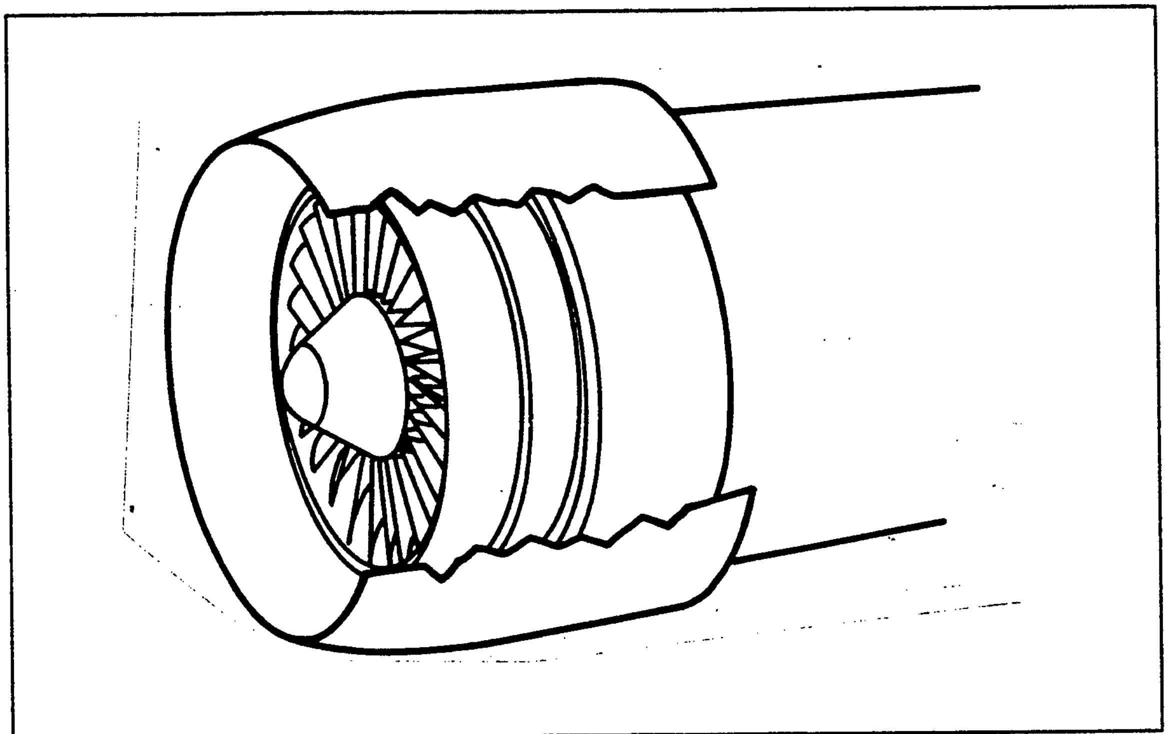


FIGURE 7.106. SHROUDED PROPELLER/Q FAN

TABLE 7.15
COMPARISON OF PROPELLER AND Q FAN CHARACTERISTICS

Propulsor Characteristics	Propeller	Q Fan
Diameter (Ft.)	6.5	3.0
Number of Blades	3	9
Tip Speed/RPM	915/2700	640/4060
Thrust at 66 Kts./M= .33	880/316	880/329
Weight (lbs.)	77	175
Noise Level (dB)	99.5	83.5

When both the propeller and the shrouded fan are designed for the same climb and cruise performance characteristics, the noise level of the fan is a fraction of the noise level of the propeller, but the weight penalty of the shroud is approximately 100 lbs. This weight penalty may be offset in the fan by elimination of a gear box so that high speed engines such as rotary engines can be used to drive the fan directly.

7.11.9 F.A.A. CERTIFICATION REQUIREMENTS

The F.A.A. airworthiness standard for propellers, F.A.R. Part 35, requires that a whirl test be performed to demonstrate that the propeller can withstand 200% of the maximum centrifugal force encountered in normal operations. Vibration tests must be performed on metal propellers, and an endurance test of 10 hours at maximum RPM and maximum certified diameter must be performed. The endurance test can also be accomplished with 50 hours of flight time consisting of 5 hours at 100% RPM and 45 hours at 90% RPM for fixed pitch propellers. Variable pitch propellers require 100 hours at maximum RPM and power on the engine for which they are to be certified. Functional tests of manually controllable, constant speed, feathering, and reversible systems must also be performed for a number of cycles without malfunctions.

The F.A.A. Part 23 requirements essentially deal with installation of the propeller(s) on the airframe, calling out ground clearance, water clearance, propeller tip, and structural clearances.

7.11.10 GROUND TESTING

Most of the ground testing of propellers is performed to demonstrate the structural, vibrational, and endurance requirements for certification on a particular engine. However, performance ground testing can often be very rewarding if the necessary facilities are available. Static thrust measurements as a function of power input will give an insight into the maximum power that can efficiently be absorbed by a particular installation as shown in Figure 7.107.

7.11.11 FLIGHT TESTING

Most propeller flight test hours are spent on endurance flying and ensuring that the engine-propeller combination is vibration free and the fatigue life of the blade is acceptable. The blades are strain gauged and the signals fed through slip rings on the shafts to recording equipment in the aircraft. Flight tests to determine propeller efficiency can be performed, but they require either a strain gauged propeller shaft that can measure shaft torque and propeller thrust or a special thrust and torque load cell mounted between the engine and the propeller.

$$\text{Propeller Efficiency} = \frac{(\text{Thrust})\ (\text{Forward Velocity})}{(\text{Torque})\ (\text{Angular Velocity})}$$

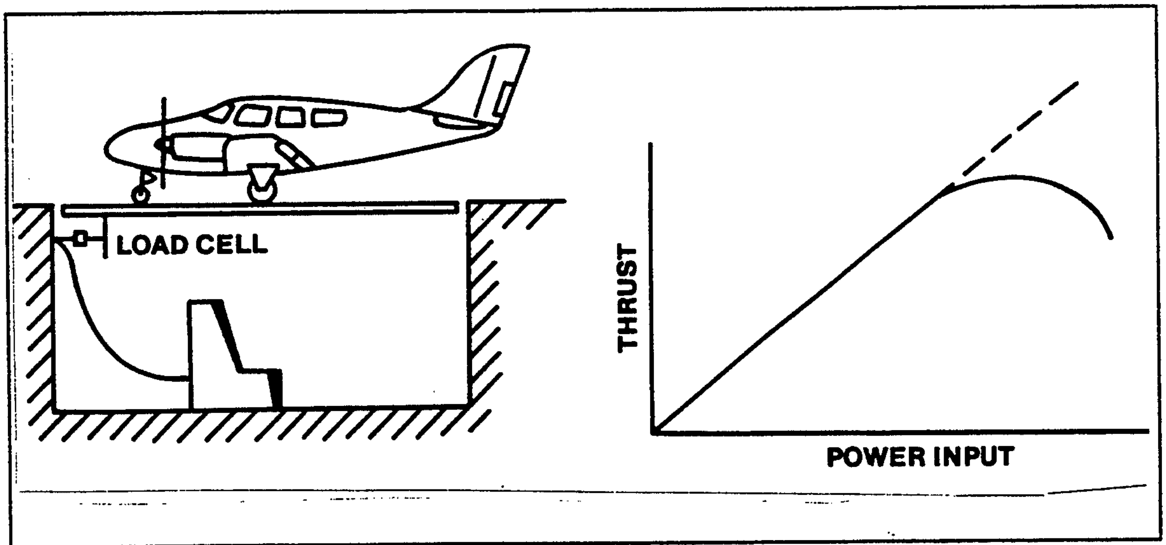


FIGURE 7.107. STATIC THRUST MEASUREMENTS OF ENGINE PROPELLER COMBINATIONS

The incremental drag method of measuring the propeller efficiency in flight is based on the assumption that the propeller efficiency does not change significantly with small incremental power changes where the propeller advance ratio is maintained constant. Instrumentation required is a small drag device such as a training cone connected to the aircraft by a load cell capable of directly measuring the drag of the trailing cone. The flight test technique is to obtain power-velocity data of the cruise configuration aircraft at a constant altitude and engine rpm and them return and attach the load cell and trailing cone and repeat the cruise tests at the same RPM and altitude. The data analysis is as follows:

Cruise Configuration Aircraft

$$\eta_p(B.H.P.) = \frac{(Drag)\ \ (Velocity)}{550} = \frac{D.V.}{550} \tag{7.84}$$

Aircraft with Come

$$\eta_p(B.H.P. + B.H.P.) = \frac{(Drag+\Delta Drag)\ (Velocity)}{550} \quad \frac{(D+\Delta D\ (V)}{550} \tag{7.85}$$

The unknown in the two equations above are propeller efficiency, (η) and aircraft drag (D). The solutions are:

$$\mathit{Propeller\ Efficiency}\ (\eta_p) = \frac{(\mathit{Velocity}(\Delta \mathit{Drag})}{550\ \ (\delta B.H.P.)} \tag{7.86}$$

$$\mathit{Aircraft\ Drag}\ (D) = \frac{(B.H.P.)\ \ (\Delta \mathit{Drag})}{(\Delta B.H.P.)} \tag{7.87}$$

The problems with the incremental drag method are that the technique consists of measuring small differences between large numbers. The drag device must be large enough to be able to measure differences in horsepower required, yet not large enough to violate the basic assumption that the propeller efficiency will not change with small increases in power absorbed. A comparison of theoretical predictions with propeller test results is shown in Figure 7.108.

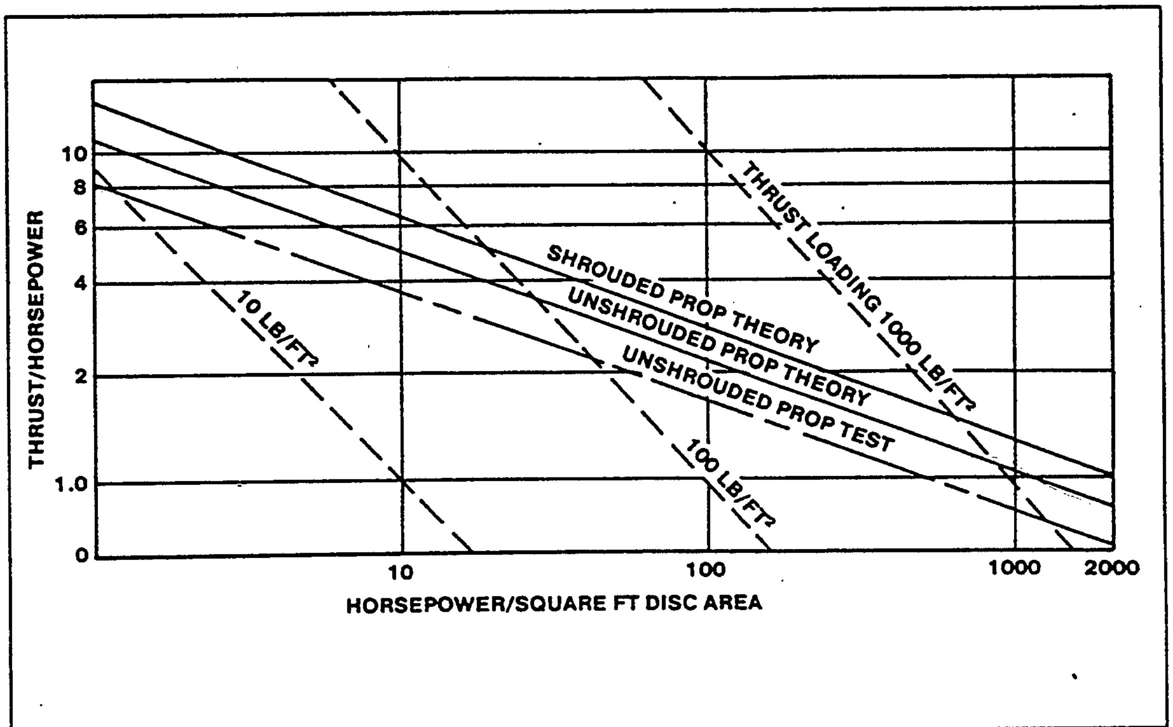


FIGURE 7.108. DUCTED PROPULSOR PERFORMANCE

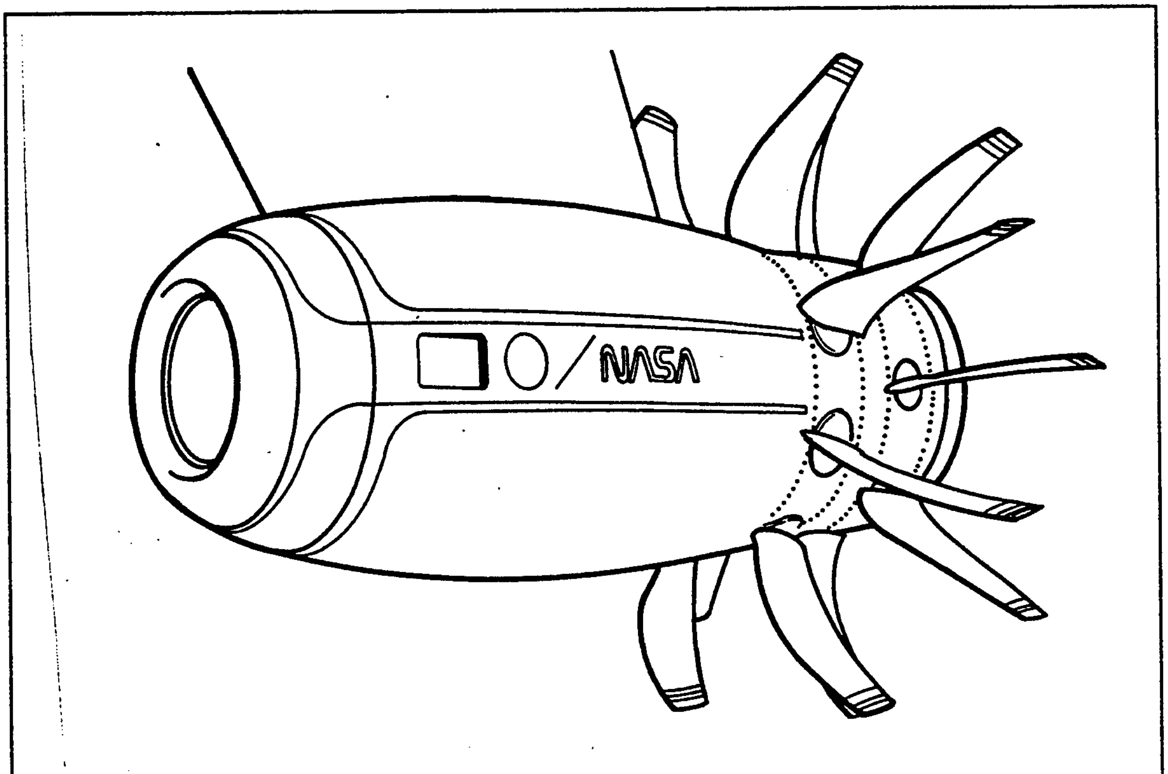


FIGURE 7.109. GENERAL ELECTRIC/NASA UNDUCTED FAN (UDF) DEMONSTRATOR ENGINE

7.11.12 ADVANCED DESIGN PROPELLERS

The previous paragraphs on propeller theory, FAA certification, and ground and flight testing all apply to so-called conventional propellers. During the last few years, due to the constant quest for higher and higher fuel efficiency, much research has been done concerning a whole new generation of advanced concept propeller designs. Most of this work has been conducted at the NASA Lewis facility in Cleveland, Ohio. Figure 7.109 is a typical example of what this new generation of propellers looks like.

These advanced propeller designs are intended for use with a turbine power plant rather than a reciprocating engine because of the much greater thrust to weight ratio of gas-turbines.

An advance design propeller would have very thin and highly swept blades to minimize both compressibility losses and propeller noise during high speed cruise. An area-ruled spinner and an integrated nacelle shape would also be used to minimize compressibility losses in the propeller-blade hub region. Propeller diameter would be kept to a minimum by using 8 to 10 blades with a high propeller power loading. These blades would be constructed using modern propeller blade fabrication techniques. The advanced propeller must be powered by a large, modern turboshaft engine and gearbox.

The basic reason for the attractiveness of this advanced turboprop concept is its potential for high propulsive efficiency in the Mach 0.7 to Mach 0.8 speed range. Older model turboprops had relatively thick, unswept propeller blades and experienced rapid increases in compressibility losses above Mach 0.6. Current high-bypass-ratio turbofans exhibit their highest propulsive efficiency (about 65%) at cruise speeds somewhere above Mach 0.8. The advanced turboprop concept is estimated to be about 20% more efficient than high-bypass-ratio turbofans at Mach 0.8. At lower cruise speeds, the efficiency advantage of the advanced turboprop is even larger. This high propulsive efficiency of the advanced turboprop makes it an attractive powerplant for many aircraft applications.

7.12 PROPULSION SYSTEM TESTING

Propulsion system flight testing remains the only true test of the marriage between an engine and an airframe. Although extensive test cell/wind tunnel testing of engines at sea level, altitude, and various Mach is accomplished prior to flight qualification, the almost infinite number of variables to which an actual flying installation is exposed can not yet be duplicated on the ground. As engines become more complex, even more variable are introduced such that ground test programs also become extremely complex. Duplication of high angles of attack, high positive or negative load factors, and high yaw angles cannot be adequately duplicated in a test cell. Efforts are continually underway to design programs and ground test facilities to reduce the risk and cost of developing an engine. The final proof of any propulsion design requires testing the installation in the environment for which it was designed, i.e., flight test.

7.12.1 PROPULSION FLIGHT TEST CATEGORIES

Propulsion flight test activity may fall into three categories: (1) flight test of a new engine, (2) flight test of components of an existing engine (Component Improvement Program, CIP), and (3) flight test of an existing engine with a new fuel. Typically, when a new engine is developed, the engine manufacturer performs the bulk of ground testing under the supervision of the Propulsion System Program Office (SPO). This ground testing starts at the factory and eventually progresses to "wind tunnel tests" simulating in-flight conditions. After the engine has been exposed to the aircraft flight envelope in the wind tunnel, it is ready for flight test.

The types of flight tests performed in a propulsion program fall into two basic categories: (1) tests to extract engine aerodynamic performance, and (2) test to evaluate systems operation. Since aircraft performance is directly linked to engine thrust, aircraft performance testing and engine performance testing are inseparable. Therefore, the same performance maneuvers are flown for engine and airframe performance data. These maneuvers consist of the classical performance tests such as cruise performance, level accelerations, turn performance, and climb and descent performance.

To evaluate the engine operation from the systems aspect, the following types of tests are typically made:

1. Installed ground tests
2. Throttle transients
3. Transfers to backup control
4. Climbs and descents
5. Airstarts
6. Engine handling/response (i.e., formation, air refueling)
7. Fuel/oil temperature exposure
8. Gas ingestion
9. Ice ingestion

7.12.2 INSTALLED GROUND TESTS

Installed ground tests are initially performed to ensure compatibility between engine and airframe systems operation. Prior to installation in the aircraft, all aircraft unique systems operations such as bleed air and power extraction are simulated. Installed ground tests serve as a final check of engine/aircraft interface prior to first flight. Additionally, instrumentation parameters are exercised as much as possible to ensure proper operation. All possible engine functions are exercised during these tests, such as starting throttle transients, transfer to backup control, etc. Typically, these tests are performed with the instrumentation system fully operational and data is telemetered to a ground station for real time monitoring of critical parameters.

7.12.2.1 GROUND STARTING

Ground starting of operational engines is usually a routine event. However, many problems can arise during initial test and evaluation of new engines of old engines in new airframes. Numerous starting mechanisms [air, gas generators, turboshaft starters (jet fuel starter), cross bleed systems, etc.] can all have their own peculiar problems. Compressor spin-up rate, fuel scheduling, starter cut-out speed, ignition systems, and ambient atmospheric conditions can all take their toll in hung starts, hot starts, no-lights, and starter failures. Unless specific problems arise, ground engine starting evaluations are confined to monitoring engine parameters during numerous starts in varying conditions.

7.12.3 THROTTLE TRANSIENTS

The ability of the engine to follow pilot-commanded change via the power lever (throttle) is essential to successful mission completion. Rapid engine acceleration and deceleration are required without any adverse effects such as turbine temperature overshoots or undershoots, compressor over or under-speed conditions, compressor stalls, or flameouts. The purpose of throttle transients is to (1) evaluate stall free operations throughout the flight envelope under worst case conditions, (2) evaluate engine acceleration/deceleration requirements and afterburner light requirements, and (3) evaluate back-up control operation.

The two basic types of transients used in flight test are BODIES and SNAPS. A BODIE is a transient that exposes the engine to operation nearest the compressor stall line. It is usually performed in dry power (nonafterburner) conditions, and involves a rapid throttle movement from military power to idle and a reversal back to military power prior to the engine RPM reading its steady state idle RPM.

A SNAP transient is a rapid throttle movement (approximately one second or less full travel time) from one end of the throttle range to another. These transients are performed under steady state flight conditions (Mach/altitude) at one g and under "maneuvering" conditions such as maximum angle of attack (a), maximum sideslip (b), and maximum a and b simultaneously. Typical throttle transients are shown in Figure 7.110.

Of utmost importance during engine testing is the use of the "buildup approach" when performing throttle transients, especially on single engine aircraft. Normally, transients should start at some agreed upon "heart of the envelope" flight condition and build up to critical parts of the envelope. A typical buildup approach is shown in

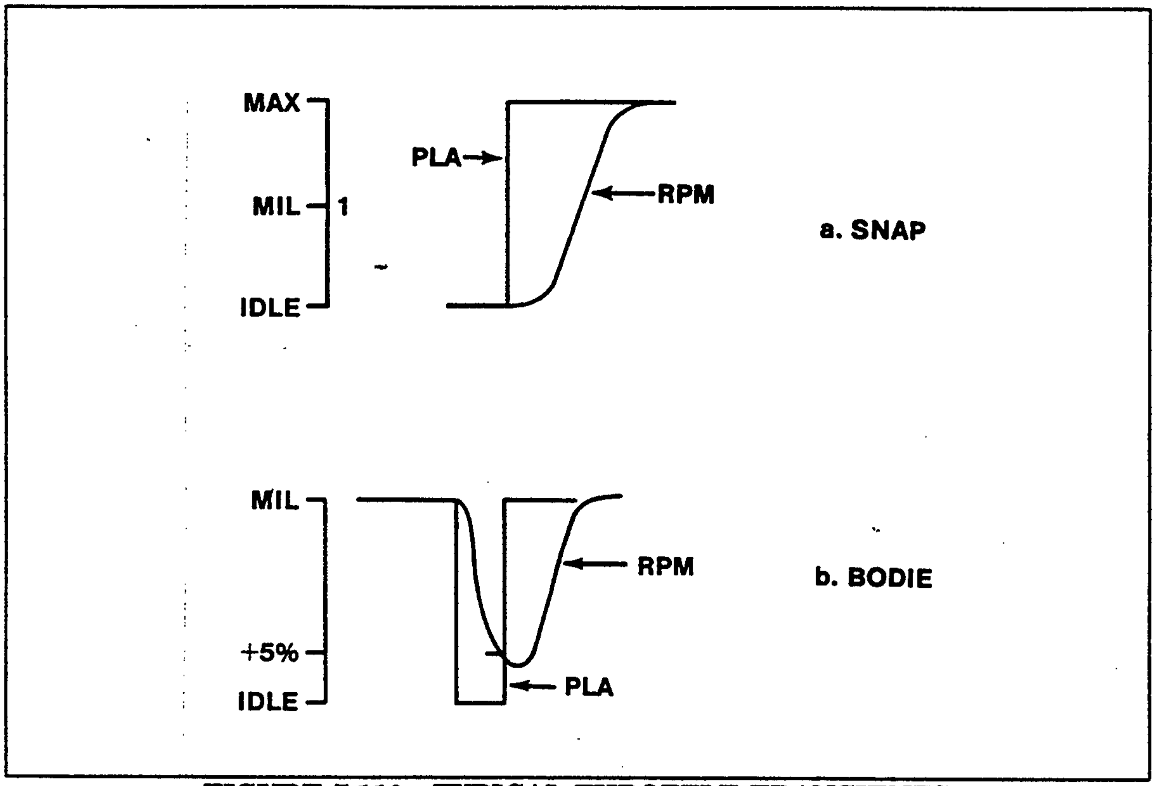


FIGURE 7.110. TYPICAL THROTTLE TRANSIENTS

Figure 7.111.

Several problems frequently occur during these tests, especially on early development models of engines. Off-idle stalls, that is, compressor stalls that occur as functions of compressor map stall/accelerating lines, fuel flow scheduling, compressor inlet guide valve scheduling, or inlet air flow distortion. Stalls of this nature are usually mild and self-clearing with throttle retardation.

Sub-idle conditions or hung stall conditions below normal idle speed can occur, usually at high altitude, upon rapid throttle retardation. Fuel control scheduling is usually the culprit in these cases. In one airplane, a "floor" was added to the fuel control schedule so the burner pressure could not decrease below that required to maintain compressor speed.

Afterburner ignition and flameout are almost always problem areas on new engines. Afterburner design is based as much on "cut and try" processes as it is on theoretical design analysis. Because of this, much wind-tunnel and flight testing is required. A new engine development program may encompass many hundreds of afterburner tests

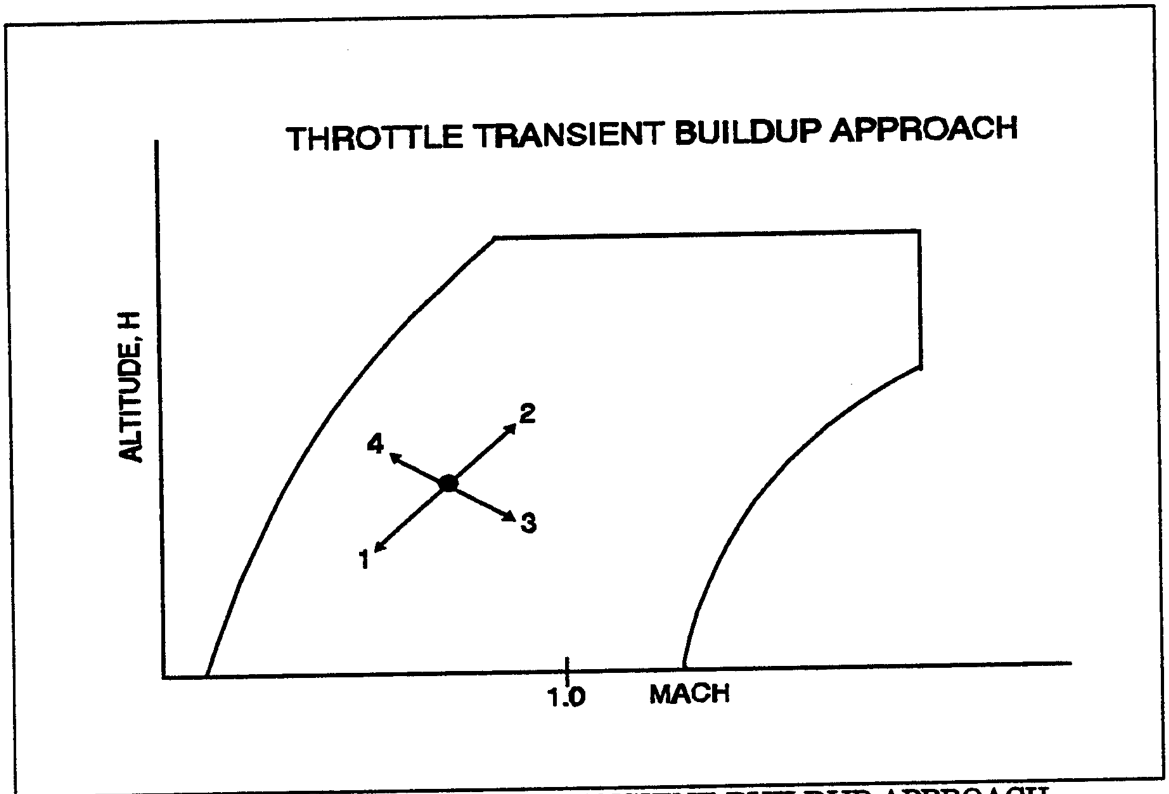


FIGURE 7.111. THROTTLE TRANSIENT BUILDUP APPROACH

using many combinations and ignition schemes. Variable exit-area nozzles are typically problem areas because of their hostile environment and complicated operating demands, especially when attached to a fan engine. Fan stalls due to duct pressure waves traveling forward in the fan duct caused by improper scheduling or operation of the nozzle are common occurrences in development engines.

Evaluation of transport type engine response is typically less demanding than it is on a fighter type aircraft, simply because of the difference in mission. Throttle transients on these types of engines are more closely related to electrical generator load and bleed air extractions. Because of the lack of an afterburner, testing is usually simpler. The advent of in-flight use of engine thrust reversers such as installed on STOL type transports may complicate this picture.

7.12.4 CLIMBS AND DESCENTS

Since ambient temperature and pressure change with altitude, these two parameters are incorporated in some manner in virtually all turbine engine fuel controls. In order to evaluate the adequacy of these controls with respect to accuracy, lag times, repeatability, drift, and the like, climbs and descents are performed at various rates of altitude change and at various engine power settings. The throttles are normally "locked" in a fixed position so that any changes detected are a function of the engine controls only and not pilot input. These maneuvers are usually combined with performance evaluations in the interest of conserving flight time.

7.12.5 AIRSTARTS

A flight test program to determine the airstart characteristics of an engine will be a critical portion of any engine test program. The purpose of in-flight testing of engine airstarts is two-fold. First, the airstart envelope must be determined and second, emergency procedures for airstarts must be developed.

In general, there are three types of airstarts: windmilling, spooldown, and assisted airstarts.

A windmilling airstart in one in which the ram air due to aircraft velocity is used to maintain compressor speed sufficient for an airstart. Most turbojet engines will windmill at 10 - 15% RPM at minimum flight speed which is enough to begin an airstart. Turbofans, however, seldom windmill satisfactorily and, if allowed to cage (engine rotation stops), may require over 400 kts to regain rotation.

Spooldown airstarts are done by initiating the airstart as the RPM decreases after shutdown. These can be time critical as beginning the airstart at too high an RPM frequently results in a hot start requiring shutdown and beginning at too low an RPM may not prevent the engine from caging.

An assisted airstart is one where either crossbleed from a good engine or a smaller motor is used to motor the compressor during main engine start. Engine starters are more common on modern turbofan engines and allow lower airspeed starts.

Airstart testing is normally done very early in an engine test program. Because of the critical nature of engine tests, especially in single-engine aircraft, the heart of the predicted airstart envelope should be verified within the first few flights. Optimizing pilot procedures, expanding the airstart envelope, and demonstrating back-up fuel control airstarts can be delayed to later phases of the test.

Typical airstart characteristics for a turbofan engine are shown in Figures 7.112 through 7.114.

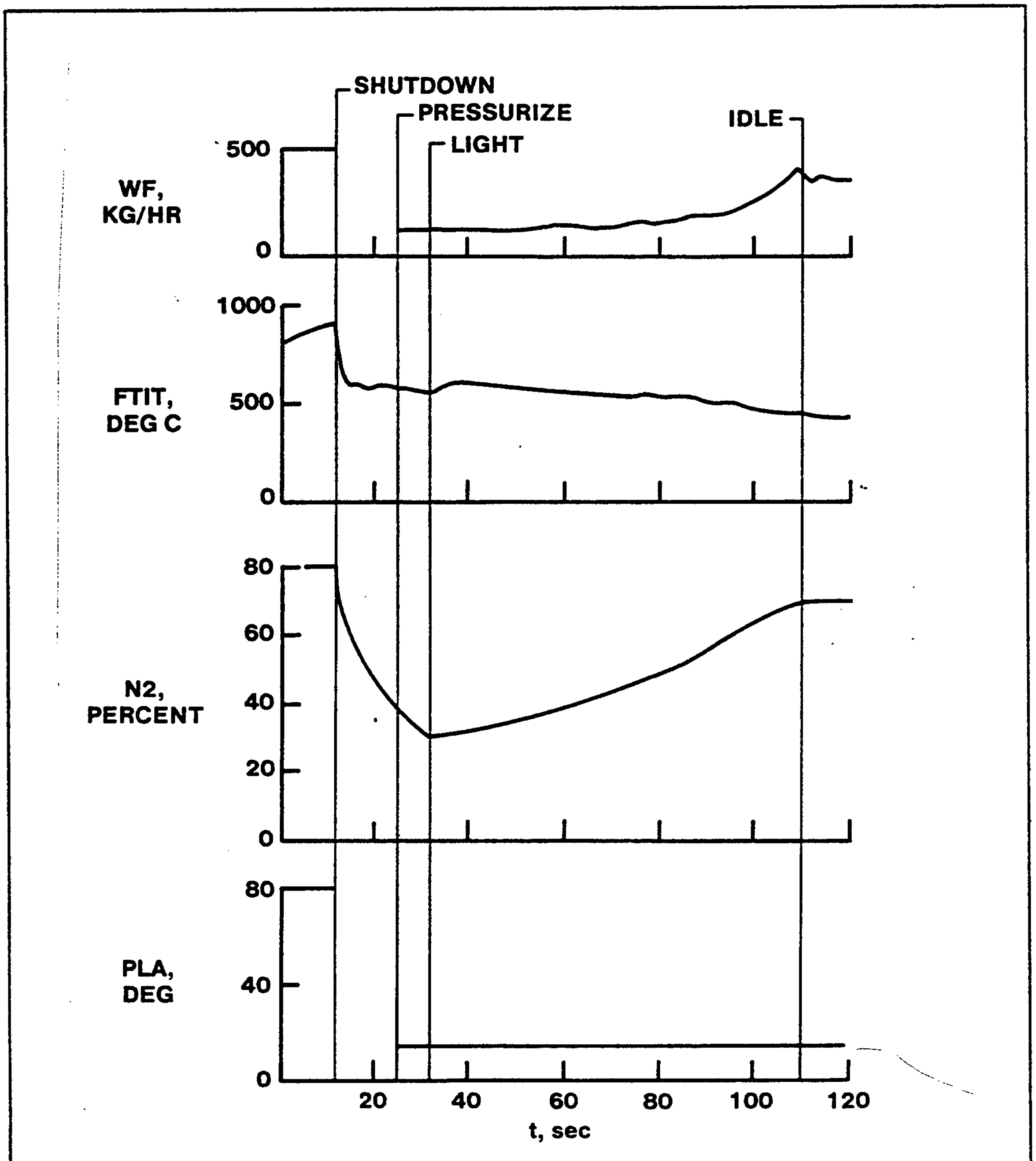


FIGURE 7.112. 40 PERCENT SPOOLDOWN AIRSTART, V = 200 KTS

7.12.6 ENGINE HANDLING AND RESPONSE

Engine handling and response will be evaluated to some degree on every flight. Pilots should report on the ease or difficulty encountered in doing all engine related tasks. However, certain tasks should be planned to investigate specifically how the thrust response affects pilot workload. Two of the best ways are to perform close formation tasks and air refueling. If there is an appreciable delay in thrust response these tasks

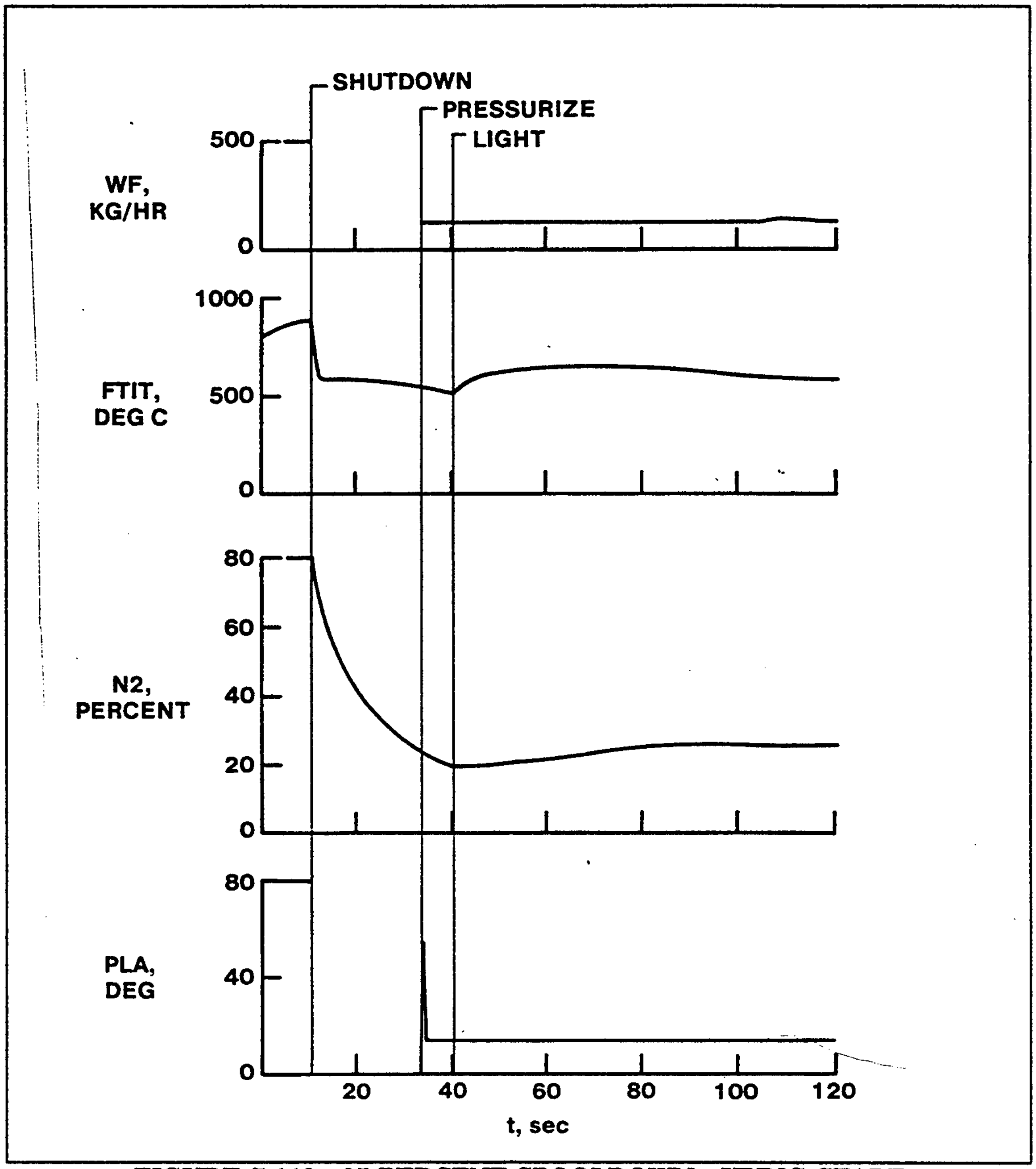


FIGURE 7.113. 25 PERCENT SPOOLDOWN - HUNG START

should show the severity of the problem.

7.12.7 GAS INGESTION

On fighter or ground support type aircraft, gas generated from firing of guns, cannons, and forward firing missiles can be ingested into engine inlets. This is a function of the physical relationship of the engine inlets to the source of the gas. Severe engine

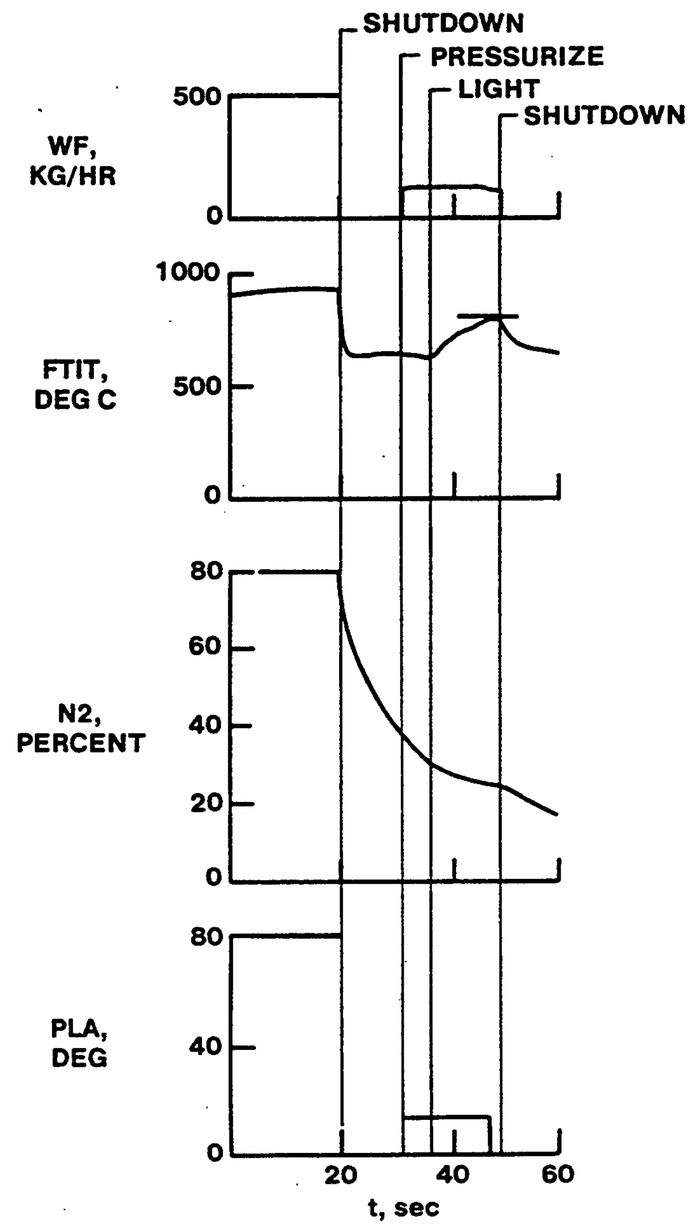


FIGURE 7.114. 40 PERCENT SPOOLDOWN - HOT START

compressor stalls and complete engine flameouts have been experienced due to ingestion of hot, noncombustible, gun firing guns and rockets at high altitude at various representative delivery conditions of airspeed, angle of attack sideslip, etc. Once the system is clear of any problems in this flight regime, air-to-air and air-to-ground firings are performed in the normal operational mode. Engine inlet rakes and engine instrumentation monitoring inlet temperature, compressor discharge pressure, and exhaust gas temperature, and chase photography are used to verify the presence or absence of foreign gas ingestion.

PROBLEMS

7.1 Calculate gross thrust,

$$M_{10} = M_0 \frac{\dot{w}_0}{g} = \frac{434.7\,lb/\text{sec}}{32.2\,ft/\text{sec}^2} = 7.76\ \frac{lb\ \text{sec}}{ft}$$

$$V_0 = 2805\,ft/\text{sec} \qquad P_{10} = P_A = 15\,lb/in^2$$

$$V_{10} = 3922\ ft/\text{sec} \qquad A_{10} = 6.0\ ft^2$$

$$\dot{w}_{FUEL} = 15\,\frac{lb}{\text{sec}}$$

7.2 Construct a typical h-s diagram for an ideal ramjet. What is required to allow a ramjet to function compared to a turbojet?

7.3 If an "ideal" compressor has a pressure ratio of 8.0 and an inlet temperature of 100°F, what would be the value to T_{T3}?

7.4 What T_{T3} should be used to get a maximum net work for an "ideal" engine designed to run in the isotherm region at $M_0 = 3.0$ if T_{T4} is limited to 3000°R?

7.5 At what compressor pressure ratio would the engine of Problem 7.4 operate?

7.6 If EPR (P_{T10}/P_{T2}) = 3.0, T_{S10} = 2000°R and WAT2C = 250 lb/sec, what is ideal net thrust at M_O = 1.0, sea level? Assume standard day conditions with 5007D ram recovery (refer to pgs. 68 - 84 of PW H/B).

7.7 An F-100 fan flows 200 lb/sec airflow at sea level standard day and $M_O = 0$ when the rotor speed is 9000 RPM. If we fly to 40,000 ft, $M_O = .8$, maintain the same corrected flow and rotor speed, at what airflow and rotor speed (w_a and N_1) will the engine be operating?

7.8 Sketch a subsonic inlet operating below Mach. What is the potential result?

7.9 Sketch a subsonic inlet at a Mach above design. What is the major penalty?

7.10 Explain how total pressure is increased in a compressor.

7.11 List the advantages of an axial flow compressor over a centrifugal compressor.

7.12 A fan is operating at S.L.S. ($M_O = 0$, Alt = 0). Its pressure ratio) ($P_{T2.5}/P_{T2}$) is 3.0, and its discharge total temperature is 272.0°F. What's is the fans?

7.13 Sketch the fan's operation (Prob 7.12) on an h-s diagram.

7.14 If the fan from Problem 7.12 had WATC2 = 250 lb/sec, how much horsepower does the rotor shaft have to deliver to the fan?

7.15 Discuss the 3 T's of combustion.

7.16 If the compressor from Problem 7.12 is powered by a turbine with $T_{T4} = 2000$°R and $P_{T4} = 30$ lb/in^2, what would T_{T5} be assuming turbine efficiency of 100%? Assume $C_p = 0.28$ and $\delta = 1.33$

7.17 List the three methods of air cooling turbine blades and vanes.

7.18 What is P_{T5} for Problem 7.16?

7.19 List 3 types of nozzles and explain good and bad points of each.

7.20 What nozzle would you expect on

a. a subsonic transport
b. a STOL fighter
c. a supersonic interceptor and why?

ANSWERS

7.1 $F_g = 52{,}947$ lb
$F_n = 37{,}868$ lb
TSFC = 1.43

7.3 $T_{T3} = 556°F$

7.4 $T_{T3} = 1082°F$

7.5 $P_{T3}/P_{T2} = 1.0$

7.6 $F_n = 37{,}868$ lb

7.7 $\dot{w}_a = 61.2\ lb/\text{sec}$

$N_1 = 8290$ RPM

7.12 90%

7.14 18,080 HP

7.16 $T_{T5} = 1817°R$

7.18 $P_{T5} = 20.4\ lb/in^2$

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