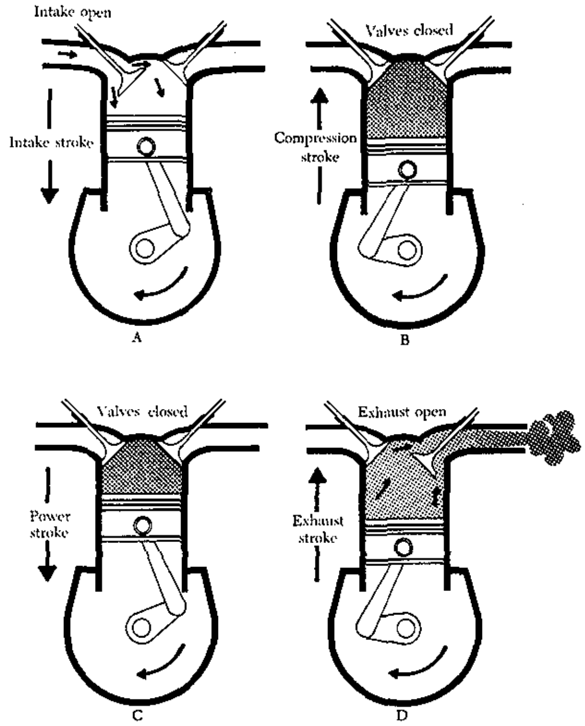


FIGURE 1–26. Four-stroke cycle.

In the following paragraphs, the timing of each event is specified in terms of degrees of crankshaft travel on the stroke during which the event occurs. It should be remembered that a certain amount of crankshaft travel is required to open a valve fully; therefore, the specified timing represents the start of opening rather than the full-open position of the valve.

Intake Stroke

During the intake stroke, the piston is pulled downward in the cylinder by the rotation of the crankshaft. This reduces the pressure in the cylinder and causes air under atmospheric pressure to flow through the carburetor, which meters the correct amount of fuel. The fuel/air mixture passes through the intake pipes and intake valves into the cylinders. The quantity or weight of the fuel/air charge depends upon the degree of throttle opening.

The intake valve is opened considerably before the piston reaches top dead center on the exhaust stroke, in order to induce a greater quantity of the fuel/air charge into the cylinder and thus increase the horsepower. The distance the valve may be opened before top dead center, however, is limited by several factors, such as the possibility that hot gases remaining in the cylinder from the previous cycle may flash back into the intake pipe and the induction system.

In all high-power aircraft engines, both the intake and the exhaust valves are off the valve seats at top dead center at the start of the intake stroke. As mentioned above, the intake valve opens before top dead center on the exhaust stroke (valve lead), and the closing of the exhaust valve is delayed considerably after the piston has passed top dead center and has started the intake stroke (valve lag). This timing is called valve overlap and is designed to aid in cooling the cylinder internally by circulating the cool incoming fuel/air mixture, to increase the amount of the fuel/air mixture induced into the cylinder, and to aid in scavenging the byproducts of combustion.

The intake valve is timed to close about 50° to 75° past bottom dead center on the compression stroke depending upon the specific engine, to allow the momentum of the incoming gases to charge the cylinder more completely. Because of the comparatively large volume of the cylinder above the piston when the piston is near bottom dead center, the slight upward travel of the piston during this time does not have a great effect on the incoming flow of gases. This late timing can be carried too far because the gases may be forced back through the intake valve and defeat the purpose of the late closing.

Compression Stroke

After the intake valve is closed, the continued upward travel of the piston compresses the fuel/air mixture to obtain the desired burning and expansion characteristics.

The charge is fired by means of an electric spark as the piston approaches top dead center. The time of ignition will vary from 20° to 35° before top dead center, depending upon the requirements of the specific engine, to ensure complete combustion of the charge by the time the piston is slightly past the top dead center position.

Many factors affect ignition timing, and the engine manufacturer has expended considerable time in research and testing to determine the best setting. All engines incorporate devices for adjusting the ignition timing, and it is most important that the ignition system be timed according to the engine manufacturer's recommendations.

Power Stroke

As the piston moves through the top dead center position at the end of the compression stroke and starts down on the power stroke, it is pushed downward by the rapid expansion of the burning gases within the cylinder head with a force that can be greater than 15 tons (30,000 p.s.i.) at maximum power output of the engine. The temperature of these burnning gases may be between 3,000° and 4,000°F.

As the piston is forced downward during the power stroke by the pressure of the burning gases exerted upon it, the downward movement of the connecting rod is changed to rotary movement by the crankshaft. Then the rotary movement is transmitted to the propeller shaft to drive the propeller. As the burning gases are expanded, the temperature drops to within safe limits before the exhaust gases flow out through the exhaust port.

The timing of the exhaust valve opening is determined by, among other considerations, the desirability of using as much of the expansive force as possible and of scavenging the cylinder as completely and rapidly as possible. The valve is opened considerably before bottom dead center on the power stroke (on some engines at 50° and 75° before B.D.C.) while there is still some pressure in the cylinder. This timing is used so that the pressure can force the gases out of the exhaust port as soon as possible. This process frees the cylinder of waste heat after the desired expansion has been obtained and avoids overheating the cylinder and the piston. Thorough scavenging is very important, because any exhaust products remaining in the cylinder will dilute the incoming fuel/air charge at the start of the next cycle.

Exhaust Stroke

As the piston travels through bottom dead center at the completion of the power stroke and starts upward on the exhaust stroke, it will begin to push the burned exhaust gases out the exhaust port. The speed of the exhaust gases leaving the cylinder creates a low pressure in the cylinder. This low or reduced pressure speeds the flow of the fresh fuel/air charge into the cylinder as the intake valve is beginning to open. The intake valve opening is timed to occur at 8° to 55° before top dead center on the exhaust stroke on various engines.

RECIPROCATING ENGINE POWER AND EFFICIENCIES

All aircraft engines are rated according to their ability to do work and produce power. This section presents an explanation of work and power and how they are calculated. Also discussed are the various efficiencies that govern the power output of a reciprocating engine.

Work

The physicist defines work as "Work is force times distance. Work done by a force acting on a body is equal to the magnitude of the force multiplied by the distance through which the force acts."

$$\text{Work (W)} = \text{Force (F)} \times \text{Distance (D)}.$$

Work is measured by several standards, the most common unit is called foot-pound. If a 1-pound mass is raised 1 foot, 1 ft.-lb. (foot-pound) of work has been performed. The greater the mass and the greater the distance, the greater the work.

Horsepower

The common unit of mechanical power is the hp. (horsepower). Late in the 18th century, James Watt, the inventor of the steam engine, found that an English workhorse could work at the rate of 550 ft.-lb. per second, or 33,000 ft.-lb. per minute, for a reasonable length of time. From his observations came the hp., which is the standard unit of power in the English system of measurement. To calculate the hp. rating of an engine, divide the power developed in ft.-lb. per minute by 33,000, or the power in ft.-lb. per second by 550.

$$\text{hp.} = \frac{\text{ft.-lb. per min.}}{33{,}000}, \text{ or}$$

$$= \frac{\text{ft.-lb. per sec.}}{550}.$$

As stated above, work is the product of force and ditsance, and power is work per unit of time. Consequently, if a 33,000-lb. weight is lifted through a vertical distance of 1 ft. in 1min., the power expended is 33,000 ft.-lb. per min., or exactly 1 hp.

Work is performed not only when a force is applied for lifting; force may be applied in any direction. If a 100-lb. weight is dragged along the ground, a force is still being applied to perform work, although the direction of the resulting motion is approximately horizontal. The amount of this force would depend upon the roughness of the ground.

If the weight were attached to a spring scale graduated in pounds, then dragged by pulling on the scale handle, the amount of force required could be measured. Assume that the force required is 90 lbs., and the 100-lb. weight is dragged 660 ft. in 2 min. The amount of work performed in the 2 min. will be 59,400 ft.-lb., or 29,700 ft.-lb. per min. Since 1 hp. is 33,000 ft.-lb. per min., the hp. expended in this case will be 29,700 divided by 33,000, or 0.9 hp.

Piston Displacement

When other factors remain equal, the greater the piston displacement the greater the maximum horsepower an engine will be capable of developing. When a piston moves from bottom dead center to top dead center, it displaces a specific volume. The volume displaced by the piston is known as piston displacement and is expressed in cubic inches for most American-made engines and cubic centimeters for others.

The piston displacement of one cylinder may be obtained by multiplying the area of the cross section of the cylinder by the total distance the piston moves in the cylinder in one stroke. For multi-cylinder engines this product is multiplied by the number of cylinders to get the total piston displacement of the engine.

Since the volume (V) of a geometric cylinder equals the area (A) of the base multiplied by the altitude (H), it is expressed mathematically as:

$$V = A \times H.$$

For our purposes, the area of the base is the area of the cross section of the cylinder or of the piston top.

Area Of A Circle

To find the area of a circle it is necessary to use a number called pi. This number represents the ratio of the circumference to the diameter of any circle. Pi cannot be found exactly because it is a never-ending decimal, but expressed to four decimal places it is 3.1416, which is accurate enough for most computations.

The area of a circle, as in a rectangle or triangle, must be expressed in square units. The distance that is one-half the diameter of a circle is known as the radius. The area of any circle is found by squaring the radius and multiplying by pi. (π). The formula is expressed thus:

$$A = \pi R^2.$$

Where A is the area of a circle; pi is the given constant; and r is the radius of the circle, which is equal to ½ the diameter or $R = \frac{D}{2}$

Example

Compute the piston displacement of the PWA 14 cylinder engine having a cylinder with a 5.5 inch diameter and a 5.5 inch stroke. Formulas required are:

$$R = \frac{D}{2}$$

$$A = \pi R^2$$

$$V = A \times H$$

$$\text{Total } V = V \times N \text{ (number of cylinders)}$$

Substitute values into these formulas and complete the calculation.

$$R = \frac{D}{2} \quad R = 5.5/2 = 2.75$$

$$A = \pi R^2 \quad A = 3.1416\ (2.75 \times 2.75)$$

$$A = 3.1416 \times 7.5625 = 23.7584 \text{ sq. inches}$$

$$V = A \times H \quad V = 23.7584 \times 5.5 \quad V = 130.6712$$

Total $V = V \times N$ Total $V = 130.6712 \times 14$ Total $V = 1829.3968$

Rounded off to the next whole number total piston displacement equals 1830 cu. in.

Another method of calculating the piston displacement uses the diameter of the piston instead of the radius in the formula for the area of the base.

$A = \frac{1}{4}\ (\pi)\ (D)^2$

Substituting $A = \frac{1}{4} \times 3.1416 \times 5.5 \times 5.5$

$A = .7854 \times 30.25 \quad A = 23.758$ square inches.

From this point on the calculations are identical to the preceding example.

Compression Ratio

All internal-combustion engines must compress the fuel/air mixture to receive a reasonable amount of work from each power stroke. The fuel/air charge in the cylinder can be compared to a coil spring, in that the more it is compressed the more work it is potentially capable of doing.

The compression ratio of an engine (see figure 1–27) is a comparison of the volume of space in a cylinder when the piston is at the bottom of the stroke to the volume of space when the piston is at the top of the stroke. This comparison is expressed as a ratio, hence the term "compresion ratio." Compression ratio is a controlling factor in the maximum horsepower developed by an engine, but it is limited by present-day fuel grades and the high engine speeds and manifold pressures required for takeoff. For example, if there are 140 cu. in. of space in the cylinder when the piston is at the bottom and there are 20 cu. in. of space when the piston is at the top of the stroke, the compression ratio would be 140 to 20. If this ratio is expressed in fraction form, it would be 140/20, or 7 to 1, usually represented as 7:1.

To grasp more thoroughly the limitation placed on compression ratios, manifold pressure and its effect on compression pressures should be understood. Manifold pressure is the average absolute pressure of the air or fuel/air charge in the intake manifold and is measured in units of inches of mercury (Hg). Manifold pressure is dependent on engine speed (throttle setting) and supercharging. The engine-driven internal supercharger (blower) and the external exhaust-driven supercharger (turbo) are actually centrifugal-type air compressors. The operation of these superchargers increases the weight of the charge entering the cylinder. When either one or both are used with the aircraft engine, the manifold pressure may be considerably higher than the pressure of the outside atmosphere. The advantage of this condition is that a greater amount of charge is forced into a given cylinder volume, and a greater power results.

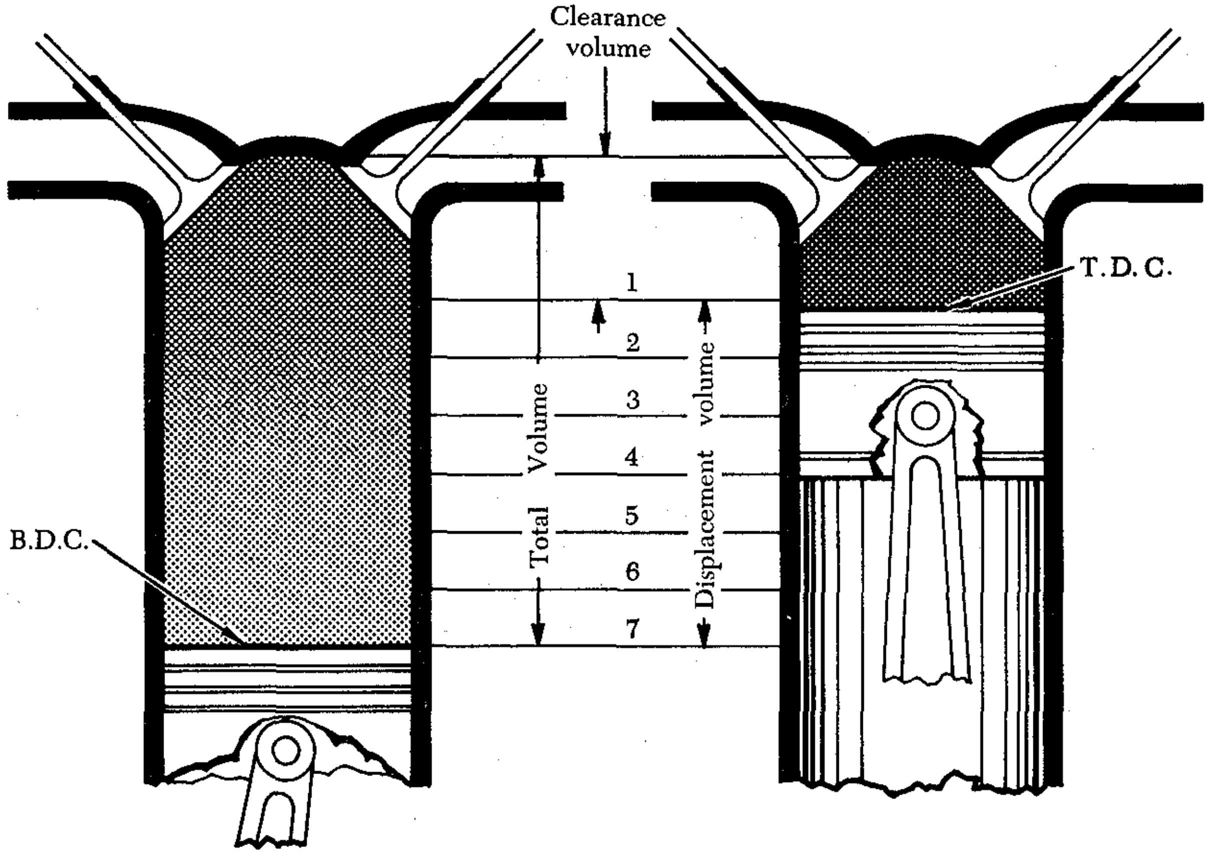


FIGURE 1–27. Compression ratio.

Compression ratio and manifold pressure determine the pressure in the cylinder in that portion of the operating cycle when both valves are closed. The pressure of the charge before compression is determined by manifold pressure, while the pressure at the height of compression (just prior to ignition) is determined by manifold pressure times the compression ratio. For example, if an engine were operating at a manifold pressure of 30″ Hg with a compression ratio of 7:1, the pressure at the instant before ignition would be approximately 210″ Hg. However, at a manifold pressure of 60″ Hg the pressure would be 420″ Hg.

Without going into great detail, it has been shown that the compression event magnifies the effect of varying the manifold pressure, and the magnitude of both affects the pressure of the fuel charge just before the instant of ignition. If the pressure at this time becomes too high, premature ignition or knock will occur and produce overheating.

One of the reasons for using engines with high compression ratios is to obtain long-range fuel economy, that is, to convert more heat energy into useful work than is done in engines of low compression ratio. Since more heat of the charge is converted into useful work, less heat is absorbed by the cylinder walls. This factor promotes cooler engine operation, which in turn increases the thermal efficiency.

Here, again, a compromise is needed between the demand for fuel economy and the demand for maximum horsepower without knocking. Some manufacturers of high-compression engines suppress knock at high manifold pressures by injecting an antiknock fluid into the fuel/air mixture. The fluid acts primarily as a coolant so that more power can be delivered by the engine for short periods, such as at takeoff and during emergencies, when power is critical. This high power should be used for short periods only.

Indicated Horsepower

The indicated horsepower produced by an engine is the horsepower calculated from the indicated mean effective pressure and the other factors which affect the power output of an engine. Indicated horsepower is the power developed in the combustion chambers without reference to friction losses within the engine.

This horsepower is calculated as a function of the actual cylinder pressure recorded during engine operation. To facilitate the indicated horsepower calculations, a mechanical indicating device, attached to the engine cylinder, scribes the actual pressure existing in the cylinder during the complete operating cycle. This pressure variation can be represented by the kind of graph shown in figure 1–28. Notice that the cylinder pressure rises on the compression stroke, reaches a peak after top center, then decreases as the piston moves down on the power stroke. Since the cylinder pressure varies during the operating cycle, an average pressure, line AB, is computed. This average pressure, if applied steadily during the time of the power stroke, would do the same amount of work as the varying pressure during the same period. This average pressure is known as indicated mean effective pressure and is included in the indicated horsepower calculation with other engine specifications. If the characteristics and the indicated mean effective pressure of an engine are known, it is possible to calculate the indicated horsepower rating.

The indicated horsepower for a four-stroke-cycle engine can be calculated from the following formula, in which the letter symbols in the numerator are arranged to spell the word "plank" to assist in memorizing the formula:

$$\text{Indicated horsepower} = \frac{\text{PLANK}}{33{,}000}.$$

Where:

P = Indicated mean effective pressure in p.s.i.

L = Length of the stroke in ft. or in fractions of a foot.

A = Area of the piston head or cross-sectional area of the cylinder, in sq. in.

N = Number of power strokes per minute; $\frac{\text{r.p.m.}}{2}$.

K = Number of cylinders.

In the formula above, the area of the piston times the indicated mean effective pressure gives the force acting on the piston in pounds. This force multiplied by the length of the stroke in feet gives the work performed in one power stroke, which, multiplied by the number of power strokes per minute, gives the number of ft.-lb. per minute of work produced by one cylinder. Multiplying this result by the number of cylinders in the engine gives the amount of work performed, in ft.-lb., by the engine. Since hp. is defined as work done at the rate of 33,000 ft.-lb. per min., the total number of ft.-lb. of work performed by the engine is divided by 33,000 to find the indicated horsepower.

Example

Given:

Indicated mean effective pressure (P)	= 165 lbs./sq. in.
Stroke (L)	= 6 in. or .5 ft.
Bore	= 5.5 in.
r.p.m.	= 3,000
No. of cylinders (K)	= 12

$$\text{Indicated hp.} = \frac{\text{PLANK}}{33{,}000 \text{ ft.-lbs./min.}}.$$

Find indicated hp.:

A is found by using the equation

$A = 1/4\pi D^2$

$A = 1/4 \times 3.1416 \times 5.5 \times 5.5$

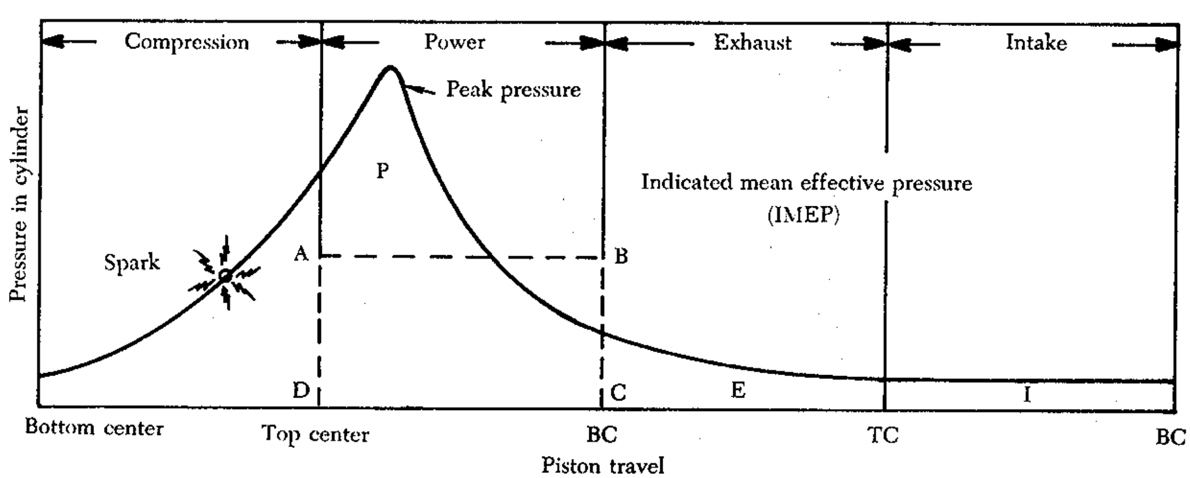


FIGURE 1–28. Cylinder pressure during power cycle.

$= 23.76$ sq. in.

N is found by multiplying the r.p.m. by 1/2:

$N = 1/2 \times 3{,}000$

$= 1{,}500$ r.p.m.

Now, substituting in the formula:

$$\text{Indicated hp.} = \frac{165 \times .5 \times 23.76 \times 1{,}500 \times 12}{33{,}000 \text{ ft.-lbs./min.}}$$

$$= 1069.20.$$

Brake Horsepower

The indicated horsepower calculation discussed in the preceding paragraph is the theoretical power of a frictionless engine. The total horsepower lost in overcoming friction must be subtracted from the indicated horsepower to arrive at the actual horsepower delivered to the propeller. The power delivered to the propeller for useful work is known as b.hp. (brake horsepower). The difference between indicated and brake horsepower is known as friction horsepower, which is the horsepower required to overcome mechanical losses such as the pumping action of the pistons and the friction of the pistons and the friction of all moving parts.

In practice, the measurement of an engine's b.hp. involves the measurement of a quantity known as torque, or twisting moment. Torque is the product of a force and the distance of the force from the axis about which it acts, or

Torque = Force x Distance (at right angles to the force).

Torque is a measure of load and is properly expressed in pound-inches (lb.-in.) or pound-feet (lb.-ft.) and should not be confused with work, which is expressed in inch-pounds (in.-lbs.) or foot-pounds (ft.-lbs.).

There are a number of devices for measuring torque, of which the Prony brake, dynamometer, and torquemeter are examples. Typical of these devices is the Prony brake (figure 1–29), which measures the usable power output of an engine on a test stand. It consists essentially of a hinged collar, or brake, which can be clamped to a drum splined to the propeller shaft. The collar and drum form a friction brake which can be adjusted by a wheel. An arm of a known length is rigidly attached to or is a part of the hinged collar and terminates at a point which bears on a set of scales. As the propeller shaft rotates, it tends to carry the hinged collar of the brake with it and is prevented from doing so only by the arm that bears on the scale. The scale reads the force necessary to arrest the motion of the arm. If the resulting force registered on the scale is multiplied by the length of the arm, the resulting product is the torque exerted by the rotating shaft. Example: If the scale registers 200 lbs. and the length of the arm is

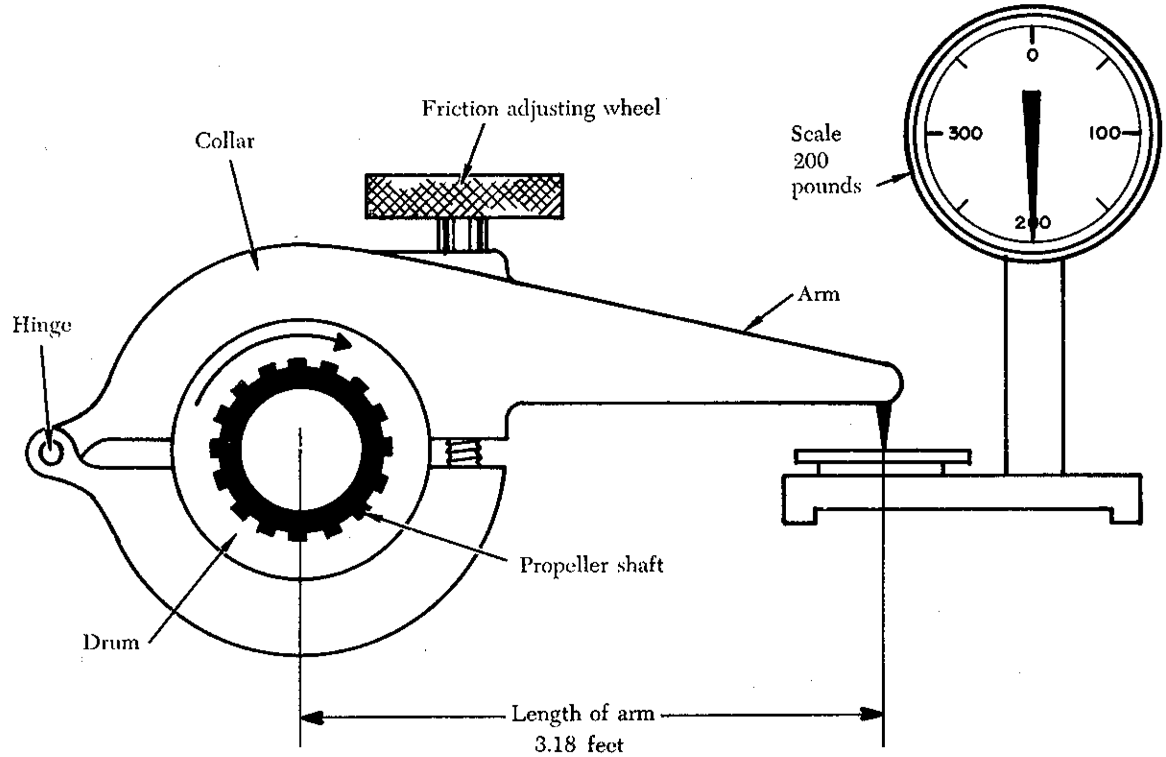


FIGURE 1–29. Typical Prony brake.

3.18 ft., the torque exerted by the shaft is:

$$200 \text{ lb.} \times 3.18 \text{ ft.} = 636 \text{ lb.-ft.}$$

Once the torque is known, the work done per revolution of the propeller shaft can be computed without difficulty by the equation:

$$\text{Work per revolution} = 2\pi \times \text{torque}.$$

If work per revolution is multiplied by the r.p.m., the result is work per minute, or power. If the work is expressed in ft.-lbs. per min., this quantity is divided by 33,000; the result is the brake horsepower of the shaft. In other words:

$$\text{Power} = \text{Work per revolution} \times \text{r.p.m.}$$

$$\text{and b.hp.} = \frac{\text{Work per revolution} \times \text{r.p.m.}}{33{,}000},$$

$$\text{or b.hp.} = \frac{2\pi \times \text{force on the scales (lbs.)} \times \text{length of arm (ft.)} \times \text{r.p.m.}}{33{,}000}.$$

Example

Given:

Force on scales = 200 lbs.
Length of arm = 3.18 ft.
r.p.m. = 3,000
π = 3.1416.

Find b.hp.:

Substituting in equation—

$$\text{b.hp.} = \frac{6.2832 \times 200 \times 3.18 \times 3{,}000}{33{,}000}$$
$$= 363.2$$
$$= 363.$$

As long as the friction between the brake collar and propeller shaft drum is great enough to impose an appreciable load on the engine, but is not great enough to stop the engine, it is not necessary to know the amount of friction between the collar and drum to compute the b.hp. If there were no load imposed, there would be no torque to measure, and the engine would "run away." If the imposed load is so great that the engine stalls, there may be considerable torque to measure, but there will be no r.p.m. In either case it is impossible to measure the b.hp. of the engine. However, if a reasonable amount of friction exists between the brake drum and the collar and the load is then increased, the tendency of the propeller shaft to carry the collar and arm about with it becomes greater, thus imposing a greater force upon the scales. As long as the torque increase is proportional to the r.p.m. decrease, the horsepower delivered at the shaft remains unchanged. This can be seen from the equation in which 2π and 33,000 are constants and torque and r.p.m. are variables. If the change in r.p.m. is inversely proportional to the change in torque, their product will remain unchanged. Therefore, b.hp. remains unchanged. This is important. It shows that horsepower is the function of both torque and r.p.m., and can be changed by changing either torque or r.p.m., or both.

Friction Horsepower

Friction horsepower is the indicated horsepower minus brake horsepower. It is the horsepower used by an engine in overcoming the friction of moving parts, drawing in fuel, expelling exhaust, driving oil and fuel pumps, and the like. On modern aircraft engines, this power loss through friction may be as high as 10 to 15% of the indicated horsepower.

Friction and Brake Mean Effective Pressures

The IMEP (indicated mean effective pressure), discussed previously, is the average pressure produced in the combustion chamber during the operating cycle and is an expression of the theoretical, frictionless power known as indicated horsepower. In addition to completely disregarding power lost to friction, indicated horsepower gives no indication as to how much actual power is delivered to the propeller shaft for doing useful work. However, it is related to actual pressures which occur in the cylinder and can be used as a measure of these pressures.

To compute the friction loss and net power output, the indicated horsepower of a cylinder may be thought of as two separate powers, each producing a different effect. The first power overcomes internal friction, and the horsepower thus consumed is known as friction horsepower. The second power, known as brake horsepower, produces useful work at the propeller. Logically, therefore, that portion of IMEP that produces brake horsepower is called BMEP (brake mean effective pressure). The remaining pressure used to overcome internal friction is called FMEP (friction mean effective pressure). This is illustrated in figure 1–30. IMEP is a useful expression of total cylinder power output, but is not a real physical quantity; likewise, FMEP and BMEP are theoretical but useful expressions of friction losses and net power output.

Although BMEP and FMEP have no real existence in the cylinder, they provide a convenient means of representing pressure limits, or rating engine performance throughout its entire operating range. This is true since there is a relationship between IMEP, BMEP, and FMEP.

One of the basic limitations placed on engine operation is the pressure developed in the cylinder during combustion. In the discussion of compression ratios and indicated mean effective pressure, it was found that, within limits, the increased pressure resulted in increased power. It was also noted that if the cylinder pressure was not controlled

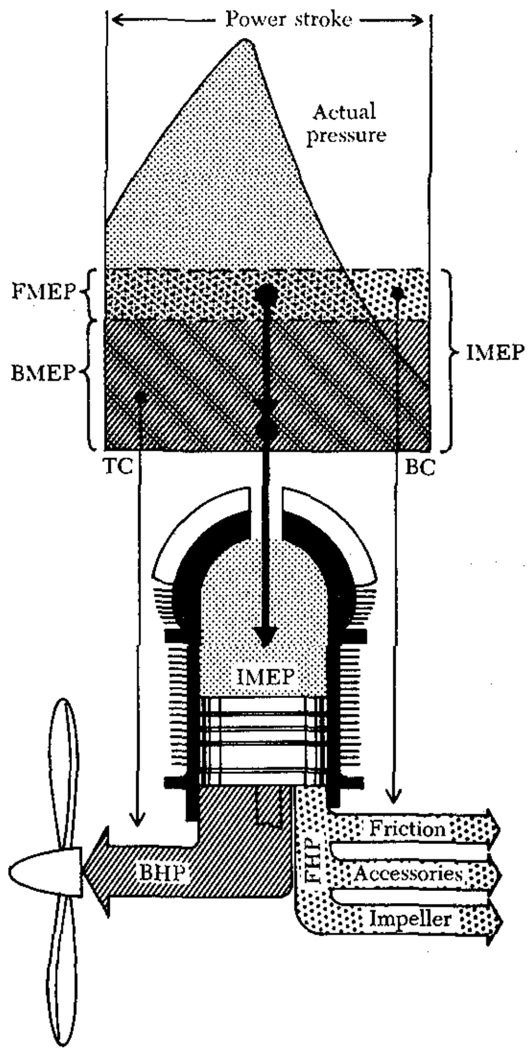


FIGURE 1–30. Powers and pressures.

within close limits, it would impose dangerous internal loads that might result in engine failure. It is therefore important to have a means of determining these cylinder pressures as a protective measure and for efficient application of power.

If the b.hp. is known, the BMEP can be computed by means of the following equation:

$$\text{BMEP} = \frac{\text{b.hp. x } 33{,}000}{\text{LANK}}.$$

Example

Given:

b.hp. = 1,000
Stroke = 6 in.
Bore = 5.5 in.
r.p.m. = 3,000
No. of cyls. = 12.

Find BMEP:

Find length of stroke (in ft.):

$L = 0.5.$

Find area of cylinder bore:

$$A = 1/4\pi D^2$$
$$= 0.7854 \text{ x } 5.5 \text{ x } 5.5$$
$$= 23.76 \text{ sq. in.}$$

Find No. of power strokes per min.:

$$N = 1/2 \text{ x r.p.m.}$$
$$= 1/2 \text{ x } 3{,}000$$
$$= 1{,}500.$$

Then substituting in the equation:

$$\text{BMEP} = \frac{1{,}000 \text{ x } 33{,}000}{.5 \text{ x } 23.76 \text{ x } 1{,}500 \text{ x } 12}$$
$$= 154.32 \text{ lbs. per sq. in.}$$

Thrust Horsepower

Thrust horsepower can be considered as the result of the engine and the propeller working together. If a propeller could be designed to be 100% efficient, the thrust- and the brake-horsepower would be the same. However, the efficiency of the propeller varies with the engine speed, attitude, altitude, temperature, and airspeed, thus the ratio of the thrust horsepower and the brake horsepower delivered to the propeller shaft will never be equal. For example, if an engine develops 1,000 b.hp., and it is used with a propeller having 85 percent efficiency, the thrust horsepower of that engine-propeller combination is 85 percent of 1,000 or 850 thrust hp. Of the four types of horsepower discussed, it is the thrust horsepower that determines the performance of the engine-propeller combination.

EFFICIENCIES

Thermal Efficiency

Any study of engines and power involves consideration of heat as the source of power. The heat produced by the burning of gasoline in the cylinders causes a rapid expansion of the gases in the cylinder, and this, in turn, moves the pistons and creates mechanical energy.

It has long been known that mechanical work can be converted into heat and that a given amount of heat contains the energy equivalent of a certain amount of mechanical work. Heat and work are theoretically interchangeable and bear a fixed relation to each other. Heat can therefore be measured in work units (for example, ft.-lbs.) as well as in heat units. The B.t.u. (British thermal unit) of heat is the quantity of heat required to raise the temperature of 1 lb. of water 1° F. It is equivalent to 778 ft.-lbs. of mechanical work. A pound of petroleum fuel, when burned with enough air to consume it completely, gives up about 20,000 B.t.u., the equivalent of 15,560,000 ft.-lbs. of mechanical work. These quantities express the heat energy of the fuel in heat and work units, respectively.

The ratio of useful work done by an engine to the heat energy of the fuel it uses, expressed in work or heat units, is called the thermal efficiency of the engine. If two similar engines use equal amounts of fuel, obviously the engine which converts into work the greater part of the energy in the fuel (higher thermal efficiency) will deliver the greater amount of power. Furthermore, the engine which has the higher thermal efficiency will have less waste heat to dispose of to the valves, cylinders, pistons, and cooling system of the engine. A high thermal efficiency also means a low specific fuel consumption and, therefore, less fuel for a flight of a given distance at a given power. Thus, the practical importance of a high thermal efficiency is threefold, and it constitutes one of the most desirable features in the performance of an aircraft engine.

Of the total heat produced, 25 to 30% is utilized for power output; 15 to 20% is lost in cooling (heat radiated from cylinder head fins); 5 to 10% is lost in overcoming friction of moving parts; and 40 to 45% is lost through the exhaust. Anything which increases the heat content that goes into mechanical work on the piston, which reduces the friction and pumping losses, or which reduces the quantity of unburned fuel or the heat lost to the engine parts, increases the thermal efficiency.

The portion of the total heat of combustion which is turned into mechanical work depends to a great extent upon the compression ratio. Compression ratio is the ratio of the piston displacement plus combustion chamber space to the combustion chamber space. Other things being equal, the higher the compression ratio, the larger is the proportion of the heat energy of combustion turned into useful work at the crankshaft. On the other hand, increasing the compression ratio increases the cylinder head temperature. This is a limiting factor, for the extremely high temperature created by high compression ratios causes the material in the cylinder to deterioriate rapidly and the fuel to detonate.

The thermal efficiency of an engine may be based on either b.hp. or i.hp. and is represented by the formula:

$$\text{Indicated thermal efficiency} = \frac{\text{i.hp. x } 33{,}000}{\text{weight of fuel burned/min. x heat value x } 778}.$$

The formula for brake thermal efficiency is the same as shown above, except the value for b.hp. is inserted instead of the value for i.hp.

Example

An engine delivers 85 b.hp. for a period of 1 hr. and during that time consumes 50 lbs. of fuel. Assuming the fuel has a heat content of 18,800 B.t.u. per lb., find the thermal efficiency of the engine:

$$\frac{85 \text{ x } 33{,}000}{.833 \text{ x } 18{,}800 \text{ x } 778} = \frac{2{,}805{,}000}{12{,}184{,}569}.$$

Brake thermal efficiency = 0.23 or 23%.

Reciprocating engines are only about 34% thermally efficient; that is, they transform only about 34% of the total heat produced by the burning fuel into mechanical energy. The remainder of the heat is lost through the exhaust gases, the cooling system, and the friction within the engine. Thermal distribution in a reciprocating engine is illustrated in figure 1–31.

Mechanical Efficiency

Mechanical efficiency is the ratio that shows how much of the power developed by the expanding gases in the cylinder is actually delivered to the output shaft. It is a comparison between the b.hp. and the i.hp. It can be expressed by the formula:

$$\text{Mechanical efficiency} = \frac{\text{b.hp.}}{\text{i.hp.}}.$$

Brake horsepower is the useful power delivered to the propeller shaft. Indicated horsepower is the total hp. developed in the cylinders. The difference between the two is f.hp. (friction horsepower), the power lost in overcoming friction.

The factor that has the greatest effect on mechanical efficiency is the friction within the engine itself. The friction between moving parts in an engine remains practically constant throughout an engine's speed range. Therefore, the mechanical efficiency of an engine will be highest when the engine is running at the r.p.m. at which maximum b.hp. is developed. Mechanical efficiency of the average aircraft reciprocating engine approaches 90%.

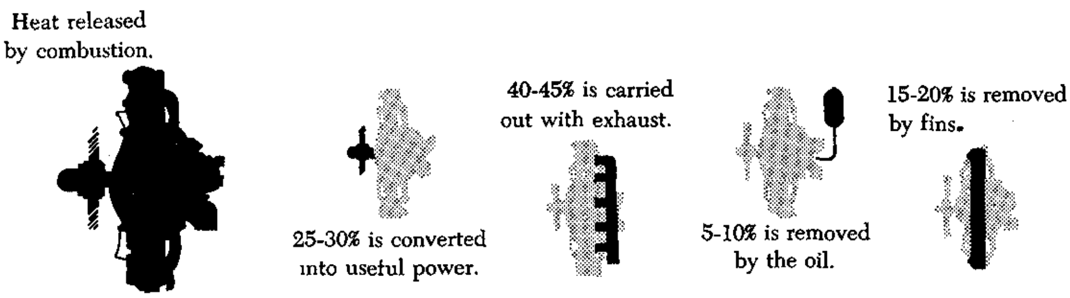


FIGURE 1–31. Thermal distribution in an engine.

Volumetric Efficiency

Volumetric efficiency, another engine efficiency, is a ratio expressed in terms of percentages. It is a comparison of the volume of fuel/air charge (corrected for temperature and pressure) inducted into the cylinders to the total piston displacement of the engine. Various factors cause departure from a 100% volumetric efficiency.

The pistons of an unsupercharged engine displace the same volume each time they sweep the cylinders from top center to bottom center. The amount of charge that fills this volume on the intake stroke depends on the existing pressure and temperature of the surrounding atmosphere. Therefore, to find the volumetric efficiency of an engine, standards for atmospheric pressure and temperature had to be established. The U.S. standard atmosphere was established in 1958 and provides the necessary pressure and temperature values to calculate volumetric efficiency.

The standard sea-level temperature is 59° F. or 15° C. At this temperature the pressure of one atmosphere is 14.69 lbs./sq. in., and this pressure will support a column of mercury 29.92 in. high. These standard sea-level conditions determine a standard density, and if the engine draws in a volume of charge of this density exactly equal to its piston displacement, it is said to be operating at 100% volumetric efficiency. An engine drawing in less volume than this has a volumetric efficiency lower than 100%. An engine equipped with a high-speed internal or external blower may have a volumetric efficiency greater than 100%. The equation for volumetric efficiency is:

$$\text{Volumetric efficiency} = \frac{\text{Volume of charge (corrected for temperature and pressure)}}{\text{Piston displacement}}.$$

Many factors decrease volumetric efficiency; some of these are:

(1) Part-throttle operation.
(2) Long intake pipes of small diameter.
(3) Sharp bends in the induction system.
(4) Carburetor air temperature too high.
(5) Cylinder-head temperature too high.
(6) Incomplete scavenging.
(7) Improper valve timing.

Propulsive Efficiency

A propeller is used with an engine to provide thrust. The engine supplies b.hp. through a rotating shaft, and the propeller absorbs the b.hp. and converts it into thrust hp. In this conversion, some power is wasted. Since the efficiency of any machine is the ratio of useful power output to the power input, propulsive efficiency (in this case, propeller efficiency) is the ratio of thrust hp. to b.hp. On the average, thrust hp. constitutes approximately 80% of the b.hp. The other 20% is lost in friction and slippage. Controlling the blade angle of the propeller is the best method of obtaining maximum propulsive efficiency for all conditions encountered in flight.

During takeoff, when the aircraft is moving at low speeds and when maximum power and thrust are required, a low propeller blade angle will give maximum thrust. For high-speed flying or diving, the blade angle is increased to obtain maximum thrust and efficiency. The constant-speed propeller is used to give required thrust at maximum efficiency for all flight conditions.

TURBINE ENGINE CONSTRUCTION

In a reciprocating engine the functions of intake, compression, combustion, and exhaust all take place in the same combustion chamber; consequently, each must have exclusive occupancy of the chamber during its respective part of the combustion cycle. A significant feature of the gas turbine engine, however, is that a separate section is devoted to each function, and all functions are performed simultaneously without interruption.

A typical gas turbine engine consists of:

(1) An air inlet.
(2) Compressor section.
(3) Combustion section.
(4) Turbine section.
(5) Exhaust section.
(6) Accessory section.
(7) The systems necessary for starting, lubrication, fuel supply, and auxiliary purposes, such as anti-icing, cooling, and pressurization.

The major components of all turbine engines are basically the same; however, the nomenclature of the component parts of various engines currently in use will vary slightly due to the difference in each manufacturer's terminology. These differences are reflected in the applicable maintenance manuals.

The greatest single factor influencing the construction features of any gas turbine engine is the type compressor (axial flow or centrifugal flow) for which the engine is designed. Later in the chapter a detailed description of compressors is given, but for the time being examine figures 1–32 and 1–33. Notice the physical effect the two types of compressors have on engine construction features. It is obvious that there is a difference in their length and diameter.

Note that in the axial-flow engine the air inlet

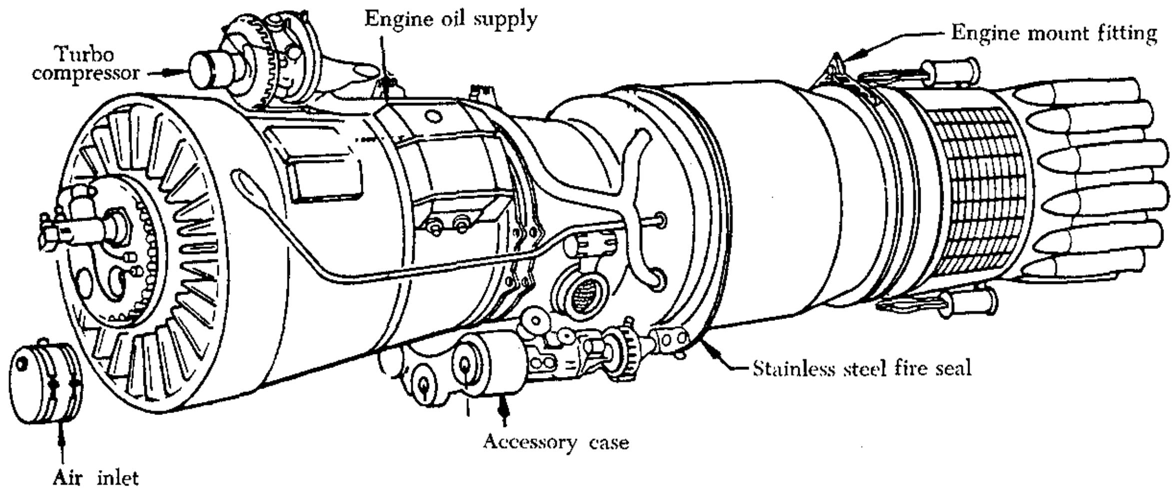


FIGURE 1–32. Axial-flow engine.

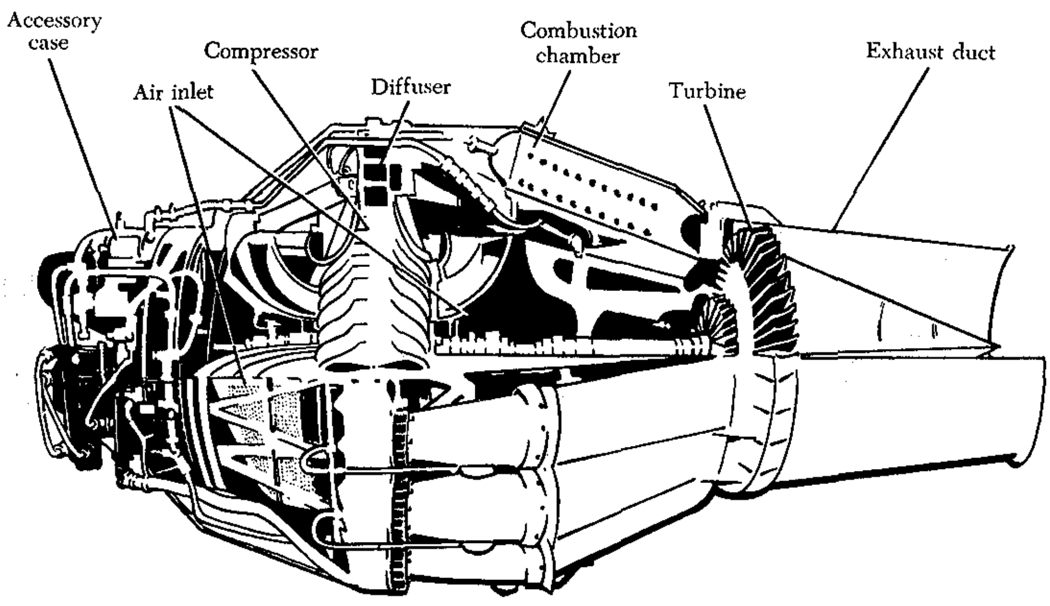


FIGURE 1–33. Centrifugal-flow engine.

duct is one of the major engine components; on the other hand, in the centrifugal-flow engine, air enters the air inlet and is directed to the compressor inducer vanes through circumferential inlets located in front and back of the impeller. The inlets are screened to prevent entry of foreign objects that could cause serious damage to the metal components if allowed to enter the compressor.

The accessories of the two types of engines are located at different points on the engines. This is necessary because of engine construction. The front of the axial-flow engine is utilized for air entrance; consequently, the accessories must be located elsewhere.

Other than the features previously mentioned, there is little visual dissimilarity between the remaining major components of the two engines.

AIR ENTRANCE

The air entrance is designed to conduct incoming air to the compressor with a minimum energy loss resulting from drag or ram pressure loss; that is, the flow of air into the compressor should be free of turbulence to achieve maximum operating efficiency. Proper design contributes materially to aircraft performance by increasing the ratio of compressor discharge pressure to duct inlet pressure.

The amount of air passing through the engine is dependent upon three factors:

(1) The compressor speed (r.p.m.).
(2) The forward speed of the aircraft.
(3) The density of the ambient (surrounding) air.

Inlets may be classified as:

(1) *Nose* inlets, located in the nose of the fuselage, or powerplant pod or nacelle.
(2) *Wing* inlets, located along the leading edge of the wing, usually at the root for single-engine installations.
(3) *Annular* inlets, encircling, in whole or in part, the fuselage or powerplant pod or nacelle.
(4) *Scoop* inlets, which project beyond the immediate surface of the fuselage or nacelle.
(5) *Flush* inlets, which are recessed in the side of the fuselage, powerplant pod, or nacelle.

There are two basic types of air entrances in use: the single entrance and the divided entrance. Generally, it is advantageous to use a single entrance with an axial-flow engine to obtain maximum ram pressure through straight flow. It is used almost exclusively on wing or external installations where the unobstructed entrance lends itself readily to a single, short, straight duct.

A divided entrance offers greater opportunity to diffuse the incoming air and enter the plenum chamber with the low velocity required to utilize efficiently a double-entry compressor. (The plenum chamber is a storage place for ram air, usually asso-

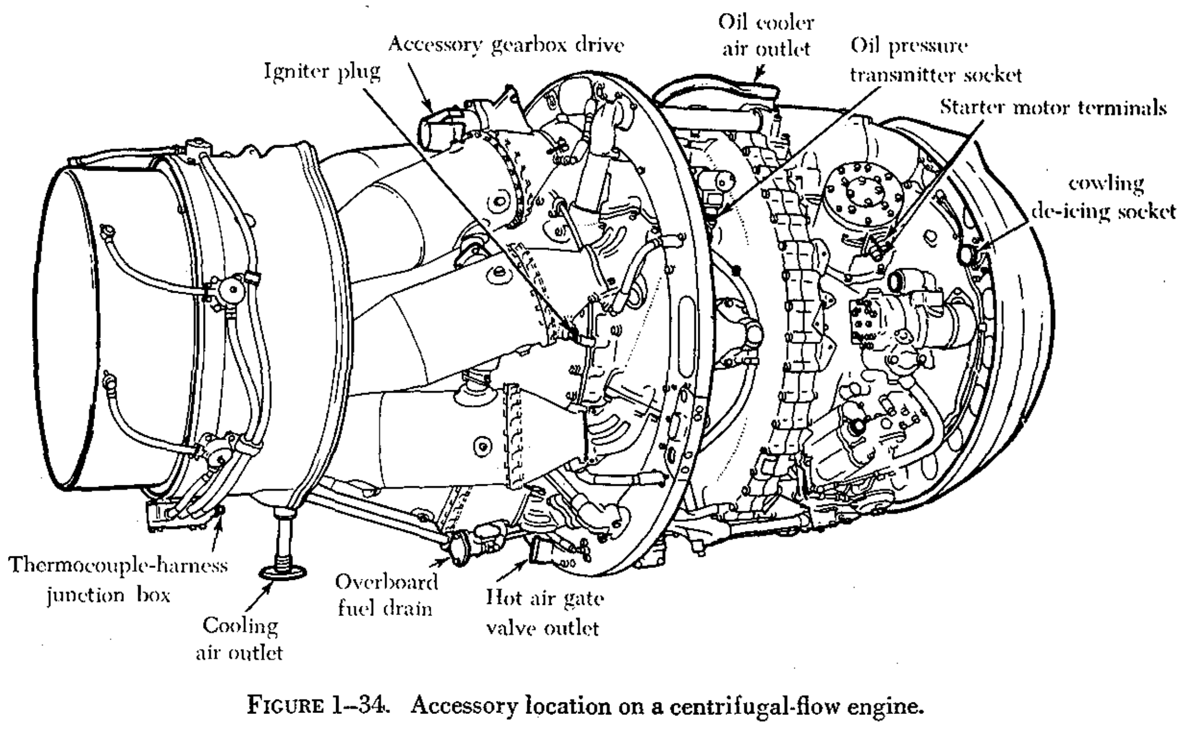


FIGURE 1-34. Accessory location on a centrifugal-flow engine.

ciated with fuselage installations.) It is also advantageous when the equipment installation or pilot location makes the use of a single or straight duct impractical. In most cases the divided entrance permits the use of very short ducts with a resultant small pressure drop through skin friction.

ACCESSORY SECTION

The accessory section of the turbojet engine has various functions. The primary function is to provide space for the mounting of accessories necessary for operation and control of the engine. Generally, it also includes accessories concerned with the aircraft, such as electric generators and fluid power-pumps. Secondary functions include acting as an oil reservoir and/or oil sump, and housing the accessory drive gears and reduction gears.

The arrangement and driving of accessories have always been major problems on gas turbine engines. Driven accessories are usually mounted on common pads either ahead of or adjacent to the compressor section, depending on whether the engine is centrifugal flow or axial flow. Figures 1–34 and 1–35 illustrate the accessory arrangement of a centrifugal-flow engine and an axial-flow engine, respectively.

The components of the accessory section of all centrifugal- and axial-flow engines have essentially the same purpose even though they often differ quite extensively in construction details and nomenclature.

The basic elements of the centrifugal-flow engine accessory section are (1) the accessory case, which has machined mounting pads for the engine-driven accessories, and (2) the gear train, which is housed within the accessory case.

The accessory case may be designed to act as an oil reservoir. If an oil tank is utilized, a sump is usually provided below the front bearing support for the drainage and scavenging of oil used to lubricate bearings and drive gears.

The accessory case is also provided with adequate tubing or cored passages for spraying lubricating oil on the gear train and supporting bearings.

The gear train is driven by the engine rotor through an accessory drive shaft gear coupling, which splines with a shaft gear and the rotor assembly compressor hub. The reduction gearing within the case provides suitable drive speeds for each engine accessory or component. Because the rotor

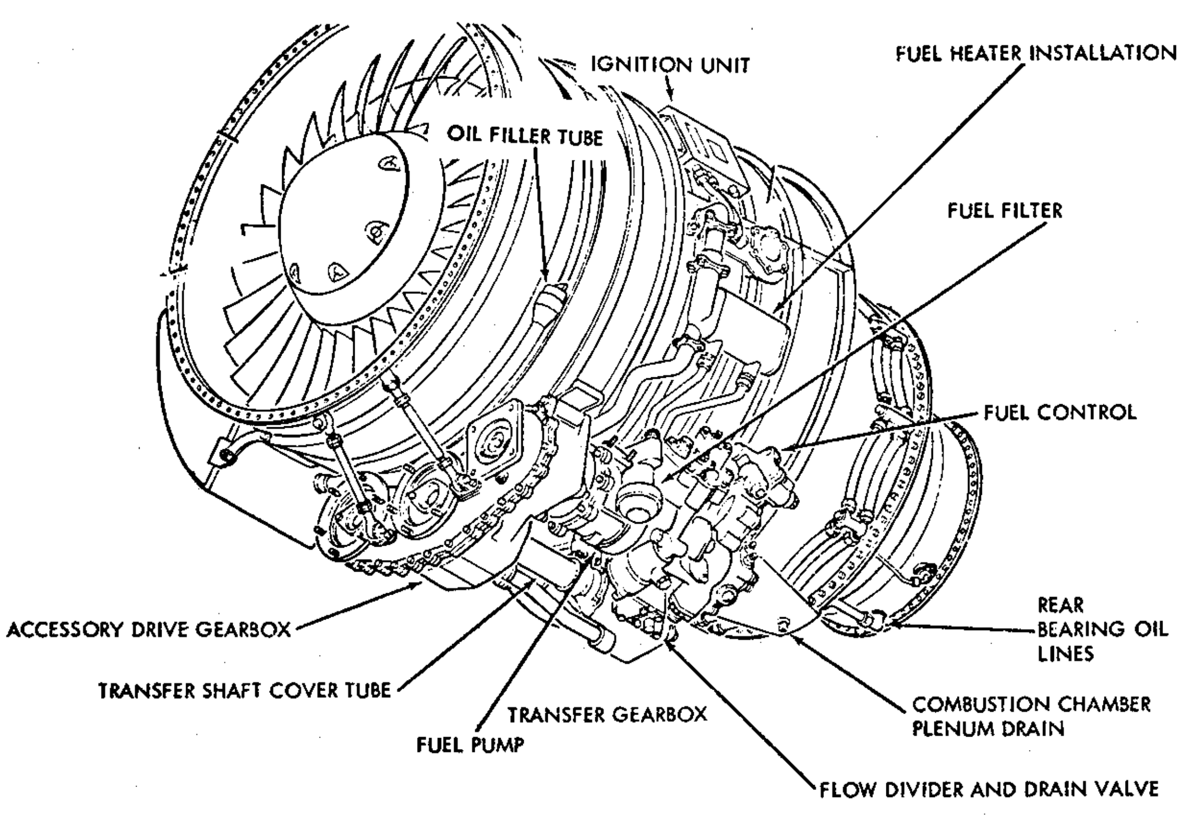


FIGURE 1–35. Accessory arrangement on an axial-flow engine.

operating r.p.m. is so high, the accessory reduction gear ratios are relatively high. The accessory drives are supported by ball bearings assembled in the mounting pad bores of the accessory case.

The components of an axial-flow engine accessory section are an accessory gearbox and a power takeoff assembly, housing the necessary drive shafts and reduction gears. Figure 1–36 shows the location of the accessory gearbox.

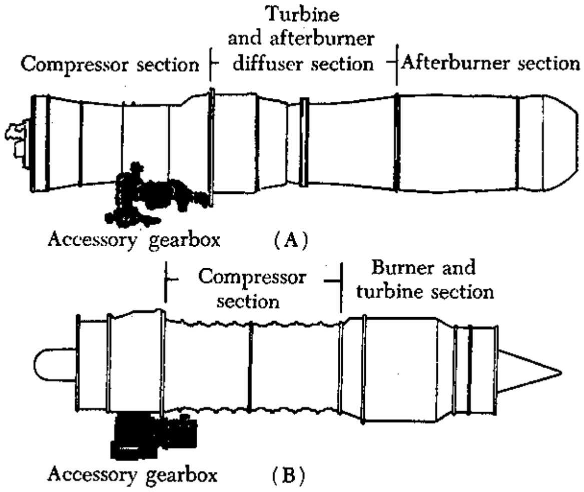


FIGURE 1–36. (A) Accessory gearbox mounted beneath the compressor; (B) Accessory gearbox mounted beneath the front bearing support.

Although the close relationship of the accessory gearbox and the power takeoff necessitates their being located near each other, two factors affect the location of gearboxes. They are engine diameter and engine installation.

Designers are forever striving to reduce engine diameter to make the engine more streamlined, thereby increasing aircraft performance by reducing drag. Also, engine installation in a particular aircraft may dictate the location or re-arrangement of the accessory gearboxes.

The accessory gearbox has basically the same functions as the accessory case of the centrifugal-flow engine. It has the usual machined mounting pads for the engine accessories, and it houses and supports the accessory drive gear trains. Also included are adequate tubing and cored passages for lubricating the gear trains and their supporting bearings.

The accessories usually provided on engines are the fuel control with its governing device; the high-pressure fuel pump(s); oil pressure pump and scavenge pump(s); auxiliary fuel pump and sometimes a starting fuel pump; and several engine accessories including starter, generator, and tachometer. Although these accessories are for the most part essential, the particular combination of engine-driven accessories depends upon the use for which the engine is designed.

The accessories mentioned above (except starters) are the engine-driven type. Also associated with the engine systems are the nondriven accessories, such as ignition exciters, fuel or oil filters, barometric units, drip valves, compressor bleed valves, and relief valves.

COMPRESSOR SECTION

The compressor section of the turbojet engine has many functions. Its primary function is to supply air in sufficient quantity to satisfy the requirements of the combustion burners. Specifically, to fulfill its purpose, the compressor must increase the pressure of the mass of air received from the air inlet duct and then discharge it to the burners in the quantity and at the pressures required.

A secondary function of the compressor is to supply bleed-air for various purposes in the engine and aircraft.

The bleed-air is taken from any of the various pressure stages of the compressor. The exact location of the bleed ports is, of course, dependent on the pressure or temperature required for a particular job. The ports are small openings in the compressor case adjacent to the particular stage from which the air is to be bled; thus, varying degrees of pressure or heat are available simply by tapping into the appropriate stage. Air is often bled from the final or highest pressure stage, since at this point, pressure and air temperature are at a maximum. At times it may be necessary to cool this high-pressure air. If it is used for cabin pressurization or other purposes where excess heat would be uncomfortable or detrimental, the air is sent through a refrigeration unit.

Bleed air is utilized in a wide variety of ways, including driving the previously mentioned remote-driven accessories. Some of the current applications of bleed air are:

(1) Cabin pressurization, heating, and cooling.
(2) Deicing and anti-icing equipment.
(3) Pneumatic starting of engines.
(4) Auxiliary drive units (ADU).

(5) Control-booster servosystems.

(6) Power for running instruments.

The compressor section's location depends on the type of compressor. Figures 1–32 and 1–33 have already illustrated how the arrangement of engine components varies with compressor type. In the centrifugal-flow engine, the compressor is located between the accessory section and the combustion section; in the axial-flow engine the compressor is located between the air inlet duct and the combustion section.

Compressor Types

The two principal types of compressors currently being used in turbojet aircraft engines are centrifugal flow and axial flow. The compressor type is a means of engine classification.

Much use has been made of the terms "centrifugal flow" and "axial flow" to describe the engine and compressor. However, the terms are applicable to the flow of air through the compressor.

In the centrifugal-flow engine, the compressor achieves its purpose by picking up the entering air and accelerating it outwardly by centrifugal action. In the axial-flow engine, the air is compressed while continuing in its original direction of flow, thus avoiding the energy loss caused by turns. From inlet to exit the air flows along an axial path and is compressed at a ratio of approximately 1.25:1 per stage. The components of each of these two types of compressors have their individual functions in the compression of air for the combustion section.

Centrifugal-flow Compressors

The centrifugal-flow compressor consists basically of an impeller (rotor), a diffuser (stator), and a compressor manifold, illustrated in figure 1–37. The two main functional elements are the impeller and the diffuser. Although the diffuser is a separate unit and is placed inside and bolted to the manifold; the entire assembly (diffuser and manifold) is often referred to as the diffuser. For clarification during compressor familiarization, the units are treated individually.

The impeller is usually made from forged aluminum alloy, heat-treated, machined, and smoothed for minimum flow restriction and turbulence.

In some types the impeller is fabricated from a single forging. This type impeller is shown in figure 1–37 (A). In other types the curved inducer vanes are separate pieces as illustrated in figure 1–38.

The impeller, whose function is to pick up and accelerate the air outwardly to the diffuser, may be either of two types—single entry or double entry. Both are similar in construction to the reciprocating engine supercharger impeller, the double-entry type being similar to two impellers back to back. However, because of the much greater combustion air requirements in turbojet engines, the impellers are larger than supercharger impellers.

The principal differences between the two types of impellers are the size and the ducting arrangement. The double-entry type has a smaller diameter, but is usually operated at a higher rotational speed to assure sufficient airflow. The single-entry impeller permits convenient ducting directly to the impeller eye (inducer vanes) as opposed to the more complicated ducting necessary to reach the rear side of the double-entry type. Although slightly more efficient in receiving air, the single-entry impeller must be large in diameter to deliver the same quantity of air as the double-entry type. This, of course, increases the overall diameter of the engine.

Included in the ducting for double-entry compressor engines is the plenum chamber. This chamber is necessary for a double-entry compressor because the air must enter the engine at almost right angles to the engine axis. Therefore, the air must, in order to give a positive flow, surround the engine compressor at a positive pressure before entering the compressor.

Included in some installations, as a necessary part of the plenum chamber, are the auxiliary air-intake doors (blow-in doors). These blow-in doors admit air to the engine compartment during ground operation, when air requirements for the engine are in excess of the airflow through the inlet ducts. The doors are held closed by spring action when the engine is not operating. During operation, however, the doors open automatically whenever engine compartment pressure drops below atmospheric pressure. During takeoff and flight, ram air pressure in the engine compartment aids the springs in holding the doors closed.

The diffuser is an annular chamber provided with a number of vanes forming a series of divergent passages into the manifold. The diffuser vanes direct the flow of air from the impeller to the manifold at an angle designed to retain the maximum amount of energy imparted by the impeller. They also deliver the air to the manifold at a velocity and pressure satisfactory for use in the combustion

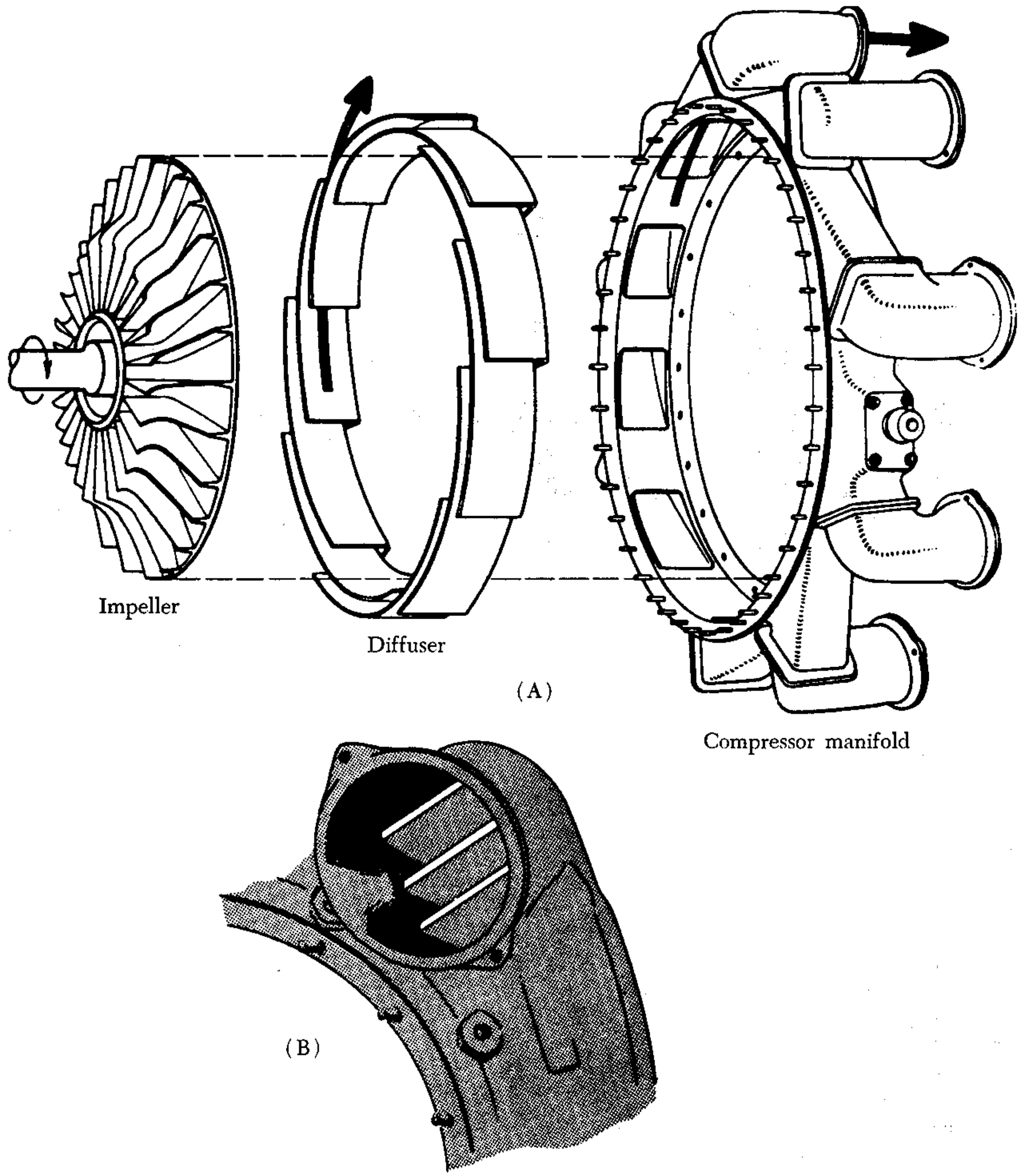


FIGURE 1–37. (A) Components of a centrifugal compressor; (B) Air outlet elbow with turning vanes for reducing air pressure losses.

chambers. Refer to figure 1–37 (A) and notice the arrow indicating the path of airflow through the diffuser, then through the manifold.

The compressor manifold shown in figure 1–37 (A) diverts the flow of air from the diffuser, which is an integral part of the manifold, into the combustion chambers. The manifold will have one outlet port for each chamber so that the air is evenly divided. A compressor outlet elbow is bolted to each of the outlet ports. These air outlets are constructed in the form of ducts and are known by a variety of names, such as air outlet ducts, outlet elbows, or

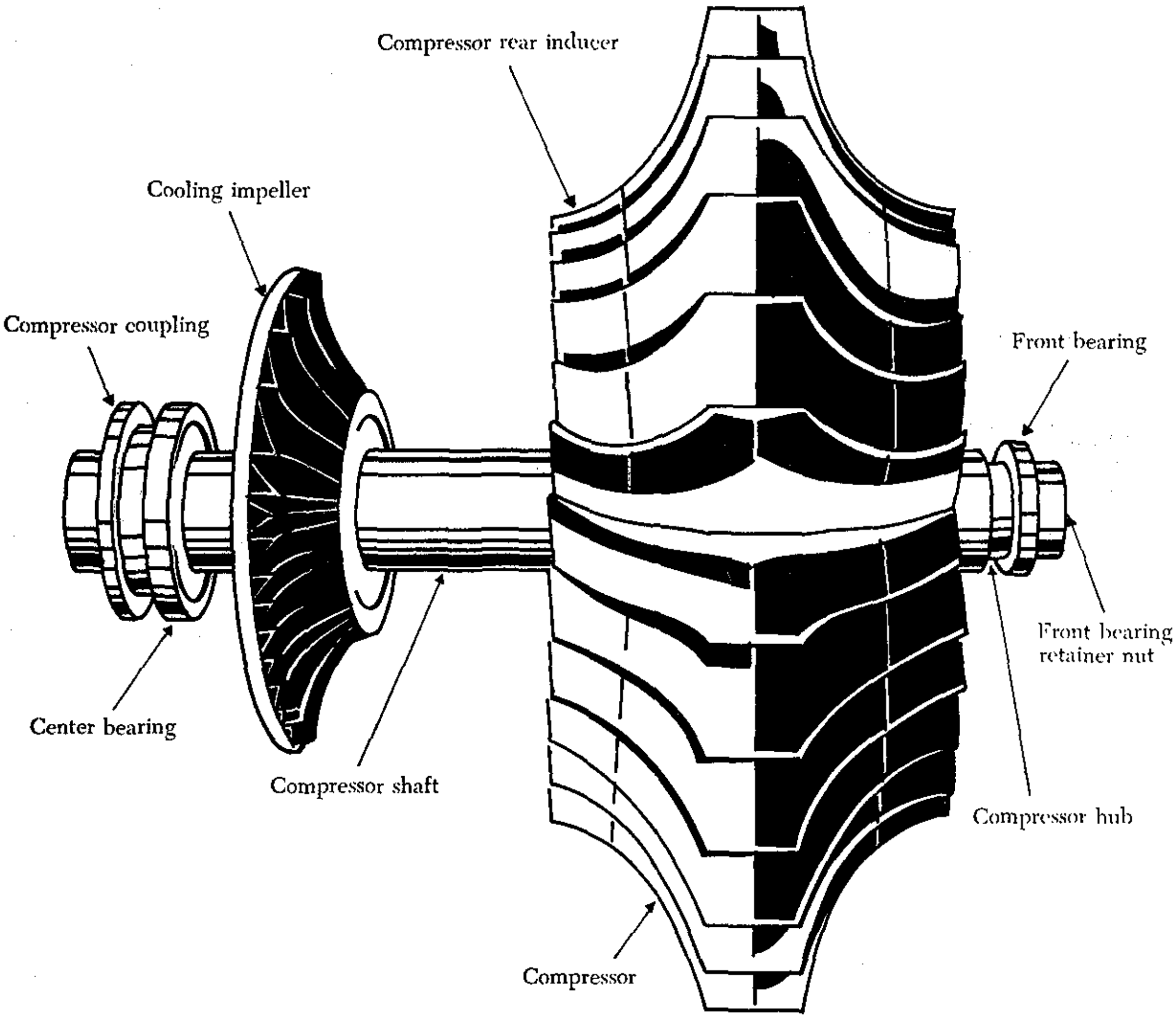


FIGURE 1–38. Double entry impeller with inducer vanes as separate pieces.

combustion chamber inlet ducts. Regardless of the terminology used, these outlet ducts perform a very important part of the diffusion process; that is, they change the radial direction of the airflow to an axial direction, where the diffusion process is completed after the turn. To help the elbows perform this function in an efficient manner, turning vanes (cascade vanes) are sometimes fitted inside the elbows. These vanes reduce air pressure losses by presenting a smooth, turning surface. (See figure 1–37 (B).)

Axial-flow Compressor

The axial-flow compressor has two main elements, a rotor and a stator. The rotor has blades fixed on a spindle. These blades impel air rearward in the same manner as a propeller because of their angle and airfoil contour. The rotor, turning at high speed, takes in air at the compressor inlet and impels it through a series of stages. The action of the rotor increases the compression of the air at each stage and accelerates it rearward through several stages. With this increased velocity, energy is transferred from the compressor to the air in the form of velocity energy. The stator blades act as diffusers at each stage, partially converting high velocity to pressure. Each consecutive pair of rotor and stator blades constitutes a pressure stage. The number of rows of blades (stages) is determined by the amount of air and total pressure rise required. The greater the number of stages, the higher the compression ratio. Most present-day engines utilize from 10 to 16 stages.

The stator has rows of blades, or vanes, dovetailed into split rings, which are in turn attached inside an enclosing case. The stator vanes project radially toward the rotor axis and fit closely on either side of each stage of the rotor.

The compressor case, into which the stator vanes are fitted, is horizontally divided into halves. Either the upper or lower half may be removed for inspection or maintenance of rotor and stator blades.

The function of the vanes is twofold. They are designed to receive air from the air inlet duct or from each preceding stage of the compressor and deliver it to the next stage or to the burners at a workable velocity and pressure. They also control the direction of air to each rotor stage to obtain the maximum possible compressor blade efficiency. Shown in figure 1–39 are the rotor and stator elements of a typical axial-flow compressor.

The rotor blades are usually preceded by an inlet guide vane assembly. The guide vanes direct the airflow into the first stage rotor blades at the proper angle and impart a swirling motion to the air entering the compressor. This pre-swirl, in the direction of engine rotation, improves the aerodynamic characteristics of the compressor by reducing the drag on the first-stage rotor blades. The inlet guide vanes are curved steel vanes usually welded to steel inner and outer shrouds. The inlet guide vanes may be preceded by a protective inlet screen. This screen reduces the chance of accidental entry of foreign bodies, such as stones, dirt, clothing, or other debris, into the compressor.

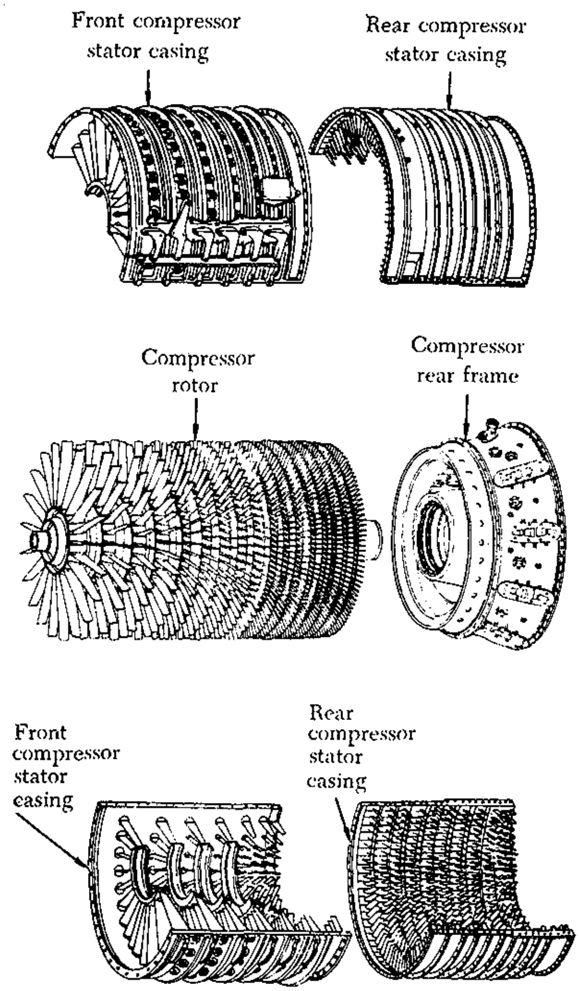


Figure 1–39. Rotor and stator components of an axial-flow compressor.

At the discharge end of the compressor, the stator vanes are constructed to straighten the airflow to eliminate turbulence. These vanes are called straightening vanes or the outlet vane assembly.

The casings of axial-flow compressors not only support the stator vanes and provide the outer wall of the axial path the air follows, but they also provide the means for extracting compressor air for various purposes.

The stator vanes are usually made of steel with corrosion- and erosion-resistant qualities. Quite frequently they are shrouded (or enclosed) by a band of suitable material to simplify the fastening problem. The vanes are welded into the shrouds, and the outer shroud is secured to the compressor housing inner wall by radial retaining screws.

The rotor blades are usually made of stainless steel. Methods of attaching the blades in the rotor disk rims vary in different designs, but they are commonly fitted into disks by either bulb-type or fir-tree-type roots. (See figure 1–40.) The blades are then locked by means of screws, peening, locking wires, pins, or keys.

Compressor blade tips are reduced in thickness by cutouts, referred to as blade "profiles." These profiles prevent serious damage to the blade or housing should the blades contact the compressor housing. This condition can occur if rotor blades become excessively loose or if rotor support is reduced by a malfunctioning bearing. Even though blade profiles greatly reduce such possibilities, occasionally a blade may break under stress of rubbing and cause considerable damage to compressor blades and stator vane assemblies.

The blades vary in length from entry to discharge because the annular working space (drum to casing) is reduced progressively toward the rear by the decrease in the casing diameter (see figure 1–41).

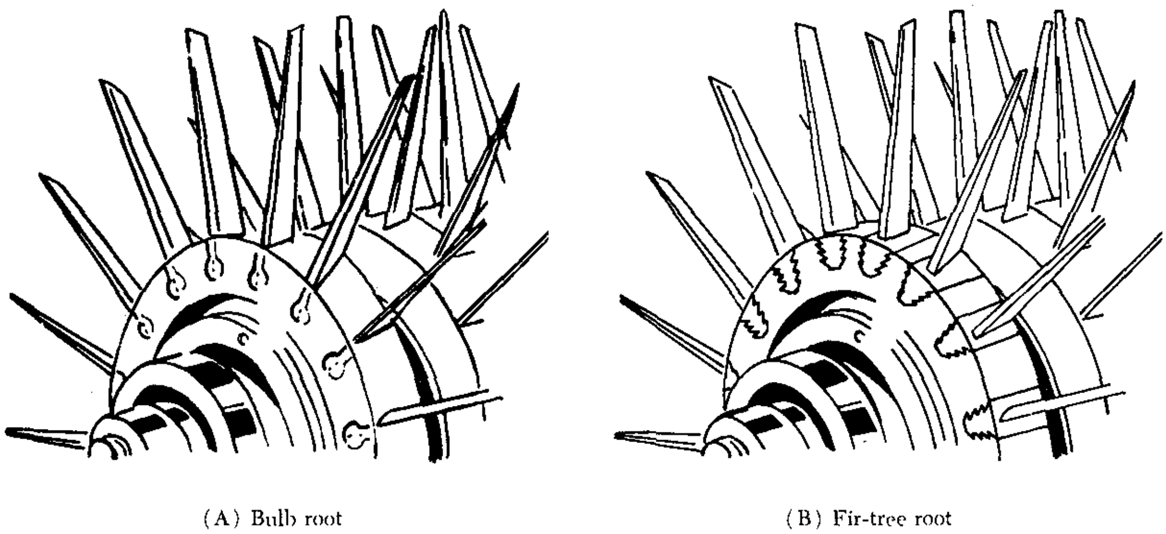


FIGURE 1–40. Common retention methods used on compressor rotor blades.

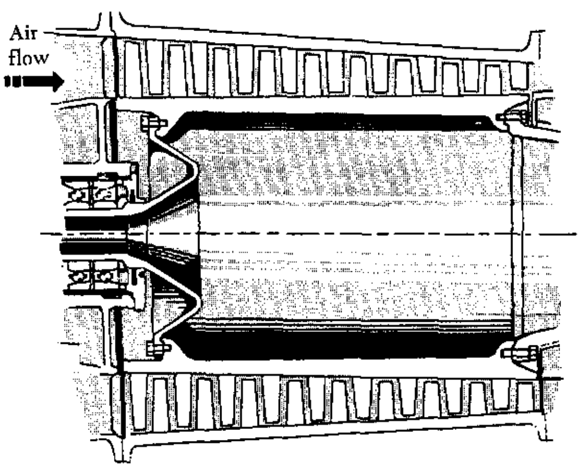


FIGURE 1–41. Drum-type compressor rotor.

Before leaving the subject of rotor familiarization, it may be well to mention that the rotor features either drum-type or disk-type construction.

The drum-type rotor consists of rings that are flanged to fit one against the other, wherein the entire assembly can then be held together by through bolts. This type of construction is satisfactory for low-speed compressors where centrifugal stresses are low.

The disk-type rotor consists of a series of disks machined from aluminum forgings, shrunk over a steel shaft, with rotor blades dovetailed into the disk rims. Another method of rotor construction is to machine the disks and shaft from a single aluminum forging, and then to bolt steel stub shafts on the front and rear of the assembly to provide bearing support surfaces and splines for joining the turbine shaft. The disk-type rotors are used almost exclusively in all present-day, high-speed engines and are the type referred to in this text. The drum-type and disk-type rotors are illustrated in figures 1–41 and 1–42, respectively.

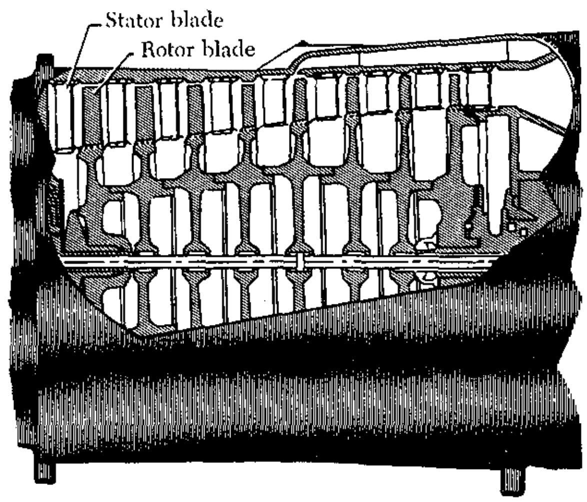


FIGURE 1–42. Disk-type compressor rotor.

The coverage of axial-flow compressors up to this point has dealt solely with the conventional single-rotor type. Actually, there are two configurations of the axial compressor currently in use, the single rotor and the dual rotor, sometimes referred to as solid spool and split spool, respectively.

One version of the solid-spool compressor uses variable inlet guide vanes. Also, the first few rows of stator vanes are variable. This is the arrangement on the General Electric CJ805 engine. It incorporates a 17-stage compressor, and the angles of the inlet guide vanes and the first six stages of the stator vanes are variable. During operation, air enters the front of the engine and is directed into the compressor at the proper angle by the variable inlet guide and variable stator vanes. The air is compressed and forced into the combustion section. A fuel nozzle which extends into each combustion liner atomizes the fuel for combustion.

These variables are controlled in direct relation to the amount of power the engine is required to produce by the pilot's power lever position.

One version of the split-spool compressor is found in Pratt and Whitney's JT3C engine. It incorporates two compressors with their respective turbines and interconnecting shafts, which form two physically independent rotor systems.

As previously mentioned, centrifugal- and axial-flow engines dominate the gas-turbine field. There are, however, several possible configurations of these engine types, some of which have been tried experimentally, while others are still in the design or laboratory stage of development.

From an analysis of the centrifugal- and axial-flow engine compressors at their present stage of development, the axial-flow type appears to have definite advantages. The advent of the split-spool axial compressor made these advantages even more positive by offering greater starting flexibility and improved high-altitude performance.

The advantages and disadvantages of both types of compressors are included in the following list. Bear in mind that even though each compressor type has merits and limitations, the performance potential is the key to further development and use.

The centrifugal-flow compressor's advantages are:

(1) High pressure rise per stage.
(2) Good efficiencies over wide rotational speed range.
(3) Simplicity of manufacture, thus low cost.
(4) Low weight.
(5) Low starting power requirements.

The centrifugal-flow compressor's disadvantages are:

(1) Large frontal area for given airflow.
(2) More than two stages are not practical because of losses in turns between stages.

The axial-flow compressor's advantages are:

(1) High peak efficiencies.
(2) Small frontal area for given airflow.
(3) Straight-through flow, allowing high ram efficiency.
(4) Increased pressure rise by increasing number of stages with negligible losses.

The axial-flow compressor's disadvantages are:

(1) Good efficiencies over only narrow rotational speed range.
(2) Difficulty of manufacture and high cost.
(3) Relatively high weight.
(4) High starting power requirements. (This has been partially overcome by split compressors.)

COMBUSTION SECTION

The combustion section houses the combustion process, which raises the temperature of the air passing through the engine. This process releases energy contained in the air/fuel mixture. The major part of this energy is required at the turbine to drive the compressor. The remaining energy creates the reaction or propulsion and passes out the rear of the engine in the form of a high-velocity jet.

The primary function of the combustion section is, of course, to burn the fuel/air mixture, thereby adding heat energy to the air. To do this efficiently the combustion chamber must:

(1) Provide the means for proper mixing of the fuel and air to assure good combustion.
(2) Burn this mixture efficiently.
(3) Cool the hot combustion products to a temperature which the turbine blades can withstand under operating conditions.
(4) Deliver the hot gases to the turbine section.

The location of the combustion section is directly between the compressor and the turbine sections. The combustion chambers are always arranged coaxially with the compressor and turbine regardless of type, since the chambers must be in a through-flow position to function efficiently.

All combustion chambers contain the same basic elements:

(1) A casing.
(2) A perforated inner liner.
(3) A fuel injection system.
(4) Some means for initial ignition.
(5) A fuel drainage system to drain off unburned fuel after engine shutdown.

There are currently three basic types of combustion chambers, variations within these types being in detail only. These types are:

(1) The multiple-chamber or can type.

(2) The annular or basket type.

(3) The can-annular type.

The can-type combustion chamber is typical of the type used on both centrifugal- and axial-flow engines. It is particularly well suited for the centrifugal compressor engine, since the air leaving the compressor is already divided into equal portions as it leaves the diffuser vanes. It is then a simple matter to duct the air from the diffuser into the respective combustion chambers, arranged radially around the axis of the engine. The number of chambers will vary, since in the past (or development years) as few as two and as many as 16 chambers have been used. The present trend indicates the use of about eight or 10 combustion chambers. Figure 1–43 illustrates the arrangement for can-type combustion chambers. On American-built engines these chambers are numbered in a clockwise direction facing the rear of the engine with the No. 1 chamber at the top.

Each of the can-type combustion chambers consists of an outer case or housing, within which there is a perforated stainless steel (highly heat-resistant) combustion chamber liner or inner liner. (See figure 1–44.) The outer case is divided to facilitate liner replacement. The larger section or chamber body encases the liner at the exit end, and the smaller chamber cover encases the front or inlet end of the liner.

The interconnector (flame propagation) tubes are a necessary part of the can-type combustion chambers. Since each can is a separate burner operating independently of the other cans, there must be some way to spread combustion during the initial starting operation. This is accomplished by interconnecting all the chambers so that as the flame is started by the spark igniter plugs in two of the lower chambers, it will pass through the tubes and ignite the combustible mixture in the adjacent chamber, and continue on until all the chambers are burning.

The flame tubes will vary in construction details from one engine to another, although the basic components are almost identical.

The interconnector tubes are shown in figure 1–45. Bear in mind that not only must the chambers be interconnected by an outer tube (in this case a ferrule), but there must also be a slightly longer tube inside the outer one to interconnect the chamber liners where the flame is located. The outer tubes or jackets around the interconnecting flame tubes not only afford airflow between the chambers, but they also fulfill an insulating function around the hot flame tubes.

The spark igniters previously mentioned are normally two in number, and are located in two

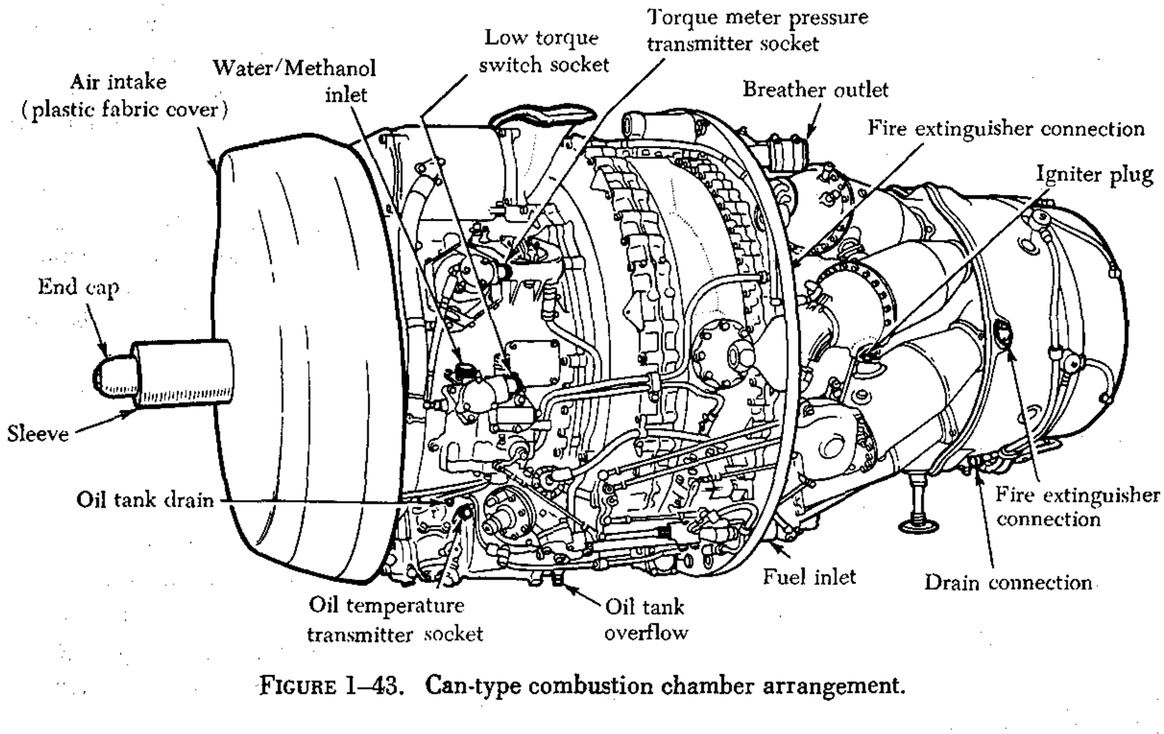


FIGURE 1–43. Can-type combustion chamber arrangement.

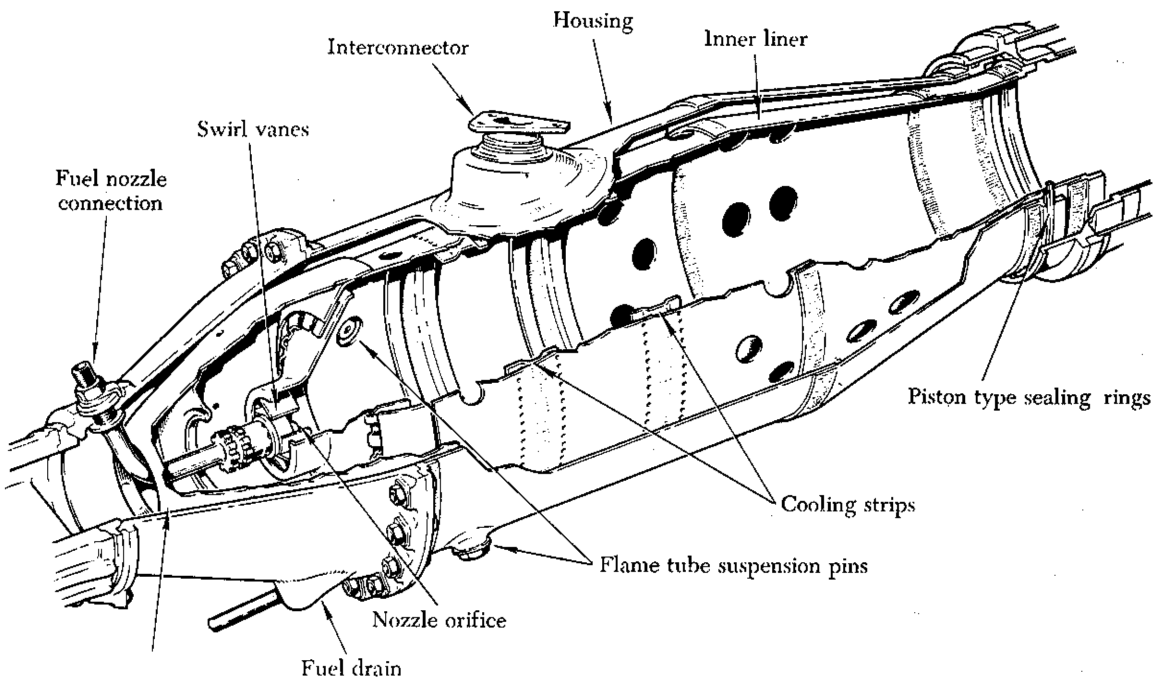


FIGURE 1–44. Can-type combustion chamber.

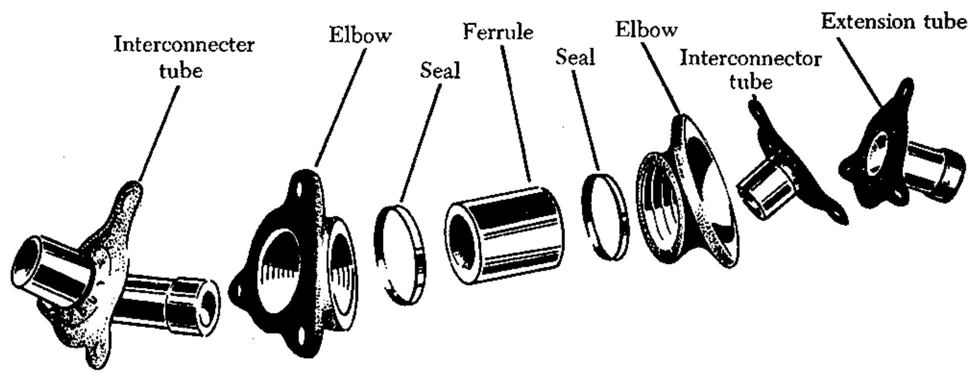


FIGURE 1–45. Interconnecting flame tubes for can-type combustion chambers.

of the can-type combustion chambers.

Another very important requirement in the construction of combustion chambers is providing the means for draining unburned fuel. This drainage prevents gum deposits in the fuel manifold, nozzles, and combustion chambers. These deposits are caused by the residue left when the fuel evaporates. Probably most important is the danger of afterfire if the fuel is allowed to accumulate after shutdown. If the fuel is not drained, a great possibility exists, that at the next starting attempt, the excess fuel in the combustion chamber will ignite, and tailpipe temperature will go beyond safe operating limits.

The liners of the can-type combustors (figure 1–44) have perforations of various sizes and shapes, each hole having a specific purpose and effect on the flame propagation within the liner. The air entering the combustion chamber is divided by the proper holes, louvers, and slots into two main streams—primary and secondary air. The primary or combustion air is directed inside the liner at the front end, where it mixes with the fuel and is burned. Secondary or cooling air passes between the outer casing and the liner and joins the combustion gases through larger holes toward the rear of the liner, cooling the combustion gases from about 3,500° F. to near 1,500° F. To aid in atomization of the fuel, holes are provided around the fuel nozzle in the dome or inlet end of the can-type combustor liner. Louvers are also provided along the axial length of the liners to direct a cooling layer of air along the inside wall of the liner. This layer of air also tends to control the flame pattern by keeping it centered in the liner, thereby preventing burning of the liner walls.

Figure 1–46 illustrates the flow of air through the louvers in the double-annular combustion chamber.

Some provision is always made in the combustion chamber case, or in the compressor air outlet elbow, for installation of a fuel nozzle. The fuel nozzle delivers the fuel into the liner in a finely atomized spray. The finer the spray, the more rapid and efficient the burning process.

Two types of fuel nozzles currently being used in the various types of combustion chambers are the simplex nozzle and the duplex nozzle. The construction features of these nozzles are covered in greater detail in Chapter 3, "Engine Fuel and Fuel Metering Systems."

The annular combustion chamber consists basically of a housing and a liner, as does the can type. The liner consists of an undivided circular shroud extending all the way around the outside of the turbine shaft housing. The chamber may be constructed of one or more baskets; that is, if two or more chambers are used, they are placed one outside the other in the same radial plane, hence, the double-annular chamber. The double-annular chamber is illustrated in figure 1–47.

The spark igniter plugs of the annular combustion chamber are the same basic type used in the can combustion chambers, although construction details may vary. There are usually two plugs mounted on the boss provided on each of the chamber housings. The plugs must be long enough to protrude from the housing into the outer annulus of the double-annular combustion chamber.

The can-annular type combustion chamber is a development by Pratt and Whitney for use in their JT3 axial-flow turbojet engine. Since this engine was to feature the split-spool compressor, it required a combustion chamber capable of meeting the stringent requirements of maximum strength and limited length with a high overall efficiency. These requirements were necessary because of the high air pressures and velocities present in a split-spool compressor, along with the shaft length limitations explained in the following two paragraphs.

The split compressor requires two concentric shafts joining the turbine stages to their respective compressors. The front compressor joined to the rear turbine stages requires the longest shaft. Because this shaft is inside the other, a limitation of diameter is imposed, with the result that the distance between the front compressor and the rear

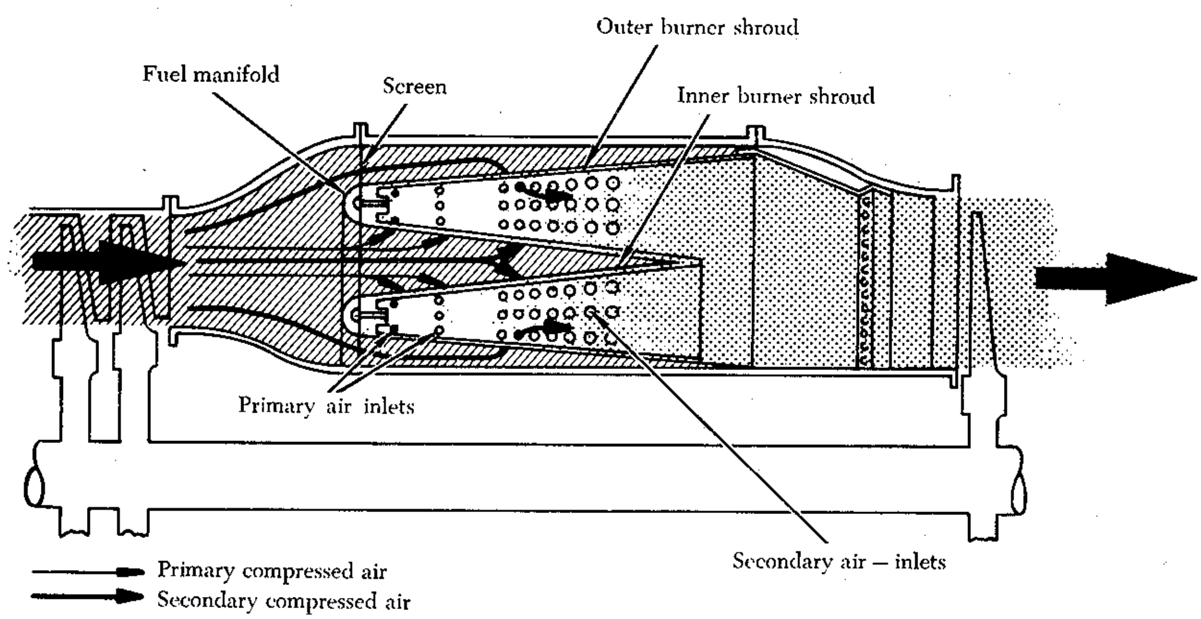


FIGURE 1–46. Components and airflow of a double-annular chamber.

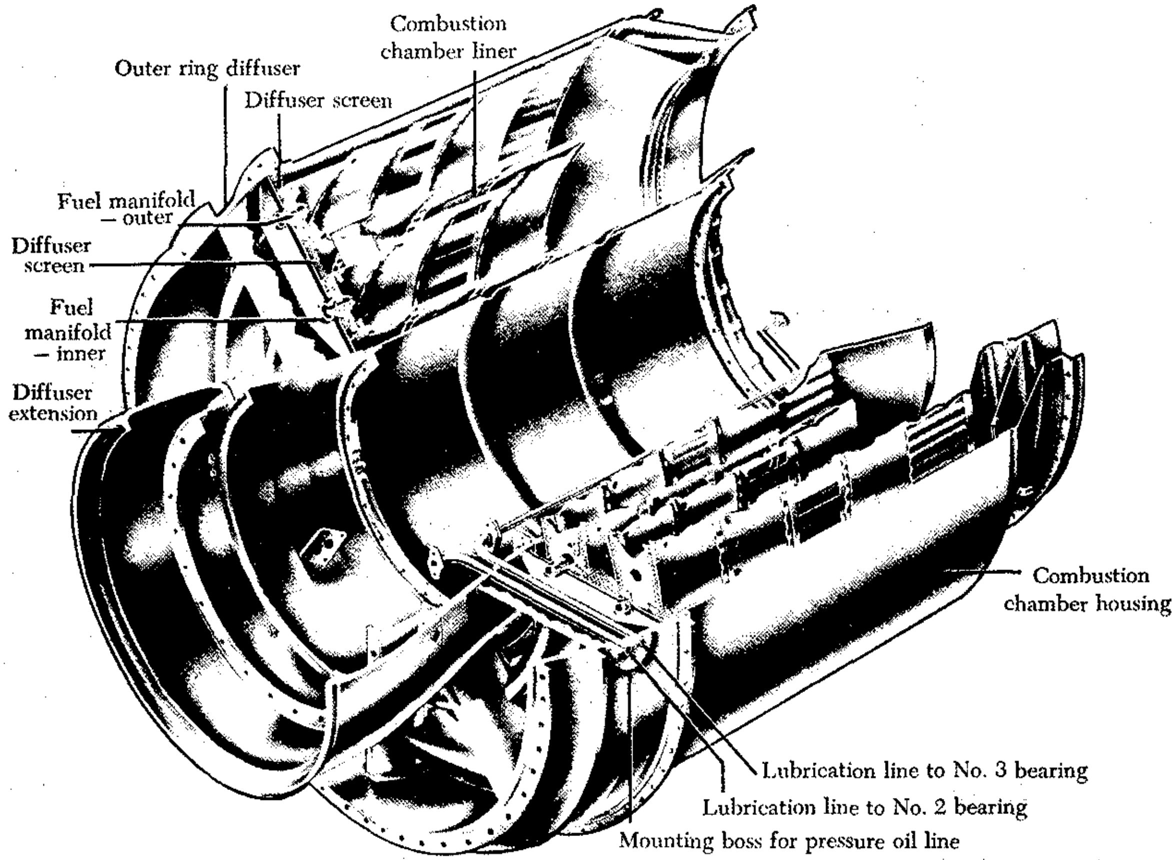


FIGURE 1–47. Double-annular combustion chamber.

turbine must be limited if critical shaft lengths are to be avoided.

Since the compressor and turbine are not susceptible to appreciable shortening, the necessary shaft length limitation had to be absorbed by developing a new type of burner. The designers had to develop a design that would give the desired performance in much less relative distance than had been previously assigned for this purpose.

The can-annular combustion chambers are arranged radially around the axis of the engine, the axis in this instance being the rotor shaft housing. Figure 1–48 shows this arrangement to advantage.

The combustion chambers are enclosed in a removable steel shroud, which covers the entire burner section. This feature makes the burners readily available for any required maintenance.

The burners are interconnected by projecting flame tubes which facilitate the engine-starting process as mentioned previously in the can-type combustion chamber familiarization. These flame tubes function identically with those previously discussed, but they differ in construction details.

Figure 1–48 also shows that each combustion chamber contains a central bullet-shaped perforated liner. The size and shape of the holes are designed to admit the correct quantity of air at the proper velocity and angle required. Cutouts are provided in two of the bottom chambers for installation of the spark igniters. Notice also in figure 1–48 how the combustion chambers are supported at the aft end by outlet duct clamps which secure them to the turbine nozzle assembly.

Again refer to figure 1–48 and notice how the forward face of each chamber presents six apertures which align with the six fuel nozzles of the corresponding fuel nozzle cluster. These nozzles are the dual-orifice (duplex) type requiring the use of a flow-divider (pressurizing valve), as mentioned in the can-type combustion chamber discussion. Around each nozzle are pre-swirl vanes for imparting a swirling motion to the fuel spray, which results in better atomization of the fuel, better burning and efficiency.

The swirl vanes perform two important functions imperative to proper flame propagation:

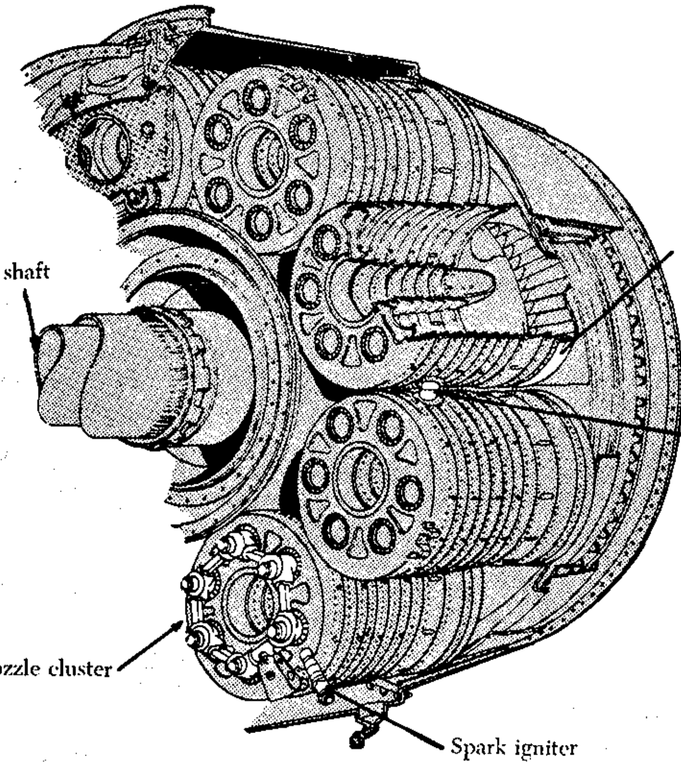


FIGURE 1–48. Can-annular combustion chamber components and arrangement.

(1) High flame speed: Better mixing of air and fuel, ensuring spontaneous burning.
(2) Low air velocity axially: Swirling eliminates flame moving axially too rapidly.

The swirl vanes greatly aid flame propagation, since a high degree of turbulence in the early combustion and cooling stages is desirable. The vigorous mechanical mixing of the fuel vapor with the primary air is necessary, since mixing by diffusion alone is too slow. This same mechanical mixing is also established by other means, such as placing coarse screens in the diffuser outlet, as is the case in most axial-flow engines.

The can-annular combustion chambers also must have the required fuel drain valves located in two or more of the bottom chambers, assuring proper drainage and elimination of residual fuel burning at the next start.

The flow of air through the holes and louvers of the can-annular chambers is almost identical with the flow through other types of burners. Special baffling is used to swirl the combustion airflow and to give it turbulence. Figure 1–49 shows the flow of combustion air, metal cooling air, and the diluent or gas cooling air. Pay particular attention to the direction of airflow indicated by the arrows.

Turbine Section

The turbine transforms a portion of the kinetic (velocity) energy of the exhaust gases into mechanical energy to drive the compressor and accessories. This is the sole purpose of the turbine and this function absorbs approximately 60 to 80% of the total pressure energy from the exhaust gases. The exact amount of energy absorption at the turbine is determined by the load the turbine is driving; that is, the compressor size and type, number of accessories, and a propeller and its reduction gears if the engine is a turbo-propeller type.

The turbine section of a turbojet engine is located aft, or downstream of the combustion chamber section. Specifically, it is directly behind the combustion chamber outlet.

The turbine assembly consists of two basic elements, the stator and the rotor, as does the compressor unit. These two elements are shown in figures 1–50 and 1–51, respectively.

The stator element is known by a variety of names, of which turbine nozzle vanes, turbine guide vanes, and nozzle diaphragm are three of the most commonly used. The turbine nozzle vanes are located directly aft of the combustion chambers and immediately forward of the turbine wheel.

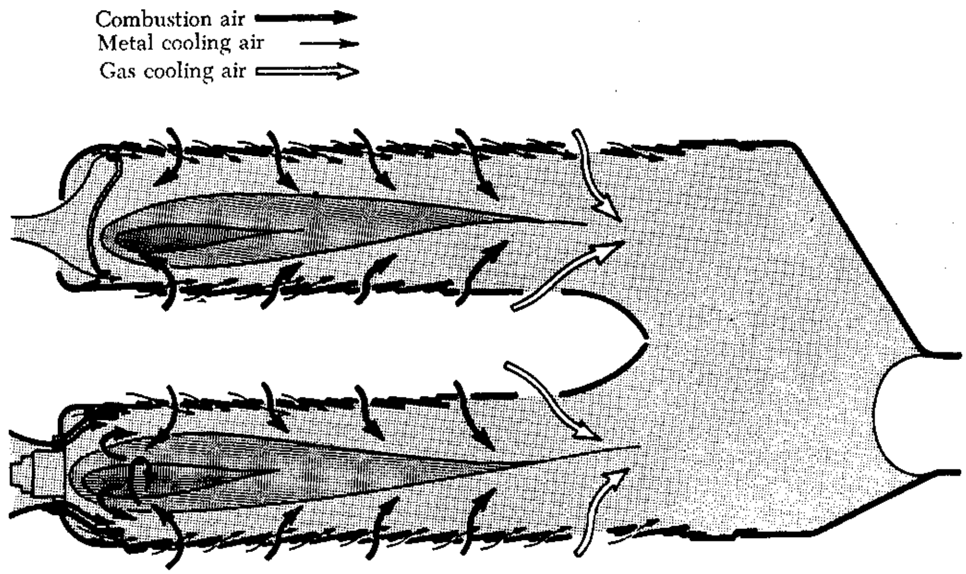


FIGURE 1–49. Airflow through a can-annular chamber.

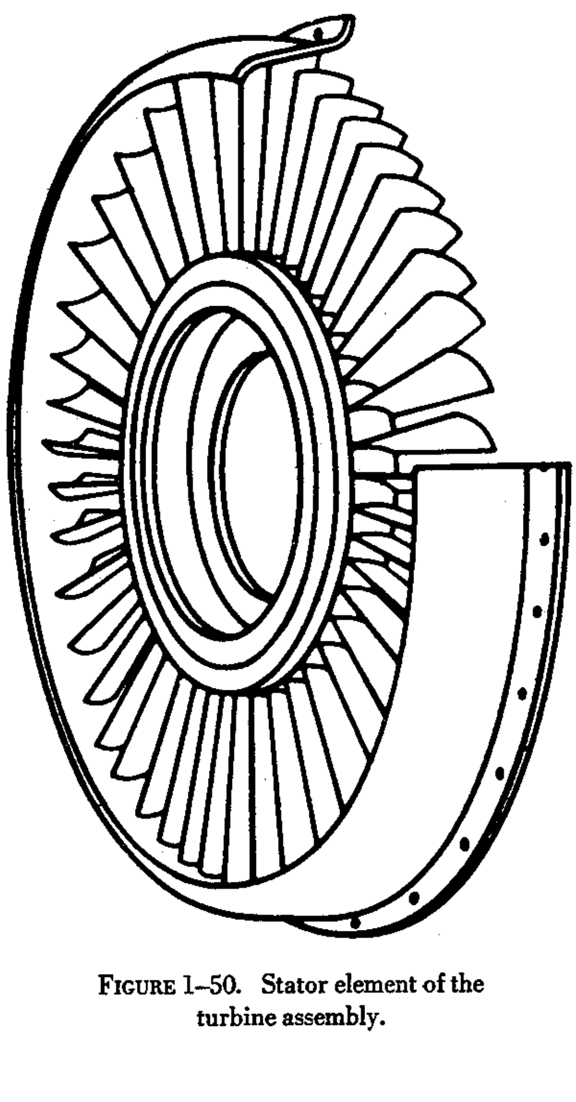


FIGURE 1–50. Stator element of the turbine assembly.

The function of the turbine nozzles is twofold. First, after the combustion chamber has introduced the heat energy into the mass airflow and delivered it evenly to the turbine nozzles, it becomes the job of the nozzles to prepare the mass air flow for driving the turbine rotor. The stationary blades or vanes of the turbine nozzles are contoured and set at such an angle that they form a number of small nozzles discharging the gas at extremely high speed; thus, the nozzle converts a varying portion of the heat and pressure energy to velocity energy which

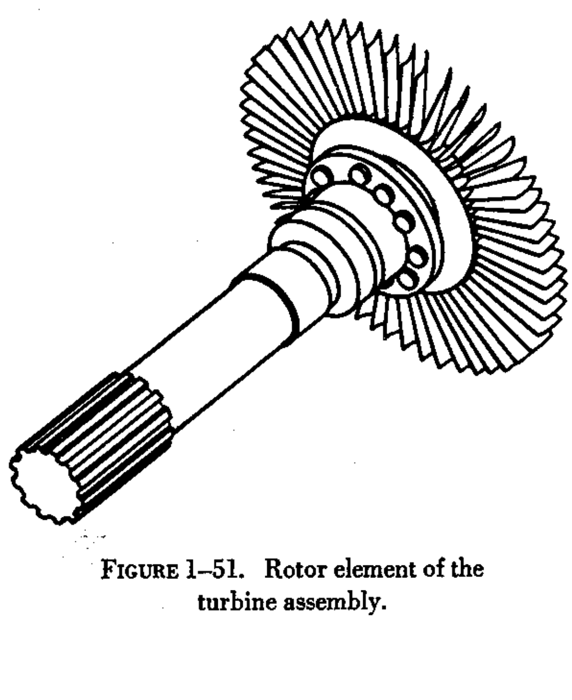
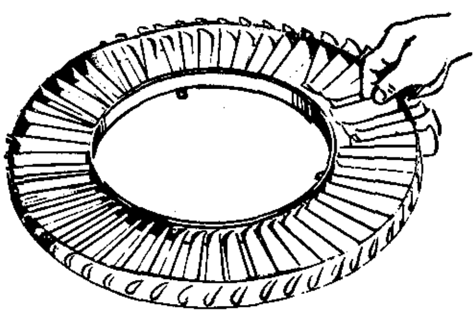


FIGURE 1–51. Rotor element of the turbine assembly.

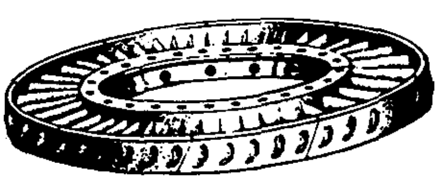
can then be converted to mechanical energy through the rotor blades.

The second purpose of the turbine nozzle is to deflect the gases to a specific angle in the direction of turbine wheel rotation. Since the gas flow from the nozzle must enter the turbine blade passageway while it is still rotating, it is essential to aim the gas in the general direction of turbine rotation.

The turbine nozzle assembly consists of an inner shroud and an outer shroud between which are fixed the nozzle vanes. The number of vanes employed vary with different types and sizes of engines. Figure 1–52 illustrates typical turbine nozzles featuring loose and welded vanes.



(A) Turbine nozzle vane assembly with loose fitting vanes;



(B) Turbine nozzle vane assembly with welded vanes.

FIGURE 1–52. Typical turbine nozzles.

The blades or vanes of the turbine nozzle may be assembled between the outer and inner shrouds or rings in a variety of ways. Although the actual elements may vary slightly in their configuration and construction features, there is one characteristic peculiar to all turbine nozzles; that is, the nozzle vanes must be constructed to allow for thermal expansion. Otherwise, there would be severe distortion or warping of the metal components because of rapid temperature changes.

The thermal expansion of turbine nozzles is accomplished by one of several methods. One method necessitates the vanes being assembled loosely in the supporting inner and outer shrouds. (See figure 1–52(A).)

Each vane fits into a contoured slot in the shrouds, which conforms with the airfoil shape of the vane. These slots are slightly larger than the vanes to give a loose fit. For further support the inner and outer shrouds are encased by an inner and an outer support ring, which give increased strength and rigidity. These support rings also facilitate removal of the nozzle vanes as a unit; otherwise, the vanes could fall out as the shrouds were removed.

Another method of thermal expansion construction is to fit the vanes into inner and outer shrouds; however, in this method the vanes are welded or riveted into position. (See figure 1–52(B).) Some means must be provided to allow for thermal expansion; therefore, either the inner or the outer shroud ring is cut into segments. These saw cuts dividing the segments will allow sufficient expansion to prevent stress and warping of the vanes.

The rotor element of the turbine section consists essentially of a shaft and a wheel. (See figure 1–51.)

The turbine wheel is a dynamically balanced unit consisting of blades attached to a rotating disk. The disk, in turn, is attached to the main power-transmitting shaft of the engine. The jet gases leaving the turbine nozzle vanes act on the blades of the turbine wheel, causing the assembly to rotate at a very high rate of speed. The high rotational speed imposes severe centrifugal loads on the turbine wheel, and at the same time the elevated temperatures result in a lowering of the strength of the material. Consequently, the engine speed and temperature must be controlled to keep turbine operation within safe limits.

The turbine disk is referred to as such when in an unbladed form. When the turbine blades are installed, the disk then becomes the turbine wheel. The disk acts as an anchoring component for the turbine blades. Since the disk is bolted or welded to the shaft, the blades can transmit to the rotor shaft the energy they extract from the exhaust gases.

The disk rim is exposed to the hot gases passing through the blades and absorbs considerable heat from these gases. In addition, the rim also absorbs heat from the turbine buckets (blades) by conduction. Hence, disk rim temperatures normally are high and well above the temperatures of the more remote inner portion of the disk. As a result of these temperature gradients, thermal stresses are added to the rotational stresses.

There are various means to relieve, at least partially, the aforementioned stresses. One such means is to bleed cooling air back onto the face of the disk.

Another method of relieving the thermal stresses of the disk is incidental to blade installation. A series of grooves or notches, conforming to the blade root design, are broached in the rim of the disk. These grooves attach the turbine blades to the disk, and at the same time space is provided by the notches for thermal expansion of the disk. Sufficient clearance exists between the blade root and the notch to permit movement of the turbine blade when the disk is cold. During engine operation, expansion of the disk decreases the clearance. This causes the bucket root to fit tightly in the disk rim.

The turbine shaft, illustrated in figure 1–51, is usually fabricated from alloy steel. It must be capable of absorbing the high torque loads that are exerted when a heavy axial-flow compressor is started.

The methods of connecting the shaft to the turbine disk vary. In one method, the shaft is welded to the disk, which has a butt or protrusion provided for the joint. Another method is by bolting. This method requires that the shaft have a hub which matches a machined surface on the disk face. The bolts then are inserted through holes in the shaft hub and anchored in tapped holes in the disk. Of the two methods, the latter is more common.

The turbine shaft must have some means for attachment to the compressor rotor hub. This is usually accomplished by a spline cut on the forward end of the shaft. The spline fits into a coupling device between the compressor and turbine shafts. If a coupling is not used, the splined end of the turbine shaft may fit into a splined recess in the compressor rotor hub. This splined coupling arrangement is used almost exclusively with centrifugal compressor engines, while the axial compressor engine may use either of these described methods.

There are various ways of attaching turbine blades or buckets, some similar to compressor blade attachment. The most satisfactory method used is the fir-tree design shown in figure 1–53.

The blades are retained in their respective grooves by a variety of methods; some of the more common ones are peening, welding, locktabs, and riveting. Figure 1–54 shows a typical turbine wheel using rivets for blade retention.

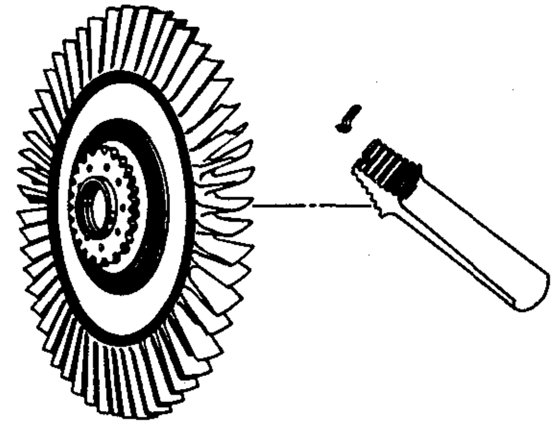


FIGURE 1–53. Turbine blade with fir-tree design and lock-tab method of blade retention.

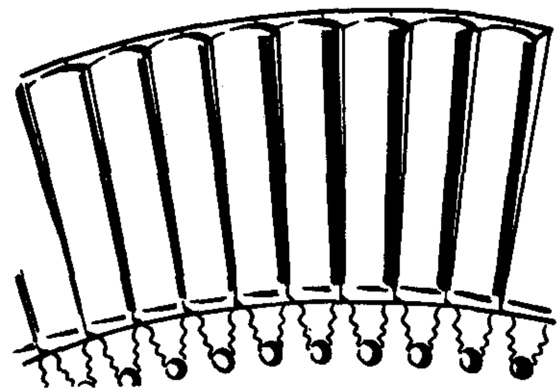


FIGURE 1–54. Riveting method of turbine blade retention.

The peening method of blade retention is used frequently in various ways. One of the most common applications of peening requires a small notch to be ground in the edge of the blade fir-tree root prior to the blade installation. After the blade is inserted into the disk, the notch is filled by the disk metal, which is "flowed" into it by a small punch-mark made in the disk adjacent to the notch. The tool used for this job is similar to a center punch.

Another method of blade retention is to construct the root of the blade so that it will contain all the elements necessary for its retention. This method, illustrated in figure 1–55, shows that the blade root has a stop made on one end of the root so that the blade can be inserted and removed in one direction only, while on the opposite end is a tang. This tang is bent to secure the blade in the disk.

Turbine blades may be either forged or cast, depending on the composition of the alloys. Most

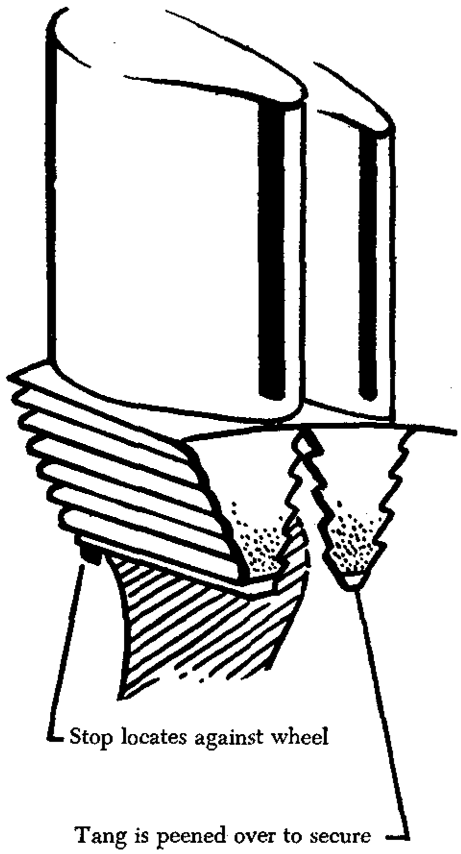


FIGURE 1–55. Turbine bucket, featuring tang method of blade retention.

blades are precision-cast and finish-ground to the desired shape.

Most turbines are open at the outer perimeter of the blades; however, a second type called the shrouded turbine is sometimes used. The shrouded turbine blades, in effect, form a band around the outer perimeter of the turbine wheel. This improves efficiency and vibration characteristics, and permits lighter stage weights; on the other hand, it limits turbine speed and requires more blades (see figure 1–56).

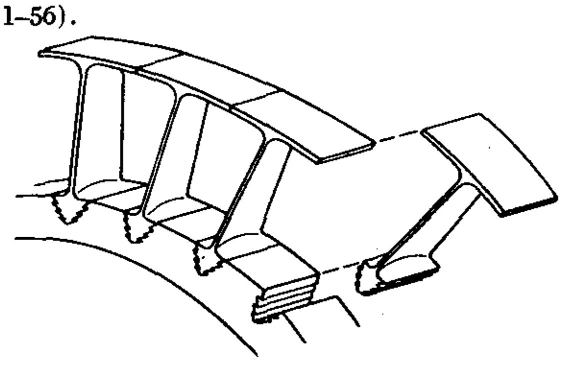


FIGURE 1–56. Shrouded turbine blades.

In turbine rotor construction, it occasionally becomes necessary to utilize turbines of more than one stage. A single turbine wheel often cannot absorb enough power from the exhaust gases to drive the components dependent on the turbine for rotative power, and thus, it is necessary to add additional turbine stages.

A turbine stage consists of a row of stationary vanes or nozzles, followed by a row of rotating blades. In some models of turboprop engine, as many as five turbine stages have been utilized successfully. It should be remembered that, regardless of the number of wheels necessary for driving engine components, there is always a turbine nozzle preceding each wheel.

As was brought out in the preceding discussion of turbine stages, the occasional use of more than one turbine wheel is warranted in cases of heavy rotational loads. It should also be pointed out that the same loads that necessitate multiple-stage turbines often make it advantageous to incorporate multiple compressor rotors.

In the single-stage rotor turbine (figure 1–57), the power is developed by one rotor, and all engine-driven parts are driven by this single wheel. This arrangement is used on engines where the need for low weight and compactness predominates.

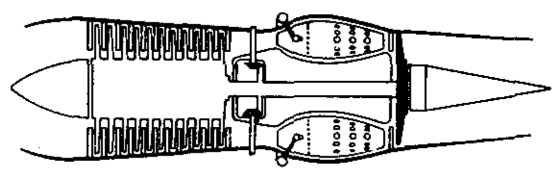


FIGURE 1–57. Single-stage rotor turbine.

In the multiple-rotor turbine the power is developed by two or more rotors. It is possible for each turbine rotor to drive a separate part of the engine. For example, a triple-rotor turbine can be so arranged that the first turbine drives the rear half of the compressor and the accessories, the second turbine drives the front half of the compressor, and the third turbine furnishes power to a propeller. (See figure 1–58.)

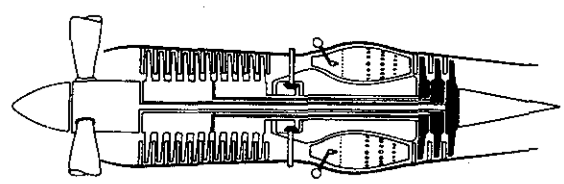


FIGURE 1–58. Multiple-rotor turbine.

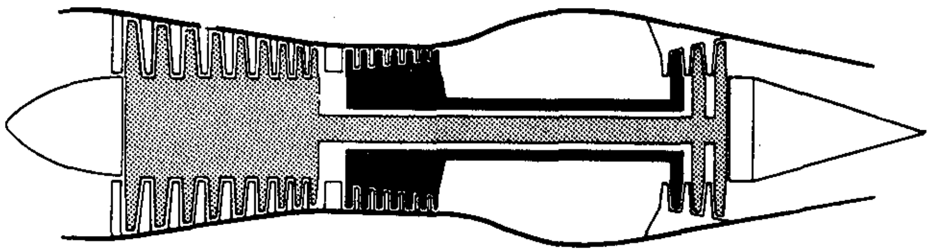


FIGURE 1–59. Dual-rotor turbine for split-spool compressor.

The turbine rotor arrangement for a dual-rotor turbine, such as required for a split-spool compressor, is similar to the arrangement in figure 1–58. The difference is that where the third turbine is used for a propeller in figure 1–58, it would be joined with the second turbine to make a two-stage turbine for driving the front compressor. This arrangement is shown in figure 1–59.

The remaining element to be discussed concerning turbine familiarization is the turbine casing or housing. The turbine casing encloses the turbine wheel and the nozzle vane assembly, and at the same time gives either direct or indirect support to the stator elements of the turbine section. It always has flanges provided front and rear for bolting the assembly to the combustion chamber housing and the exhaust cone assembly, respectively. A turbine casing is illustrated in figure 1–60.

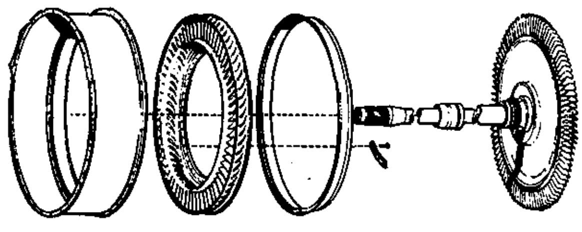


FIGURE 1–60. Turbine casing assembly.

Exhaust Section

The exhaust section of the turbojet engine is made up of several components, each of which has its individual functions. Although the components have individual purposes, they also have one common function: They must direct the flow of hot gases rearward in such a manner as to prevent turbulence and at the same time impart a high final or exit velocity to the gases.

In performing the various functions, each of the components affects the flow of gases in different ways as described in the following paragraphs.

The exhaust section is located directly behind the turbine section and ends when the gases are ejected at the rear in the form of a high-velocity jet. The components of the exhaust section include the exhaust cone, tailpipe (if required), and the exhaust or jet nozzle. Each component is discussed individually.

The exhaust cone collects the exhaust gases discharged from the turbine buckets and gradually converts them into a solid jet. In performing this, the velocity of the gases is decreased slightly and the pressure increased. This is due to the diverging passage between the outer duct and the inner cone; that is, the annular area between the two units increases rearward.

The exhaust cone assembly consists of an outer shell or duct, an inner cone, three or four radial hollow struts or fins, and the necessary number of tie rods to aid the struts in supporting the inner cone from the outer duct.

The outer shell or duct is usually made of stainless steel and is attached to the rear flange of the turbine case. This element collects the exhaust gases and delivers them either directly or via a tailpipe to the jet nozzle, depending, of course, on whether or not a tailpipe is required. In some engine installations a tailpipe is not needed. For instance, when the engine is installed in nacelles or pods, a short tailpipe is all that is required, in which case the exhaust duct and exhaust nozzle will suffice. The duct must be constructed to include such features as a predetermined number of thermocouple bosses for installing tailpipe temperature thermocouples, and there must also be the insertion holes for the supporting tie rods. In some cases, tie rods are not used for supporting the inner cone. If such is the case, the hollow struts provide the sole support of the inner cone, the struts being spot-welded in position to the inside surface of the duct and to the inner cone, respectively (see figure 1–61).

The radial struts actually have a twofold function. They not only support the inner cone in the exhaust

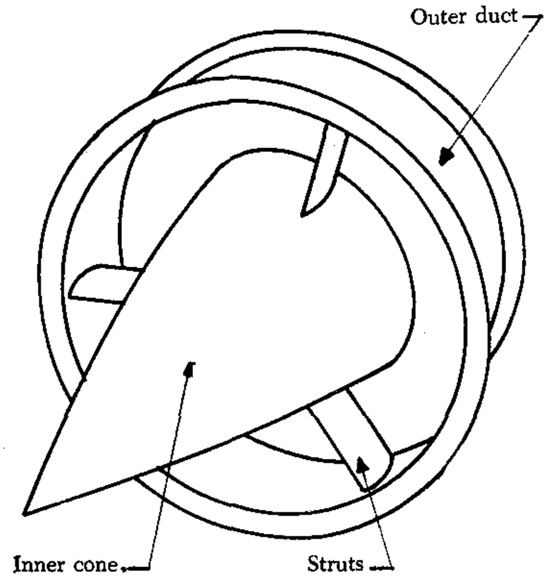


FIGURE 1–61. Exhaust collector with welded support struts.

duct, but they also perform the important function of straightening the swirling exhaust gases that would otherwise leave the turbine at an angle of approximately 45°.

The centrally located inner cone fits rather closely against the rear face of the turbine disk, preventing turbulence of the gases as they leave the turbine wheel. The cone is supported by the radial struts. In some configurations a small hole is located in the exit tip of the cone. This hole allows cooling air to be circulated from the aft end of the cone, where the pressure of the gases is relatively high, into the interior of the cone and consequently against the face of the turbine wheel. The flow of air is positive, since the air pressure at the turbine wheel is relatively low due to rotation of the wheel; thus air circulation is assured. The gases used for cooling the turbine wheel will return to the main path of flow by passing through the clearance between the turbine disk and the inner cone.

The exhaust cone assembly is the terminating component of the basic engine. The remaining components (the tailpipe and jet nozzle) are usually considered airframe components.

The tailpipe is used primarily to pipe the exhaust gases out of the airframe. The use of a tailpipe imposes a penalty on the operating efficiency of the engine in the form of heat and duct (friction) losses. These losses materially affect the final velocity of the exhaust gases and, hence, the thrust.

The tailpipe terminates in a jet nozzle located just forward of the end of the fuselage. Most installations use a single direct exhaust as opposed to a dual exit exhaust to obtain the advantages of low-weight, simplicity, and minimum duct losses (see figure 1–62).

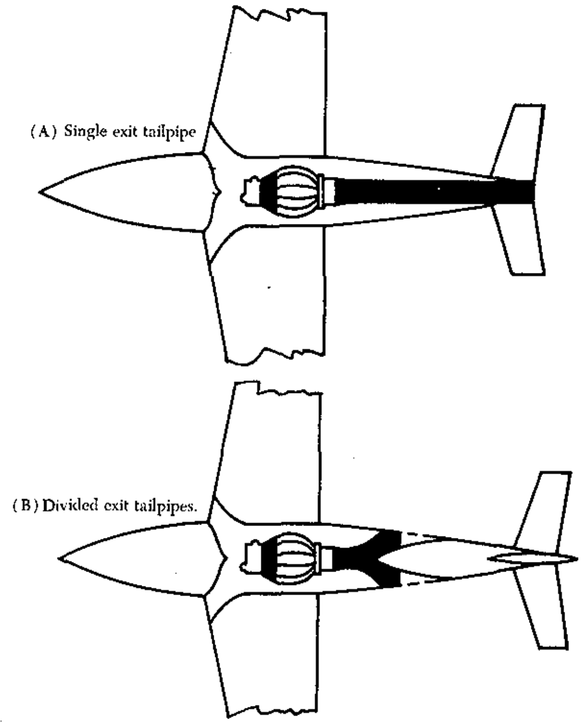


FIGURE 1–62. (A) Single exit tailpipe, (B) Divided exit tailpipes.

The tailpipe is usually constructed so that it is semiflexible. Again the necessity for this feature is dependent on its length. On extremely long tailpipes, a bellows arrangement is incorporated in its construction, allowing movement both in installation and maintenance, and in thermal expansion. This eliminates stress and warping which would otherwise be present.

The heat radiation from the exhaust cone and tailpipe could damage the airframe components surrounding these units. For this reason, some means of insulation had to be devised. There are several suitable methods of protecting the fuselage structure; two of the most common are insulation blankets and shrouds.

The insulation blanket, illustrated in figures 1–63 and 1–64, consists of several layers of aluminum foil, each separated by a layer of fiber glass or some other suitable material. Although these blankets protect the fuselage from heat radiation, they are primarily used to reduce heat losses from the exhaust system. The reduction of heat losses improves engine performance. A typical insulation blanket

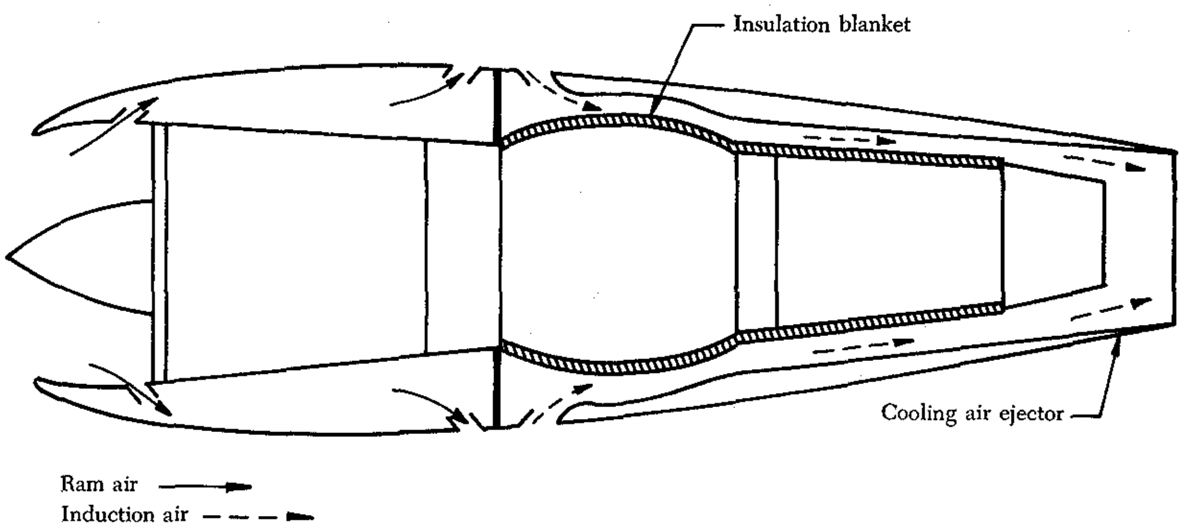


FIGURE 1–63. Exhaust system insulation blanket.

and the temperatures in the exhaust section are shown in figure 1–64. This blanket contains fiber glass as the low conductance material and aluminum foil as the radiation shield. The blanket should be suitably covered to prevent its becoming soaked with oil.

The heat shroud consists of a stainless steel envelope enclosing the exhaust system (see figure 1–65).

The exhaust or jet nozzle imparts to the exhaust gases the all-important final boost in velocity. The jet nozzle, like the tailpipe, is not included as part of the basic powerplant, but is supplied as a component of the airframe. The nozzle is attached to the rear of the tailpipe, if a tailpipe is required, or to the rear flange of the exhaust duct if a tailpipe is not necessary.

There are two types of jet nozzle design. They

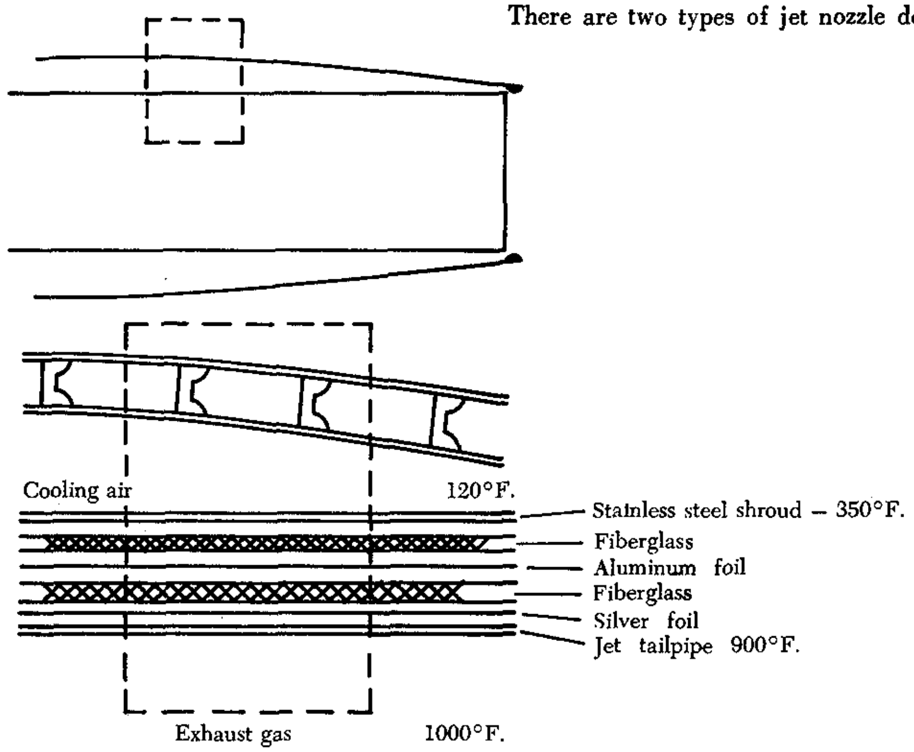


FIGURE 1–64. Insulation blanket with the temperatures which would be obtained at the various locations shown.

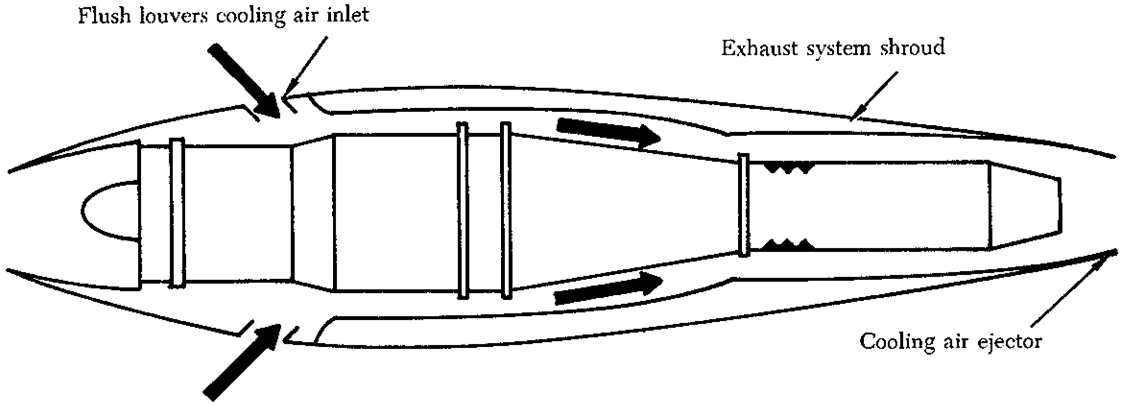


FIGURE 1–65. Exhaust system shroud.

are the converging design, for subsonic gas velocities, and the converging-diverging design for supersonic gas velocities. These are discussed in greater detail in Chapter 2, "Induction and Exhaust Systems."

The jet nozzle opening may be either fixed-area or variable-area. The fixed-area is the simpler of the two jet nozzles. Since there are no moving parts, any adjustment in nozzle area must be made mechanically.

Adjustments in nozzle area are sometimes necessary because the size of the exit orifice will directly affect the operating temperature of the engine. When necessary, a fixed-area nozzle can be adjusted in one of several ways. One method of changing nozzle area is to use inserts, which fit inside the nozzle and are held in place by screws.

The inserts are of varying curvatures and sizes. The different size inserts allow a change in nozzle area to be made in varying increments. Thus, through experience a mechanic can run the engine at maximum speed with one combination of inserts, check the temperature, and substitute another combination to make up a temperature deficiency or remedy an excess temperature situation.

MAJOR SUBASSEMBLIES

The assemblies included in the discussion that follows are integral parts of, or a combination of, the components which comprise the major sections of a turbojet engine.

Diffuser

The diffuser is the divergent section of the engine. It has the all-important function of changing high-velocity compressor discharge air to static pressure. This prepares the air for entry into the burner cans at low velocity so that it will burn with a flame that will not blow out.

Air Adapters

The air adapters of the centrifugal compressor are illustrated in figure 1–37 along with the diffuser. The purpose of the air outlet ducts is to deliver air from the diffuser to the individual can-type combustion chambers. In some instances, fuel nozzles or igniter plugs are also mounted in the air outlet duct.

Engine Rotor

The engine rotor is a combination of the compressor and turbine rotors on a common shaft. The common shaft is provided by joining the turbine and compressor shafts by a suitable method. The engine rotor is supported by bearings, which are seated in suitable bearing housings.

Main Bearings

The main bearings have the critical function of supporting the main engine rotor. The number of bearings necessary for proper engine support will, for the most part, be decided by the length and weight of the engine rotor. The length and weight are directly affected by the type of compressor used in the engine. Naturally, a split-spool axial compressor will require more support than a centrifugal compressor.

Probably the minimum number of bearings required would be three, while some of the later models of split-spool axial compressor engines require six or more.

The gas turbine rotors are usually supported by either ball or roller bearings. Hydrodynamic or slipper-type bearings are receiving some attention for use on turbine powerplants where operating rotor speeds approach 45,000 r.p.m. and where excessive bearing loads during flight are anticipated. (See figure 1–66.) In general, the ball or roller anti-

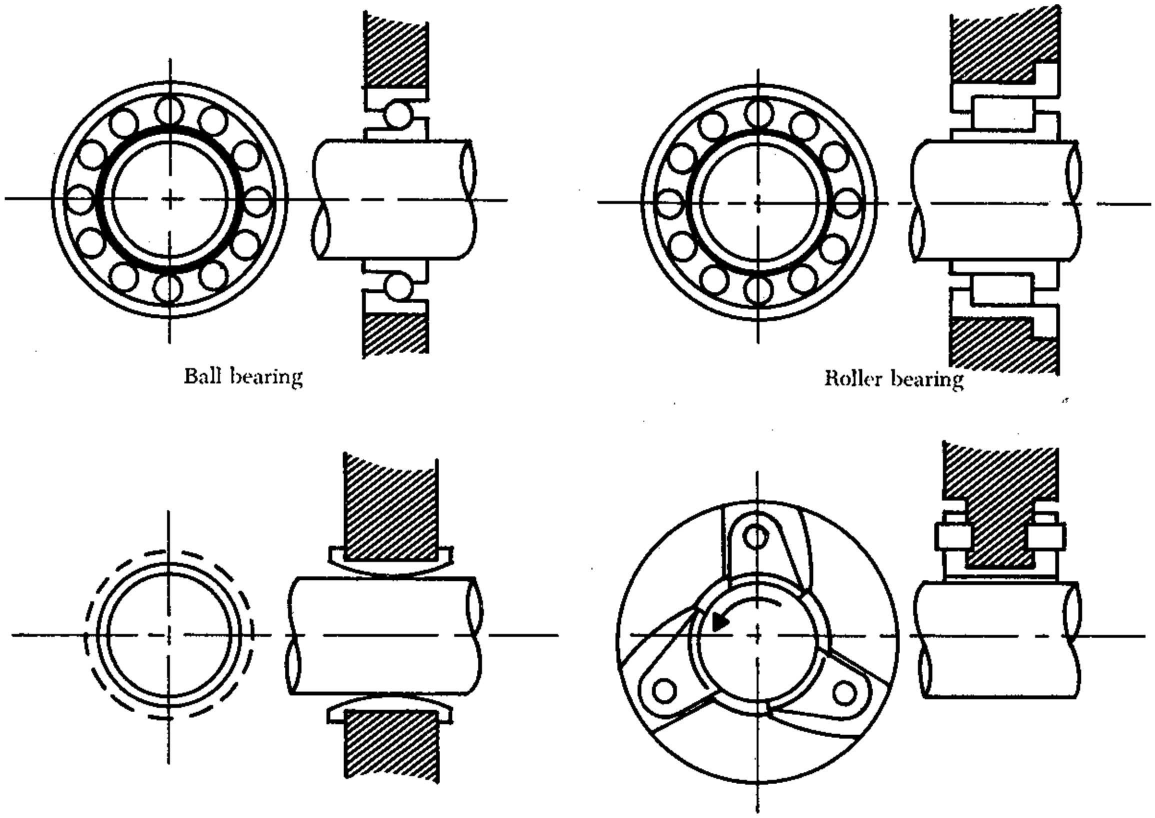


FIGURE 1–66. Type of main bearings used for gas turbine rotor support.

friction bearings are preferred largely on the basis that they:

(1) Offer little rotational resistance.
(2) Facilitate precision alignment of rotating elements.
(3) Are relatively inexpensive.
(4) Are easily replaced.
(5) Withstand high momentary overloads.
(6) Are simple to cool, lubricate, and maintain.
(7) Accommodate both radial and axial lods.
(8) Are relatively resistant to elevated temperatures.

The main disadvantages are their vulnerability to foreign matter and tendency to fail without appreciable warning.

Usually the ball bearings are positioned on the compressor or turbine shaft so that they can absorb any axial (thrust) loads or radial loads. Because the roller bearings present a larger working surface, they are better equipped to support radial loads than thrust loads. Therefore, they are used primarily for this purpose.

A typical ball or roller bearing assembly includes a bearing support housing, which must be strongly constructed and supported in order to carry the radial and axial loads of the rapidly rotating rotor. The bearing housing usually contains oil seals to prevent the oil leaking from its normal path of flow. It also delivers the oil to the bearing for its lubrication, usually through spray nozzles.

The oil seals may be the labyrinth or thread (helical) type. These seals also may be pressurized to minimize oil leaking along the compressor shaft. The labyrinth seal is usually pressurized, but the helical seal depends solely on reverse threading to stop oil leakage. These two types of seals are very similar, differing only in thread size and the fact that the labyrinth seal is pressurized.

Another type of oil seal used on some of the later engines is the carbon seal. These seals are usually spring loaded and are similar in material and application to the carbon brushes used in electrical motors. Carbon seals rest against a surface provided to create a sealed bearing cavity or void; thus, the oil is prevented from leaking out along the shaft into the compressor airflow or the turbine section.

Figure 1–67 illustrates an oil seal of the spring-loaded carbon type.

The ball or roller bearing is fitted into the bearing housing and may have a self-aligning feature. If a bearing is self-aligning, it is usually seated in a spherical ring, thus allowing the shaft a certain amount of radial movement without transmitting stress to the bearing inner race.

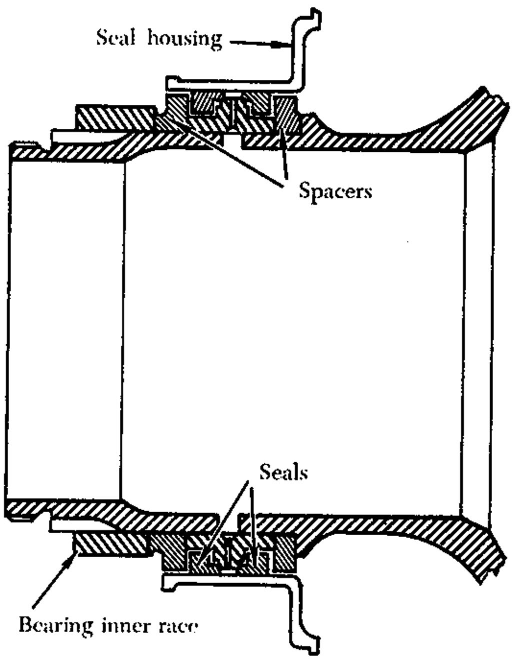


FIGURE 1–67. Carbon oil seal.

The bearing surface is usually provided by a machined journal on the appropriate shaft. The bearing is usually locked in position by a steel snapring, or other suitable locking device.

The rotor shaft also provides the matching surface for the oil seals in the bearing housing. These machined surfaces are called lands and fit in rather close to the oil seal, although not actually touching. If rubbing occurs, eventual wear and leakage will result.

TURBOPROP ENGINES

The turboprop (turbo-propeller) engine is a combination of a gas turbine and a propeller. Turboprops are basically similar to turbojet engines in that both have a compressor, combustion chamber(s), turbine, and a jet nozzle, all of which operate in the same manner on both engines. However, the difference is that the turbine in the turboprop engine usually has more stages than that in the turbojet engine. In addition to operating the compressor and accessories, the turboprop turbine transmits increased power forward, through a shaft and a gear train, to drive the propeller. The increased power is generated by the exhaust gases passing through additional stages of the turbine.

Refer to figure 1–58, which shows a multiple-rotor turbine with coaxial shafts for independent driving of the compressor and propeller. Although there are three turbines utilized in this illustration, as many as five turbine stages have been used for driving the two rotor elements, propeller, and accessories.

The exhaust gases also contribute to engine power output through jet reaction, although the amount of energy available for jet thrust is considerably reduced.

Since the basic components of the turbojet and the turboprop engines differ only slightly in design features, it should be fairly simple to apply acquired knowledge of the turbojet to the turboprop.

The typical turboprop engine can be broken down into assemblies as follows:

(1) The power section assembly, which contains the usual major components of gas turbine engines (compressor, combustion chamber, turbine, and exhaust sections).

(2) The reduction gear or gearbox assembly which contains those sections peculiar to turboprop configurations.

(3) The torquemeter assembly, which transmits the torque from the engine to the gearbox of the reduction section.

(4) The accessory drive housing assembly.

These assemblies are illustrated in figure 1–68.

The turboprop engine can be used in many different configurations. It is often used in transport aircraft, but can be adapted for use in single-engine aircraft.

TURBOSHAFT ENGINES

A gas turbine engine that delivers power through a shaft to operate something other than a propeller is referred to as a turboshaft engine. Turboshaft engines are similar to turboprop engines. The power takeoff may be coupled directly to the engine turbine, or the shaft may be driven by a turbine of its own (free turbine) located in the exhaust stream. The free turbine rotates independently. This principle is used extensively in current production turboshaft engines. The turboshaft engine is currently being used to power helicopters.

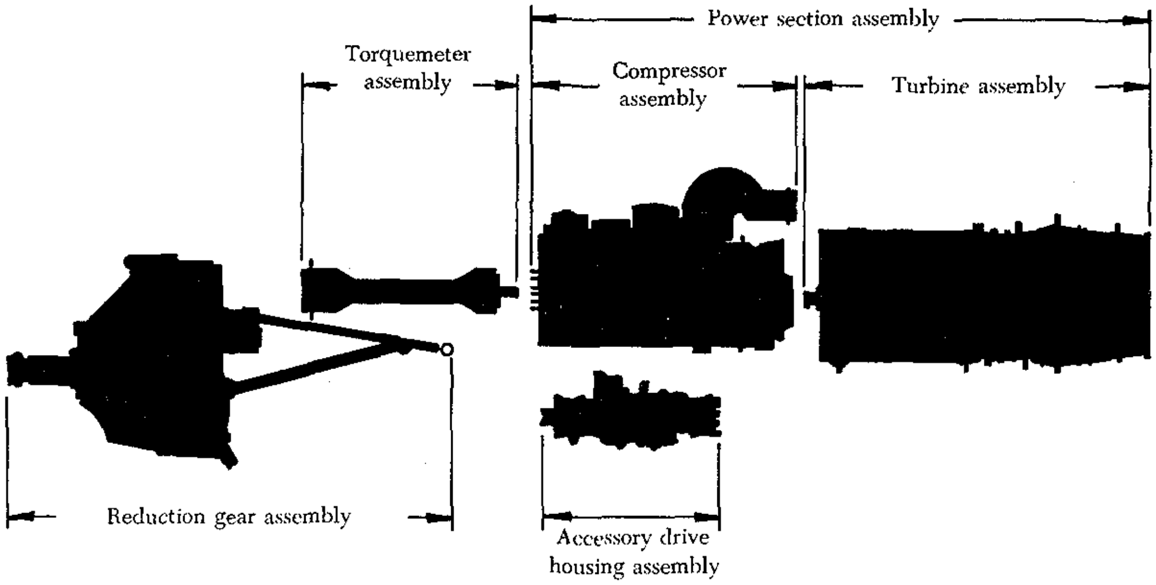


FIGURE 1–68. Turboprop engine major components.

TURBOFAN ENGINES

The turbofan gas turbine engine (figure 1–69) is, in principle, the same as a turboprop, except that the propeller is replaced by a duct-enclosed axial-flow fan. The fan can be a part of the first-stage compressor blades or can be mounted as a separate set of fan blades. The blades can be mounted forward of the compressor, or aft of the turbine wheel.

The general principle of the fan engine is to convert more of the fuel energy into pressure. With more of the energy converted to pressure, a greater product of pressure times area can be achieved. One of the big advantages is that the turbofan produces this additional thrust without increasing fuel flow. The end result is savings in fuel with the consequent increase in range.

Because more of the fuel energy is turned into pressure in the turbofan engine, another stage must be added in the turbine to provide the power to drive the fan, and thus increase the expansion through the turbine. This means that there will be less energy left over and less pressure in back of the turbine. Also, the jet nozzle has to be larger in area. The end result is that the main engine does not develop as much jet nozzle thrust as a straight turbojet engine.

The fan more than makes up for the dropoff in thrust of the main engine. Depending on the fan design, it will produce somewhere around 50% of the turbofan engine's total thrust. In an 18,000 lb. thrust engine about 9,000 lbs. will be developed by the fan and the remaining 9,000 lbs. by the main engine. The same basic turbojet engine without a fan will develop about 12,000 lbs. of thrust.

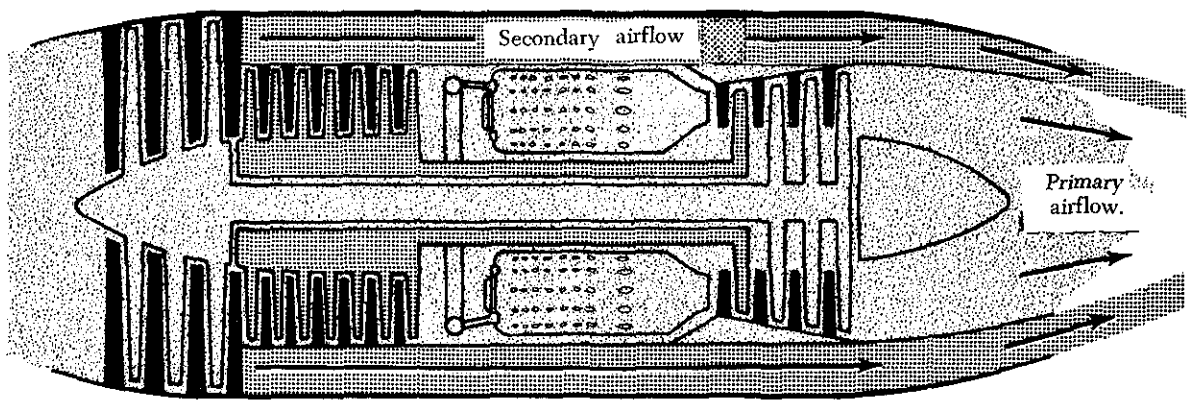


FIGURE 1–69. Forward turbofan engine.

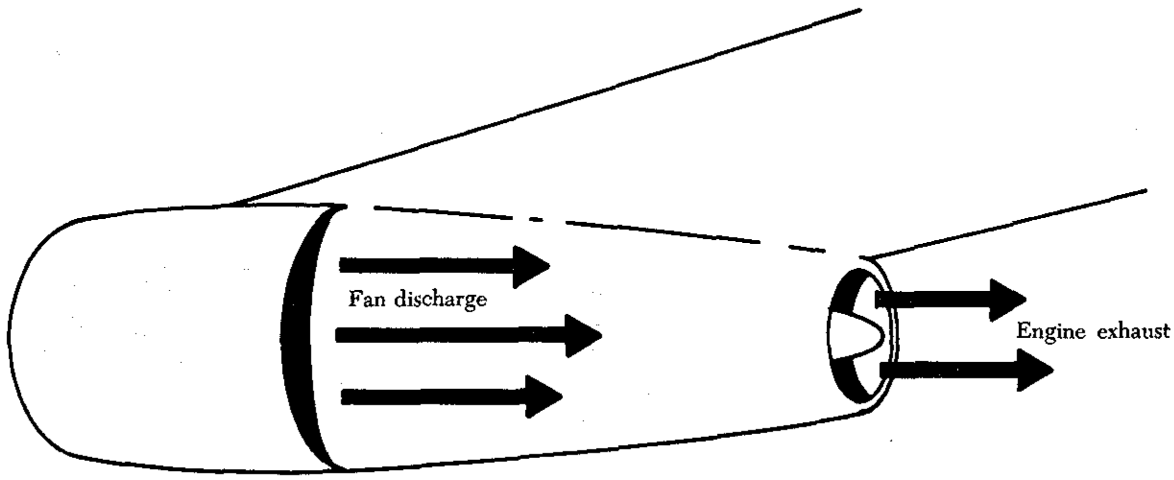


FIGURE 1–70. Forward-fan turbofan engine installation.

Two different duct designs are used with forward-fan engines. The air leaving the fan can be ducted overboard (figure 1–70), or it can be ducted along the outer case of the basic engine to be discharged through the jet nozzle. The fan air is either mixed with the exhaust gases before it is discharged or it passes directly to the atmosphere without prior mixing.

Turbofans, sometimes called fanjets, are becoming the most widely used gas turbine engine. The turbofan is a compromise between the good operating efficiency and high-thrust capability of a turboprop and the high-speed, high-altitude capability of a turbojet.

TURBINE ENGINE OPERATING PRINCIPLES

The principle used by a turbojet engine as it provides force to move an airplane is based on Newton's law of momentum. This law shows that a force is required to accelerate a mass; therefore, if the engine accelerates a mass of air, it will apply a force on the aircraft. The propeller and turbojet engines are very closely related. The propeller generates thrust by giving a relatively small acceleration to a large quantity of air. The turbojet engine achieves thrust by imparting greater acceleration to a smaller quantity of air.

The mass of air is accelerated within the engine by the use of a continuous-flow cycle. Ambient air enters the inlet diffuser where it is subjected to changes in temperature, pressure, and velocity due to ram effect. The compressor then increases pressure and temperature of the air mechanically. The air continues at constant pressure to the burner section where its temperature is increased by combustion of fuel. The energy is taken from the hot gas by expanding through a turbine which drives the compressor, and by expanding through a tailpipe designed to discharge the exhaust gas at high velocity to produce thrust.

The high-velocity jet from a turbojet engine may be considered a continuous recoil, imparting force against the aircraft in which it is installed, thereby producing thrust. The formula for thrust can be derived from Newton's second law, which states that force is proportional to the product of mass and acceleration. This law is expressed in the formula:

$$F = M \times A$$

where;

F = Force in pounds.
M = Mass in slugs.
A = Acceleration in ft. per sec. per sec.

In the above formula "mass" is similar to "weight," but it is actually a different quantity. Mass refers to the quantity of matter, while weight refers to the pull of gravity on that quantity of matter. At sea level under standard conditions, 1 lb. of mass will have a weight of 1 lb.

To calculate the acceleration of a given mass, the gravitational constant is used as a unit of comparison. The force of gravity is 32.2 ft./sec.^{2} (or feet per second squared). This means that a free-falling 1-lb. object will accelerate at the rate of 32.2 feet per second each second that gravity acts on it. Since the object mass weighs 1 lb., which is also the actual force imparted to it by gravity, we can assume that a force of 1 lb. will accelerate a 1-lb. object at the rate of 32.2 ft./sec.^{2}.

Also, a force of 10 lbs. will accelerate a mass of 10 lbs. at the rate of 32.2 ft./sec.^{2}. This is assuming there is no friction or other resistance to overcome. It is now apparent that the ratio of the force (in pounds) is to the mass (in pounds) as the acceleration in ft./sec.^{2} is to 32.2. Using M to represent the mass in pounds, the formula may be expressed thus:

$$\frac{F}{M}=\frac{A}{g}$$

or,

$$F=\frac{MA}{g}$$

where;

$F =$ Force.
$M =$ Mass.
$A =$ Acceleration.
$g =$ gravity.

In any formula involving work, the time factor must be considered. It is convenient to have all time factors in equivalent units; i.e., seconds, minutes, or hours. In calculating jet thrust, the term "pounds of air per second" is convenient, since the time factor is the same as the time in the force of gravity, namely, seconds.

THRUST

Using the foregoing formula, compute the force necessary to accelerate a mass of 50 lbs., 100 ft./sec.^{2}, as follows:

$$F=\frac{50 \text{ lb. x } 100 \text{ ft./sec.}^2}{32.2 \text{ ft./sec.}^2}$$

$$F=\frac{50 \times 100}{32.2}$$

$$F = 155 \text{ lb.}$$

This illustrates that if the velocity of 50 lbs. of mass per sec. is increased by 100 ft./sec.^{2}, the resulting thrust is 155 lbs.

Since the turbojet engine accelerates a mass of air, the following formula can be used to determine jet **thrust:**

$$F=\frac{M_s\,(V_2-V_1)}{g}$$

where;

F = Force in lbs.
M_s = Mass flow in lbs./sec.
V_1 = Inlet velocity.
V_2 = Jet velocity (exhaust).
$V_2 - V_1$ = Change in velocity; difference between inlet velocity and jet velocity.
g = Acceleration of gravity, or 32.2 ft./sec.^{2}.

As an example, to use the formula for changing the velocity of 100 lbs. of mass airflow per sec. from 600 ft./sec. to 800 ft./sec., the formula can be applied as follows:

$$F=\frac{100\ (800-600)}{32.2}$$

$$F=\frac{20{,}000}{32.2}$$

$$F = 621 \text{ pounds.}$$

As shown by the formula, if the mass airflow per second and the difference in the velocity of the air from the intake to the exhaust is known, it is easy to compute the force necessary to produce the change in the velocity. Therefore, the jet thrust of the engine must be equal to the force required to accelerate the airmass through the engine. Then, by using the symbol "T" for thrust pounds, the formula becomes:

$$T=\frac{M_s\,(V_2-V_1)}{g}\,.$$

It is easy to see from this formula that the thrust of a gas turbine engine can be increased by two methods: first, by increasing the mass flow of air through the engine, and second, by increasing the jet velocity.

If the velocity of the turbojet engine remains constant with respect to the aircraft, the jet thrust will decrease if the speed of the aircraft is increased. This is because V_1 will increase in value. This does not present a serious problem, however, because as the aircraft speed increases, more air enters the engine, and jet velocity increases. The resultant net thrust is almost constant with increased airspeed.

The Brayton cycle is the name given to the thermodynamic cycle of a gas turbine engine to produce thrust. This is a varying volume constant-pressure cycle of events and is commonly called the constant-pressure cycle. A more recent term is continuous combustion cycle.

The four continuous and constant events are the intake, compression, expansion (includes power), and exhaust. These cycles will be discussed as they apply to a gas turbine engine.

In the intake cycle, air enters at ambient pressure and a constant volume. It leaves the intake at an increased pressure and a decrease in volume. At the compressor section, air is received from the intake at an increased pressure, slightly above ambient, and a slight decrease in volume. Air enters the compressor where it is compressed. It leaves the compressor with a large increase in pressure and decrease in volume. This is caused by the mechanical action of the compressor. The next step, the expansion, takes place in the combustion chamber by burning fuel which expands the air by heat. The pressure remains relatively constant, but a marked increase in volume takes place. The expanding gases move rearward through the turbine assembly and are converted from velocity energy to mechanical energy by the turbine.

The exhaust section, which is a convergent duct, converts the expanding volume and decreasing pressure of the gases to a final high velocity. The force created inside the jet engine to keep this cycle continuous has an equal and opposite reaction (thrust) to move the aircraft forward.

Bernoulli's principle (whenever a stream of any fluid has its velocity increased at a given point, the pressure of the stream at that point is less than the rest of the stream) is applied to the jet engine through the design of the air ducts. The two types of ducts are the convergent and the divergent.

The convergent duct increases velocity and decreases pressure. The divergent duct decreases velocity and increases pressure. The convergent principle is usually used for the tailpipe and exhaust nozzle. The divergent principle is used in the compressor where the air is slowing and pressurizing.

GAS TURBINE ENGINE PERFORMANCE

Thermal efficiency is a prime factor in gas turbine performance. It is the ratio of net work produced by the engine to the chemical energy supplied in the form of fuel.

The three most important factors affecting the thermal efficiency are turbine inlet temperature, compression ratio, and the component efficiencies of the compressor and turbine. Other factors that affect thermal efficiency are compressor inlet temperature and burner efficiency.

Figure 1–71 shows the effect that changing compression ratio has on thermal efficiency when compressor inlet temperature and the component efficiencies of the compressor and turbine remain constant.

The effect that compressor and turbine component efficiencies have on thermal efficiency when turbine and compressor inlet temperatures remain constant is shown in figure 1–72. In actual operation, the turbine engine tailpipe temperature varies directly with turbine inlet temperature at a constant

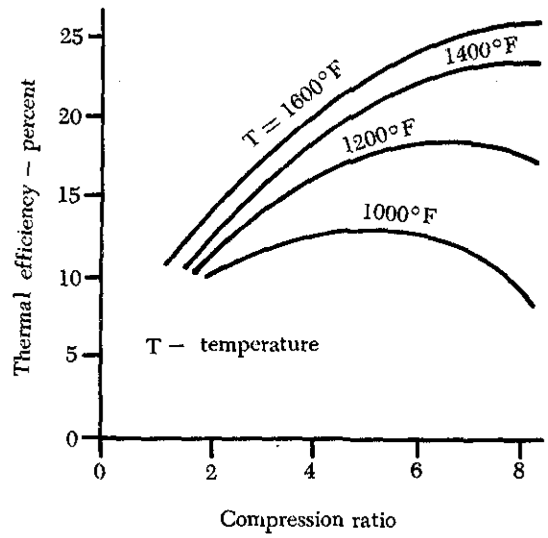


FIGURE 1–71. The effect of compression ratio on thermal efficiency.

compression ratio. R.P.M. is a direct measure of compression ratio; therefore, at constant r.p.m. maximum thermal efficiency can be obtained by maintaining the highest possible tailpipe temperature. Since engine life is greatly reduced at a high turbine inlet temperature, the operator should not exceed the tailpipe temperatures specified for continuous operation. Figure 1–73 illustrates the effect of turbine inlet temperature on turbine bucket life.

In the previous discussion, it has been assumed that the state of the air at the inlet to the com-

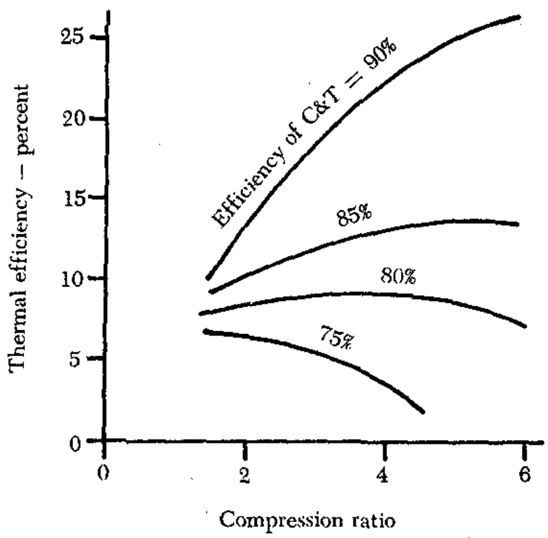


FIGURE 1–72. Turbine and compressor efficiency vs. thermal efficiency.

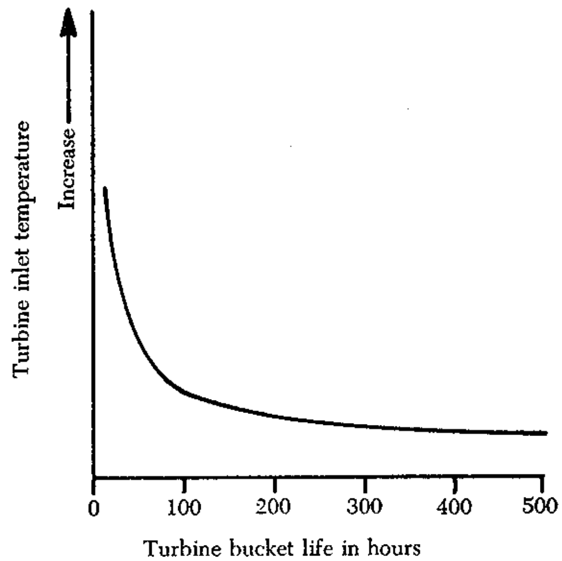


FIGURE 1–73. Effect of turbine inlet temperature on turbine bucket life.

pressor remained constant. Since the turbojet engine is a practical application of a turbine engine, it becomes necessary to analyze the effect of varying inlet conditions on the power produced. The three principal variables that affect inlet conditions are the speed of the aircraft, the altitude of the aircraft, and the ambient temperature. To make the analysis simpler, the combination of these three variables can be represented by a single variable, called "stagnation density."

The power produced by a turbine engine is proportional to the stagnation density at the inlet. The next three illustrations show how changing the density by varying altitudes, airspeed, and outside air temperature affects the power level of the engine.

Figure 1–74 shows that the thrust output improves rapidly with a reduction in OAT (outside air temperature) at constant altitude, r.p.m., and airspeed. This increase occurs partly because the energy required per pound of airflow to drive the compressor varies directly with the temperature, thus leaving more energy to develop thrust. In addition, the thrust output will increase since the air at reduced temperature has an increased density. The increase in density causes the mass flow through the engine to increase.

The altitude effect on thrust, as shown in figure 1–75, can also be discussed as a density and temperature effect. In this case, an increase in altitude causes a decrease in pressure and temperature. Since the temperature lapse rate is less than the

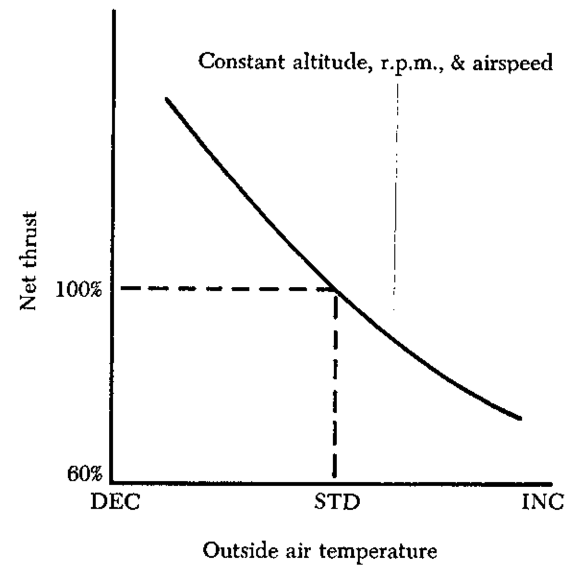


FIGURE 1–74. Effect of OAT on thrust output.

pressure lapse rate as altitude is increased, the density is decreased. Although the decreased temperature increases thrust, the effect of decreased density more than offsets the effect of the colder temperature. The net result of increased altitude is a reduction in the thrust output.

The effect of airspeed on the thrust of a turbojet engine is shown in figure 1–76. To explain the airspeed effect, it is first necessary to understand the effect of airspeed on the factors which combine to produce net thrust. These factors are specific thrust and engine airflow. Specific thrust is the pounds of net thrust developed per pound of airflow

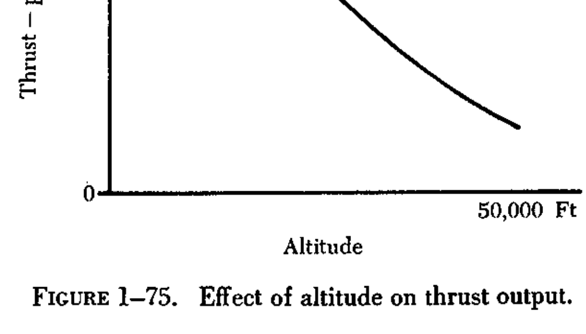


FIGURE 1–75. Effect of altitude on thrust output.

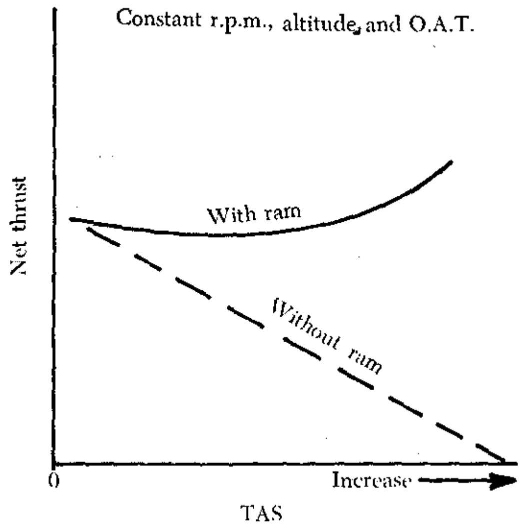


FIGURE 1–76. Effect of airspeed on net thrust.

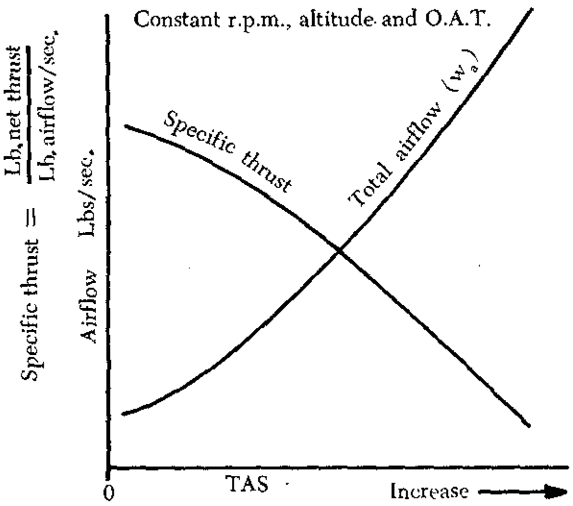


FIGURE 1–77. Effect of airspeed on specific thrust and total engine airflow.

per second. It is the remainder of specific gross thrust minus specific ram drag.

As airspeed is increased, the ram drag increases rapidly. The exhaust jet velocity remains relatively constant; thus the effect of the increase in airspeed results in decreased specific thrust as shown in figure 1–76. In the low-speed range, the specific thrust decreases faster than the airflow increases and causes a decrease in net thrust. As the airspeed increases into the higher range, the airflow increases faster than the specific thrust decreases and causes the net thrust to increase until sonic velocity is reached. The effect of the combination on net thrust is illustrated in figure 1–77.

Ram Recovery

A rise in pressure above existing outside atmospheric pressure at the engine inlet, as a result of the forward velocity of an aircraft, is referred to as ram. Since any ram effect will cause an increase in compressor entrance pressure over atmospheric, the resulting pressure rise will cause an increase in the mass airflow and jet velocity, both of which tend to increase thrust.

Although ram effect increases the engine thrust, the thrust being produced by the engine decreases for a given throttle setting as an aircraft gains airspeed. Therefore, two opposing trends occur when an aircraft's speed is increased. What actually takes place is the net result of these two different effects.

An engine's thrust output temporarily decreases as aircraft speed increases from static, but soon ceases to decrease; towards the high speeds, thrust output begins to increase again.