MINISTRY OF EDUCATION AND SCIENCE,
YOUTH AND SPORT OF UKRAINE
National Aerospace University
“Kharkiv Aviation Institute”

V.Yu. Nezym

GAS TURBINES: DESIGN PROBLEMS

Lecture synopsis

Kharkiv “KhAI” 2012

UDC 621.452.3.01(075.8)
N56

Розглянуто головні проблеми теорії газотурбінних двигунів і установок. Описано процеси і цикли роботи двигунів та їхніх елементів, а також питання перспективних розробок.

Для іноземних студентів, що вивчають курс «Газотурбінні двигуни та установки» англійською мовою.

Reviewers: Dr Tech. Sci., Prof. V.I. Gnesin,
Dr Tech. Sci., Prof. S.V. Yershov

Nezym, V.Yu.
N56 Gas Turbines: Design Problems: lecture synopsis in English / V.Yu. –Nezym.Kh.: Nat. Aerospace Univ “Kh. Aviation Inst”, 2012.–84 pp.

Lecture synopsis covers fundamental problems of the theory of gas turbine engines and power plants. It considers processes and cycles of gas turbines and their components operation including problems of perspective development.

Intended for foreign students studying the course “Gas Turbine Engines and Power Plants” in English.

Ill. 48. Bibl.: 10 titles **UDC 621.452.3.01(075.8)**

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INTRODUCTION

Aircraft designers have always been limited by the efficiency of the available power plants. Their constant plea has been for higher power, less weight, lower frontal area, better cooling characteristics, and lower fuel consumption. These requirements have been met to a certain degree by the designers of reciprocating engines, but the design of the piston engine has been carried to such a point that to obtain further increase in power, more cylinders would have to be added. This would immediately raise more complex problems, which must be solved before an increase in power can be achieved.

The aircraft designers' pleas have been answered with the development of the gas turbine engine. A gas turbine is an engine designed to convert the energy of a fuel into some form of useful power, such as high-speed thrust of a jet or mechanical (shaft) power. A gas turbine consists basically of a gas generator section and a power conversion section. The gas generator section consists of a compressor, combustion chamber, and turbine, the turbine extracting only sufficient power to drive the compressor. This results in a high-temperature, high-pressure gas at the turbine exit. The different types of gas turbines result by adding various inlet and nozzle components to the gas generator.

Today, the gas turbine is widely used. Since the end of World War II, progress in the gas turbine field has been rapid. Development of improved materials, high temperature metals, effective cooling systems and better fuels should expedite further progress in this field. To the present, the gas turbine has been developed into a very reliable, versatile engine with a high power-to-weight ratio. The gas turbine engine's greatest contribution to aviation is that it has lifted all previous limits that were imposed by the reciprocating engine. It is used exclusively to power all new commercial and military airplanes.

The shaft power gas turbine has been used to power boats and trains, has been widely accepted for electric power generators and gas pipeline compressor drives, and is being tested for use in buses and tracks. It has been tested extensively as a power plant for an automobile.

This lecture synopsis provides a guide to the fundamentals of gas turbine process, including the requirements of all major applications, gas turbine components operation and problems.

Topic 1 BASIC THERMO - GAS DYNAMIC EQUATIONS AND RELATIONS

Fundamental to a study of gas turbines is an understanding of thermodynamics and gas dynamics. The theory of gas turbine engines is based on the laws and principles of physics presented in the chapter that follows.

1.1 First Law of Thermodynamics

The first law of thermodynamics is a statement of the conservation of energy. A system is defined as a fixed, identifiable quantity of mass.

The first law of thermodynamics for a control volume undergoing steady flow, unit mass, one stream entering and leaving, and neglecting potential energy change, is

$$q_{12}+i_1+\frac{c_1^2}{2}=L_{12}+i_2+\frac{c_2^2}{2}, \qquad (1.1)$$

where
q_{12} - heat transferred to the system in going from state 1 to state 2, J/kg;
i - enthalpy, J/kg;
C - absolute velocity, m/s;
$C^2/2$ - kinetic energy, J/kg;
L_{12} - work done by the system in going from state 1 to state 2, J/kg.

The enthalpy is defined as

$$i \equiv U+pv, \qquad (1.2)$$

where
U - internal energy, J/kg;
p - pressure, Pa;
v - specific volume of the fluid, m^3/kg.

1.2 Mechanical Work for a Frictionless Steady-Flow Process

An expression for the mechanical work of a frictionless steady-flow process is very useful. For a frictionless, steady-flow process

$$-L_{12} = +\int_1^2 v\,dp+ \Delta KE+ \Delta PE, \qquad (1.3)$$

where ΔKE is the change in the kinetic energy and ΔPE is the change in the potential energy (J/kg both).

As is shown in Fig. 1.1, the value of $-\int v\,dp$ is represented by an area on a pressure-specific volume ***(p-v)*** diagram. Therefore, the pressure-volume diagram will be a valuable tool in the analysis of the work. It must be remembered that the area has meaning only for a reversible process.

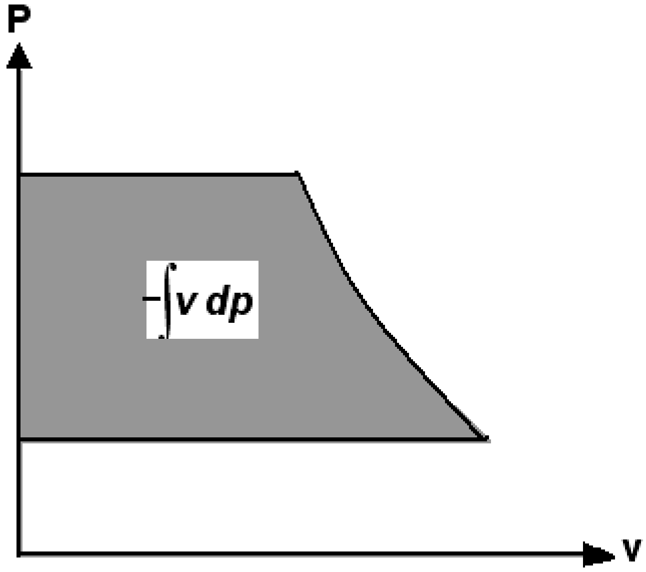

Figure 1.1. Pressure-specific volume diagram

1.3 Continuity Equation

The general form of the one-dimensional continuity equation is

$$G=\rho F C_a, \qquad (1.4)$$

where
G - mass rate of flow, kg/s;
ρ - density of fluid, kg/m^3;
F - cross-sectional area, m^2;
C_a - average axial velocity across the section, m/s.

1.4 Ideal Gas

An ideal gas is defined as a substance that has the equation of state, for unit mass,

$$pv=RT, \qquad (1.5)$$

where
R is called the gas constant (J/kg.K) and is a different constant for each gas.
R is also equal to the universal gas constant, $\overline{R}$, divided by the molecular weight $\overline{M}$ of the gas.

Other useful form of the equation of state is

$$pV=mRT, \qquad (1.6)$$

where
V - total volume, m^3;

m - mass, kg;
T - temperature, K.

$$R = c_p - c_v, \tag{1.7}$$

where
c_p – constant pressure specific heat, J/kg.K;
c_v – constant volume specific heat, J/kg.K.

The specific heat ratio is defined as

$$k = c_p/c_v. \tag{1.8}$$

1.5 Entropy

Entropy, ***s***, is a thermodynamic property that is defined by the relation

$$ds = (dQ/T)_{reversible}, \tag{1.9}$$

where
Q - total heat transfer, J/kg.

For an infinitesimal reversible process, for unit mass,

$$ds = \frac{\delta q}{T}. \tag{1.10}$$

For an infinitesimal, irreversible (spontaneous) process, for unit mass,

$$ds > \frac{\delta q}{T}. \tag{1.11}$$

Entropy is a function of the state of a system, which means that the change in its value is independent of the path between the initial and final states. However, since the equation (1.9) applies only for a reversible process, the entropy change must be calculated from values of *heat* and *temperature* along a *reversibl*e path between the initial and final states.

As shown in Fig. 1.2, the heat transfer for a reversible process 1-2 is the area beneath the curve.

1.6 Important Relations for a Pure Substance Involving Entropy

Very useful and important relations for a pure substance involving entropy can be derived. The first law of thermodynamics for a closed system, neglecting changes in kinetic and potential energy, is written in differential form,

$$\delta q = dU + \delta L. \tag{1.12}$$

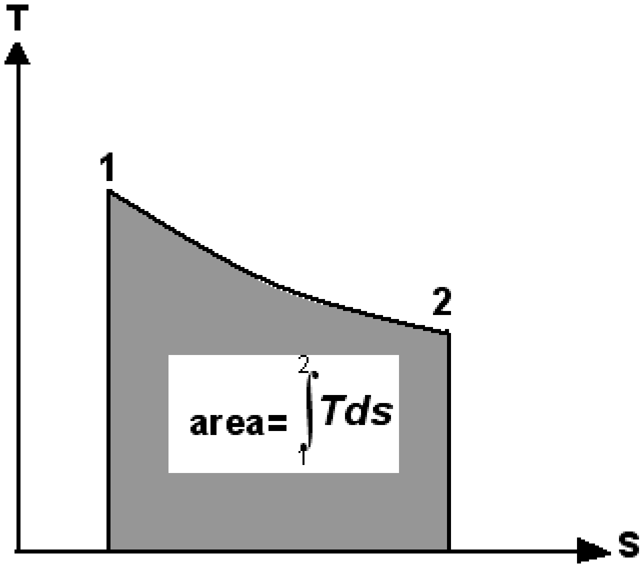

Figure 1.2. Temperature-entropy diagram

For a reversible process in which the only work involved is that at a moving boundary of the system,

$$\delta L = p\,dv. \tag{1.13}$$

Combining equations (1.12), (1.10) and (1.13) yields

$$Tds = dU + p\,dv. \tag{1.14}$$

However, once this equation has been written, it is realized that it involves only changes in properties and involves no path functions. Therefore, it must be concluded that this equation is valid for all processes, both reversible and irreversible, and that is applies to a substance undergoing a change of state as the result of flow across a boundary of a system as well as to a closed system.

Another useful form can be derived by using the definition of enthalpy,

$$i = U + pv \tag{1.15}$$

or, in differential form

$$di = dU + p\,dv + v\,dp. \tag{1.16}$$

Combining equations (1.14) and (1.16) yields

$$Tds = di - v\,dp. \tag{1.17}$$

For an ideal gas, on a mole basis,

$$d\bar{i} = \bar{c}_p\, dT, \quad (1.18)$$

$$p\bar{v} = \bar{R}\, T. \quad (1.19)$$

Therefore,

$$d\bar{s} = \frac{\bar{c}_p dT}{T} - \frac{\bar{R} dp}{p}. \quad (1.20)$$

Assuming constant specific heats, when integrated between states 1 and 2,

$$\bar{s}_2 - \bar{s}_1 = \bar{c}_p \ln\frac{T_2}{T_1} - \bar{R} \ln\frac{p_2}{p_1}. \quad (1.21)$$

For an adiabatic and reversible process (isentropic process),

$$\bar{s}_2 - \bar{s}_1 = 0 \quad (1.22)$$

or combining equation $0 = \bar{c}_p \ln\frac{T_2}{T_1} - \bar{R} \ln\frac{p_2}{p_1}$ with equations (1.7) and (1.8) yields the form:

$$\frac{T_2}{T_1} = \left(\frac{P_2}{P_1}\right)^{\frac{k-1}{k}}. \quad (1.23)$$

Topic 2 PRINCIPLES OF JET PROPULSION

The development of the gas turbine engine as an aircraft power plant has been so rapid that it is difficult to appreciate that prior to the 1950s very few people had heard of this method of aircraft propulsion. The possibility of using a reaction jet had interested aircraft designers for a long time, but initially the low speeds of early aircraft and the unsuitability of a piston engine for producing the large high velocity airflow necessary for the “jet” presented many obstacles.

Jet propulsion is a practical application of Newton's third law of motion which states that, “for every force acting on a body there is an opposite and equal reaction”. For aircraft propulsion, the “body” is atmospheric air that is caused to accelerate as it passes through the engine.

The force required to give this acceleration has an equal effect in the opposite direction acting on the apparatus producing the acceleration. A jet engine produces thrust in a similar way to the engine/propeller combination. Both propel the aircraft by thrusting a large weight of air backwards, one in the form of a large air slipstream at comparatively low speed and the other in the form of a jet of gas at very high speed.

The same principle of reaction occurs in all forms of movement and has been usefully applied in many ways. The earliest known example of jet reaction is that of classic engine (Fig. 2.1a) invented by Hero of Alexandria and produced as a toy more than 2000 years ago. This toy showed how the momentum of steam issuing from a number of jets could impart an equal and opposite reaction to the jets themselves, thus causing the engine to revolve.

The familiar whirling garden sprinkler (Fig. 2.1b) is a more practical example of this principle, for the mechanism rotates by virtue of the reaction to the water jets. The high pressure jets of modern fire-fighting equipment are an example of “jet reaction”, for often, due to the reaction of the water jet; the hose cannot be held or controlled by one fireman. Perhaps the simplest illustration of this principle is afforded by the carnival balloon which, when the air or gas is released, rushes rapidly away in the direction opposite to the jet.

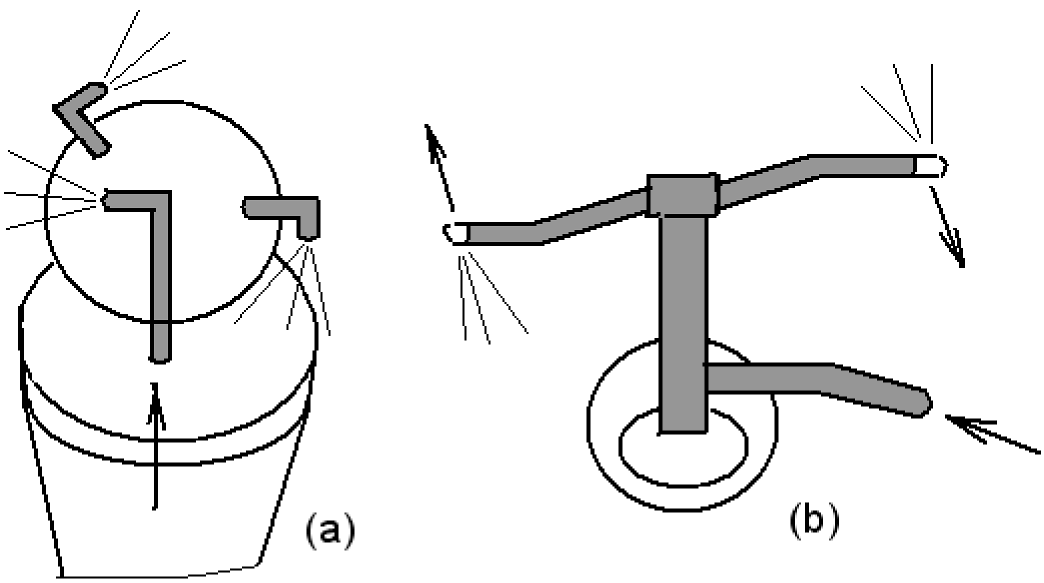

Figure 2.1. Known forms of jet reaction

Jet reaction is definitely an internal phenomenon and does not, as is frequently assumed, result from the pressure of the jet on the atmosphere. In fact, the jet propulsion engine, whether rocket, ramjet, or turbojet (Fig. 2.2), is a piece of apparatus designed to accelerate a stream of air or gas and to expel it at high velocity. There are, of course, a number of ways of doing this, but in all instances the resultant reaction or thrust exerted on the engine is proportional to the mass or weight of air expelled by the engine and to the velocity change imparted to it. In other words, the same thrust can be provided either by giving a large mass of air a little extra velocity or a small mass of air a large extra velocity. In practice the former is preferred, since by lowering the jet velocity relative to the atmosphere a higher propulsive efficiency is obtained.

In a ram jet engine forward motion is extra imparted to its duct from an external source, air is forced into the air intake (inlet) where it loses velocity or kinetic energy and increases its pressure energy as it passes through the diverging duct. The total energy is then increased by the combustion of fuel, and the expanding gases accelerate to atmosphere through the outlet duct (nozzle).

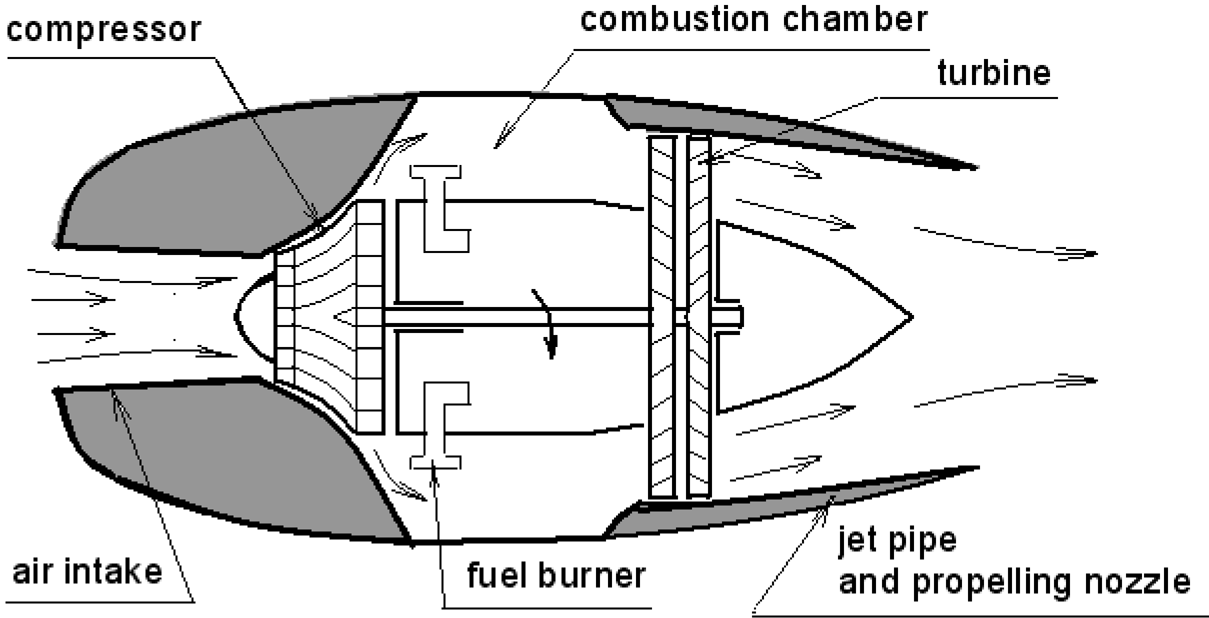

Figure 2.2. Early turbojet engine (of Whittle-type)

Although a rocket engine is a jet engine, it has one major difference in that it does not use atmospheric air as the propulsive fluid stream. Instead, it produces its own propelling fluid by the combustion of liquid or chemically decomposed fuel with oxygen, which it carries, thus enabling it to operate outside the earth’s atmosphere. It is, therefore, only suitable for operation over short periods.

Topic 3 GAS TURBINES FOR AIRCRAFT PROPULSION

There are various types of gas turbine engines used to propel aircraft. These usually fall into the categories of the turbojet, turbofan, and turboprop (turboshaft) engines. All could be constructed from the same gas generator.

3.1 Turbojet Engine

Fig. 3.1 illustrates a general layout of a nonafterburning (air standard) turbojet engine. Fig. 3.2 illustrates the temperature-entropy diagram for the entire cycle.

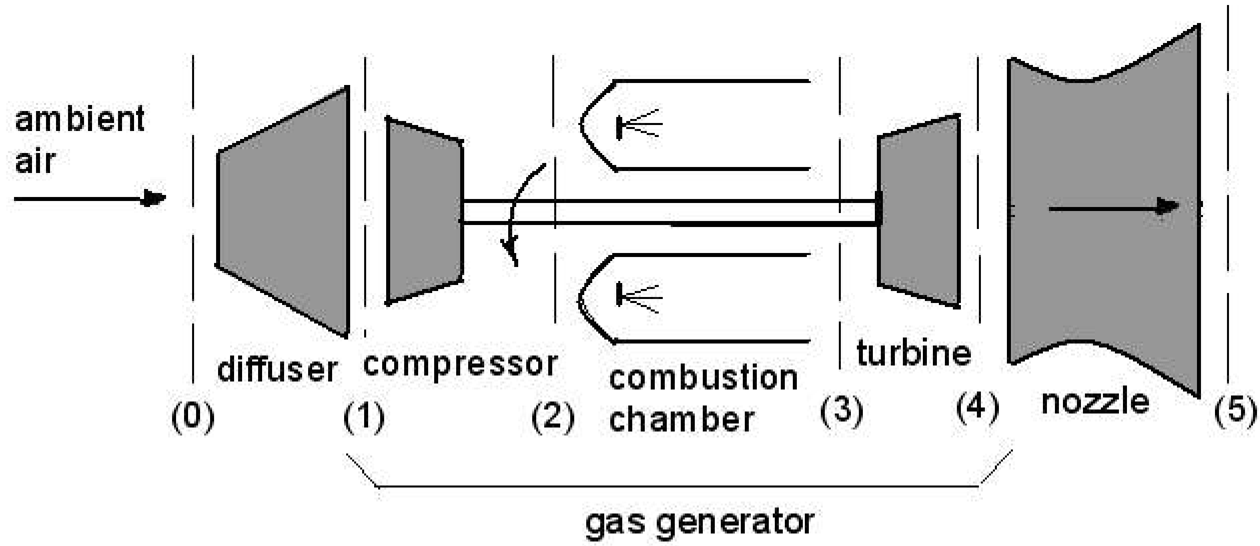

Figure 3.1. General layout of a nonafterburning turbojet engine

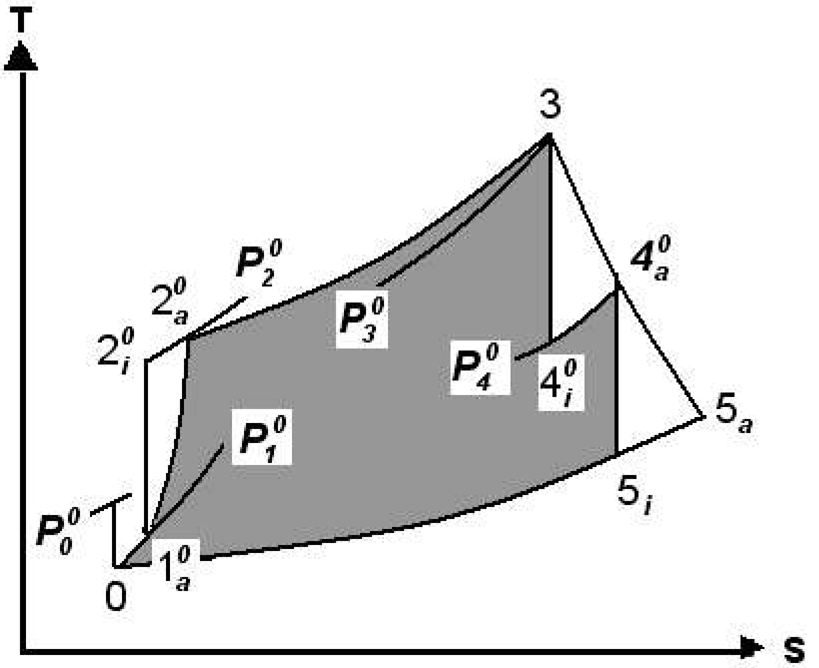

Figure 3.2. Diagram for a nonafterburning turbojet engine

Ambient air enters the diffuser where the velocity decreases and the static pressure increases. The air upon leaving the diffuser enters the compressor. The inlet and outlet states to the compressor, combustion chamber, and turbine that are used in defining the effectiveness and pressure drop are usually the total (stagnation) pressures. In the air standard analysis, the combustion chamber is replaced by a heat addition process. Ideally, no pressure drop occurs in the combustion chamber (heat addition process), in the actual case, a pressure drop does occur. Since the work (power) done by the turbine is equal to the work (power) required to drive the compressor, the pressure at the turbine exit is high. The air next expands through a nozzle, where the velocity is increased with a resulting decrease in pressure. Ideally, the nozzle operates isentropically. Actually, it operates adiabatically but irreversibly.

In the actual engine, fuel is added in the combustion chamber and burned with the air leaving the compressor. Since the maximum cycle temperature is approx. 1900K or lower, the excess air supplied is large; therefore, combustion is complete and the products of combustion have properties very close to those of air.

3.2 Turbofan Engine

The propulsion efficiency of the turbojet engine is quite low except at high flight speeds. This is evident by the large velocity of the stream of gases leaving the engine. To increase the propulsion efficiency, the nozzle exit velocity must be decreased. This can be accomplished by extracting more power in the turbine without increasing the power required to drive the compressor. This increase in power can be used to compress additional air, which is delivered around the combustion chamber, thereby increasing the mass of air compressed without increasing the amount of fuel supplied to the engine. The type of engine that does this is called a turbofan engine.

A turbofan engine is basically a turbojet engine in which some front-end compressor stages have been removed and replaced by large-diameter stages that are usually called fans. This is illustrated in Fig. 3.3. More turbine capability is needed to drive the combination fan and compressor. The added capability depends on the fan pressure ratio and the amount of air by-passing the basic gas generator (turbojet or core engine).

The fan pressure ratio (pressure ratio across the fan) varies from slightly above 1.9 to about 3.0. The by-pass ratio (BPR) is defined as the ratio of the air flow passing through the fan tips and the duct to the air flow passing through the gas generator (core) engine. Therefore,

$$1+BPR=\mathbf{G_{total}}/\mathbf{G_{gas\ generator}} \quad (3.1)$$

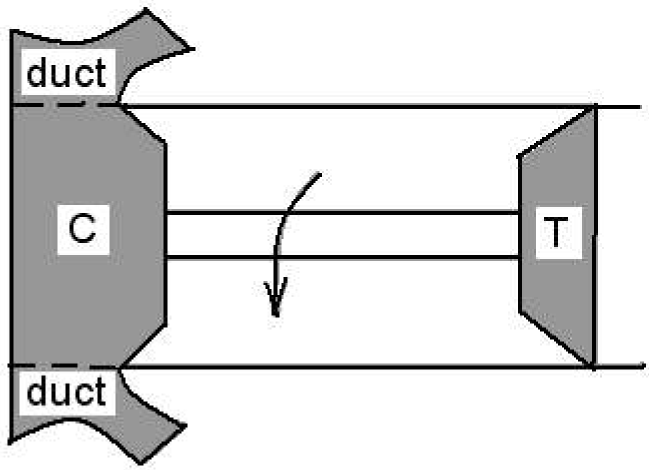

Figure 3.3. General layout of a turbofan engine

It is usually true that the higher fan pressure ratios are associated with low by-pass ratio engines and the high by-pass ratio engines are associated with low fan pressure ratios.

The turbofan has the advantage that a large increase in thrust may be achieved by taking an existing turbojet and adding a fan to it. This lowers the exit velocity and increases the propulsion efficiency. The turbofan engine, since it has no increase in fuel flow for this added thrust, has more thrust per mass of air entering the gas generator and therefore a lower thrust- specific fuel consumption.

There are two general types of turbofan engines. These are the nonmixed (a) turbofan engine (separate streams) and the mixed-flow (b) turbofan engine. These are shown in Fig. 3.4.

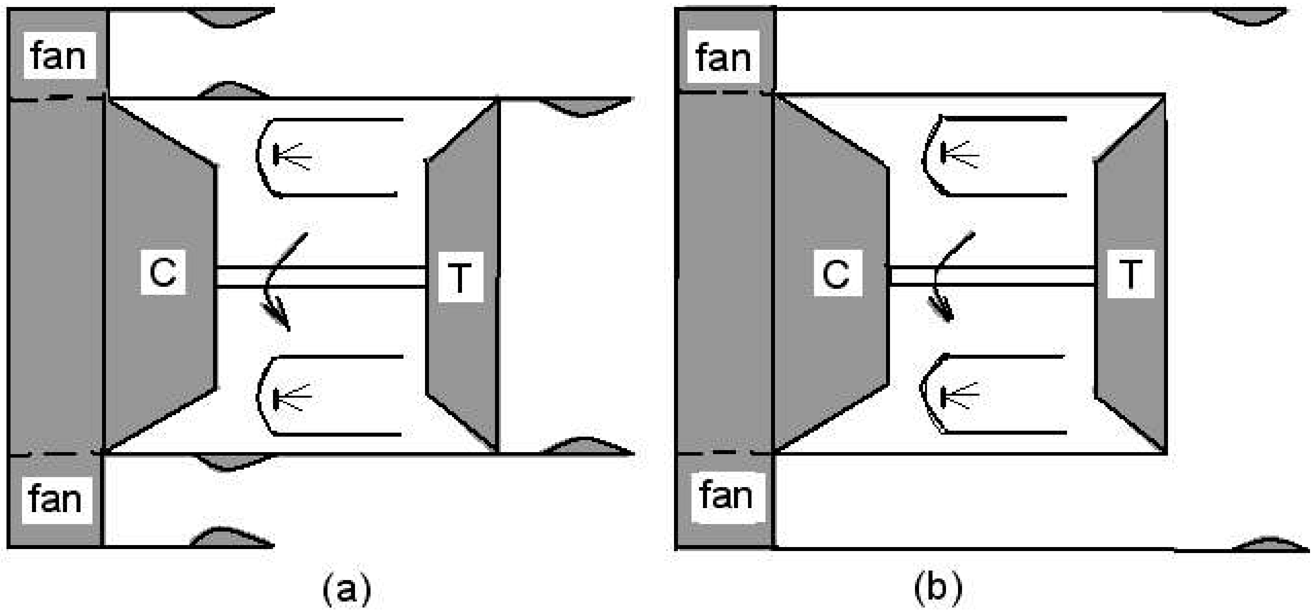

Figure 3.4. Nonmixed (a) and mixed (b) turbofan engine

In the nonmixed turbofan engine (Fig. 3.4a), the air that passes through the fan enters a duct, and then passes through its own nozzle. In the mixed-flow turbofan engine (Fig. 3.4b), the air that passes through the fan is ducted around the combustion chamber and mixed with the gas generator (core) gases behind the turbine. The static pressure at the point of mixing must be the same for both streams. The mixed streams then pass through the common nozzle. Each type of engine is currently being built. Turbofan engines built over the years have taken on many different configurations. Two of these are illustrated in Figs 3.5 and 3.6.

Fig. 3.5 illustrates the General Electric CF700 turbofan engine. This engine had the fan blades as an extension of the turbine blades. It eliminated the need to beef up the shaft between the turbine and the compressor fan to handle the increased power that would be required to drive the fan compressor unit but did require a seal between the hot, high-pressure turbine blades and the cold, low-pressure fan blades and could present high thermal stresses in the turbine fan blades because of the temperature difference of the two streams.

Fig. 3.6 illustrates the General Electric CF6 turbofan engine, a high by-pass ratio engine.

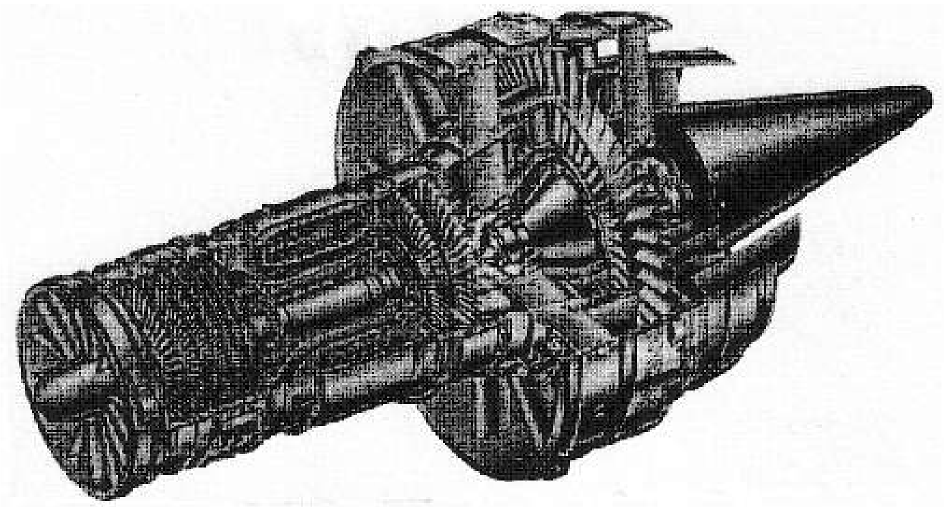

Figure 3.5. General Electric's CF700 turbofan engine

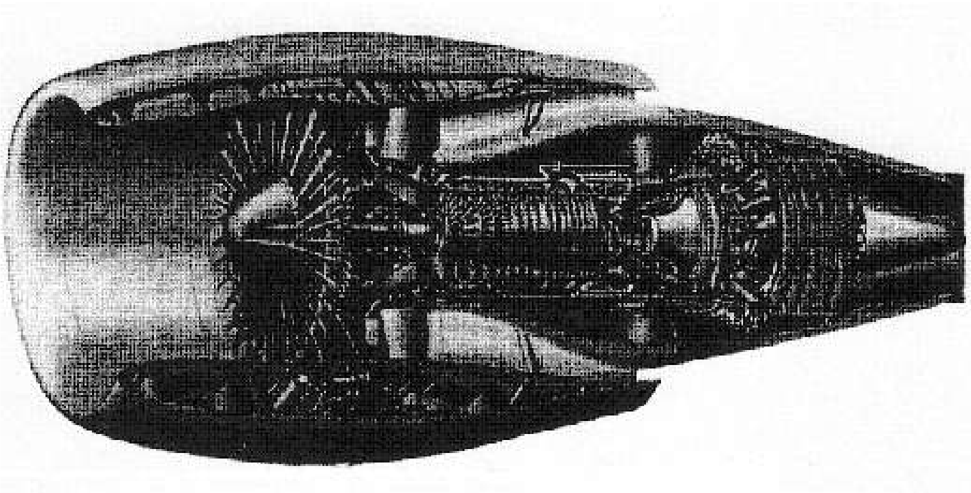

Figure 3.6. General Electric's CF6-6 turbofan engine

Observe the relative size of the fan-low compressor unit compared to the high-pressure compressor spool and the number of compressor stages compared to the number of turbine stages.

In all cases, the same gas generator (turbojet engine) could be used. Note that if the turbojet engine was converted to a turbofan engine with a by-pass ratio of 5.0, the thrust would approximately double with the thrust-specific fuel consumption decreasing by 50%. It must be remembered that the turbofan engine has some disadvantages.

The turbofan engine would have a much larger frontal area, would be much heavier, and, if one had started with an engine that had been designed as a turbojet, the shaft would have to be modified to transmit the additional power required to drive the fan and compressor, and additional turbine capacity would have to be added.

Various techniques have been used to increase the thrust from an existing engine. These include:

1. Increasing the fan pressure ratio. This may involve adding an additional fan stage, which will add weight to the engine and require that additional power be extracted in the turbine, and may require an additional turbine stage.

2. Increasing the core (overall) pressure ratio. This can be accomplished by either increasing the rotational speed of the compressor and/or by adding another compressor stage.

3. Increasing the by-pass ratio. This can be accomplished by increasing the diameter of the fan blades and therefore the diameter of the engine. In addition to requiring that additional power be extracted in the turbine, this will increase the weight of the engine and will reduce the ground clearance between the engine and the ground if the engine is mounted beneath the wing of the aircraft.

4. Increasing the turbine inlet temperature. This may require a redesign of the blades at the inlet to the turbine, might require additional turbine cooling and/or a different blade cooling technique.

3.3 Turboprop (Turboshaft) Engine

A third type of gas turbine used for propulsion of an airplane is the turboprop engine. Propulsion by a turboprop engine is accomplished through the combined action of a propeller at the front of the engine and the thrust produced by the exhaust gases from the gas turbine. A turbojet engine may be converted to a turboprop engine by adding an additional turbine to drive a propeller through a speed-reducing gear system. This type of engine is shown schematically in Fig 3.7.

A turboprop engine combines the advantages of a turbojet engine with the propulsion efficiency of a propeller. The turbojet engine derives its thrust from a large momentum change of a relatively small mass of air, whereas the turboprop engine develops its propulsive force by imparting a small momentum change to a relatively large mass of air.

The turbine of a turbojet engine extracts only the power necessary to drive the compressor and accessories, whereas the turboprop engine is designed not only to absorb the power required to drive the compressor and accessories but also to deliver to the propeller shaft the maximum torque possible.

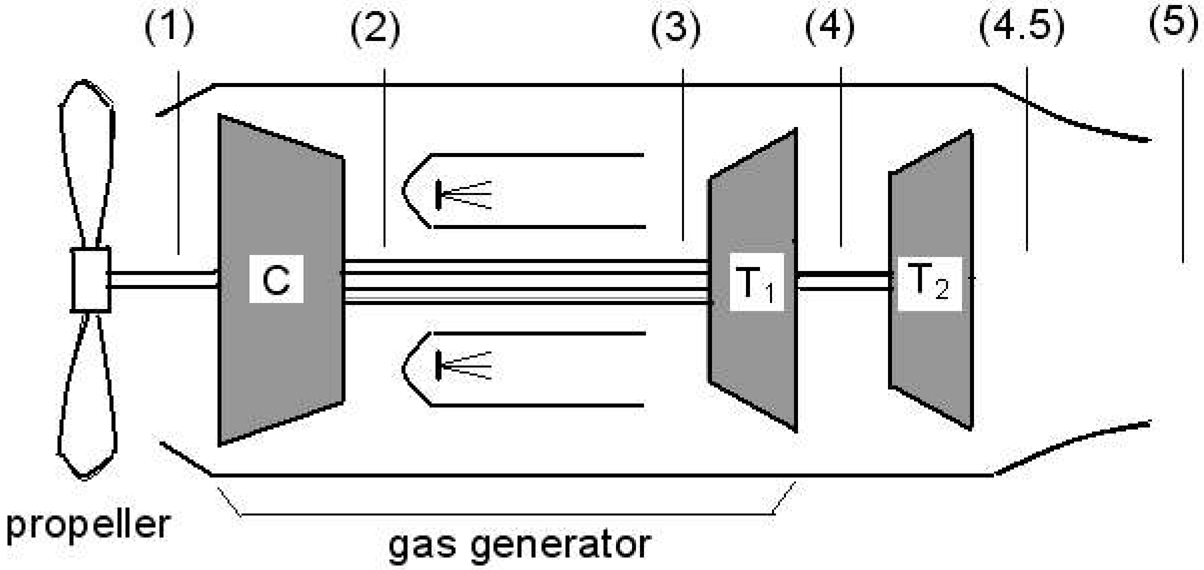

Figure 3.7. General layout of a turboprop gas turbine engine

The propeller of a typical turboprop engine is responsible for roughly 90% of the total thrust at sea level static conditions. This percentage varies with airspeed, altitude, and other engine parameters.

The turboprop engine has a lower thrust-specific fuel consumption during take-off and at low to moderate subsonic flight speeds than do the turbojet and turbofan engines, this advantage decreasing as the altitude and airspeed increase. The propeller efficiency remains fairly constant up to a Mach number of approximately 0.5, then drops rapidly. This means that aircraft propelled by turboprop engines are usually limited, by current technology, to about 650 km/h.

The choice between the turbojet/turbofan (jet thrust) and the turboprop (shaft power and jet thrust) engines involve many considerations. For instance, the higher the operating speeds, the larger may be the proportion of total output in the form of jet thrust. Also, an extra turbine stage may be required if more than a certain proportion of the total power is to be provided by the propeller.

Since the turbojet is rated in thrust and the turboprop engine in shaft horsepower plus thrust, no direct comparison may be made between the two.

A comparison can be made by converting the horsepower developed by the turboprop engine to thrust or the thrust developed by the turbojet to horsepower.

When a turboprop engine operates under static conditions, the power produced by engine is normally expressed in equivalent shaft power (ESP).

The equivalent shaft power in flight at a given airspeed will be the sum of the shaft horsepower and the horsepower equivalent of the net jet thrust. It is normally assumed for this comparison that the propeller efficiency is 80%.

In principle, the turbofan engine is the same as the turboprop, the geared propeller being replaced by a duct-enclosed fan driven at engine speed, or, at any part the turbine speed. One fundamental operational difference between the turboprop and the turbofan is that the airflow through the fan of the turbofan engine is controlled by design so that the air velocity relative to the fan blades is unaffected by the airspeed of the aircraft. Also, the total airflow through the fan is much less than that through the propeller of a turboprop engine.

One should understand the problem a designer faces in designing the inlet duct for a turboprop engine. The reduction gear normally is located on the same end of the engine as the propeller. The reduction gear, on the order of approx. 9 to 1, is needed to reduce the high operating rpm of the turbine to a speed acceptable for driving the propeller. It must be capable of handling the heavy loads imposed on it yet be light in weight and small in frontal area.

Turboshaft engines are used to power almost all modern helicopters. The first shaft bears the compressor and the high speed turbine (often referred to as "Gas Generator"), while the second shaft bears the low speed turbine (“Free Turbine” or "Power Turbine"). This arrangement is used to increase speed and power output flexibility.

3.4 Afterburning and Duct Heater Engines

Many times, large increases in thrust are required for short periods of time, such as during take-off or climb.

The afterburner is designed as an extension to the engine. The gases leaving the turbine are reheated in order to increase the momentum of the gas stream at the engine exit. Afterburners have been used on both turbojet and turbofan engines. When an afterburner is added to a turbofan engine, the afterburning can be done in the primary stream (gas generator stream), in the ducted stream, or in both streams. Most afterburning turbofan engines are of mixed type, the one exception to this being the duct-burning turbofan engine. The reason for using the mixed (static pressure balanced) turbofan engine is that it eliminates a wall between two very hot streams which, if not eliminated, presents complicated cooling problems. Schematic diagrams of the turbojet with an afterburner, turbofan with afterburner, and turbofan with a duct heater are shown in Figs 3.8 and 3.9.

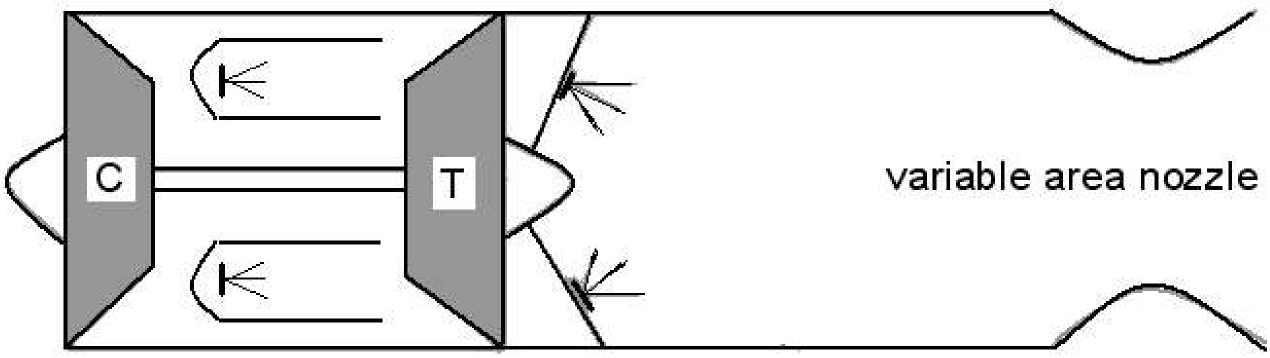

Figure 3.8. Turbojet engine with afterburner

Afterburning provides a means whereby the engine thrust can be increased well over 40% at sea level take-off to higher percentage increases at higher flight Mach numbers. The exact increase in thrust is dependent on the maximum afterburner temperature, by-pass ratio, and what stream or streams are used for afterburning.

This increase in thrust is obtained at the expense of a large increase in fuel consumption and with considerable length being added to the engine. Therefore, afterburning is used only when the maximum thrust is needed for a short period of time.

The reason the turbine exhaust gases can be reheated is that only a small percentage of the oxygen available in the compressor discharge air is used in the primary burner. Also, any air passing through the fan is ducted around the primary burner and mixed with the primary gas stream behind the turbine. Therefore, a large percentage of the oxygen in the original air is available in the stream behind the turbine.

The addition of an afterburner to an engine requires some limitations on the other components, especially the turbine. First, the turbine exit diffuser must be designed to reduce the velocity to an acceptable level at the afterburner combustion chamber inlet. Turbine exit stators are usually provided to reduce the turbine exit swirl to within limits required for good afterburner performance.

The reason that afterburning increases the thrust of a jet engine can best be visualized by referring to Fig. 3.10, which is an enthalpy-entropy diagram showing both an afterburning and non-afterburning turbojet engine. This figure, which is not drawn to scale, shows that for the same turbine discharge conditions, the enthalpy change across the nozzle for the afterburning engine is considerably greater than that for the non-afterburning engine. This is easily explained by the fact that constant pressure lines diverge on an enthalpy-entropy (and therefore temperature) increases as shown in Fig. 3.10.

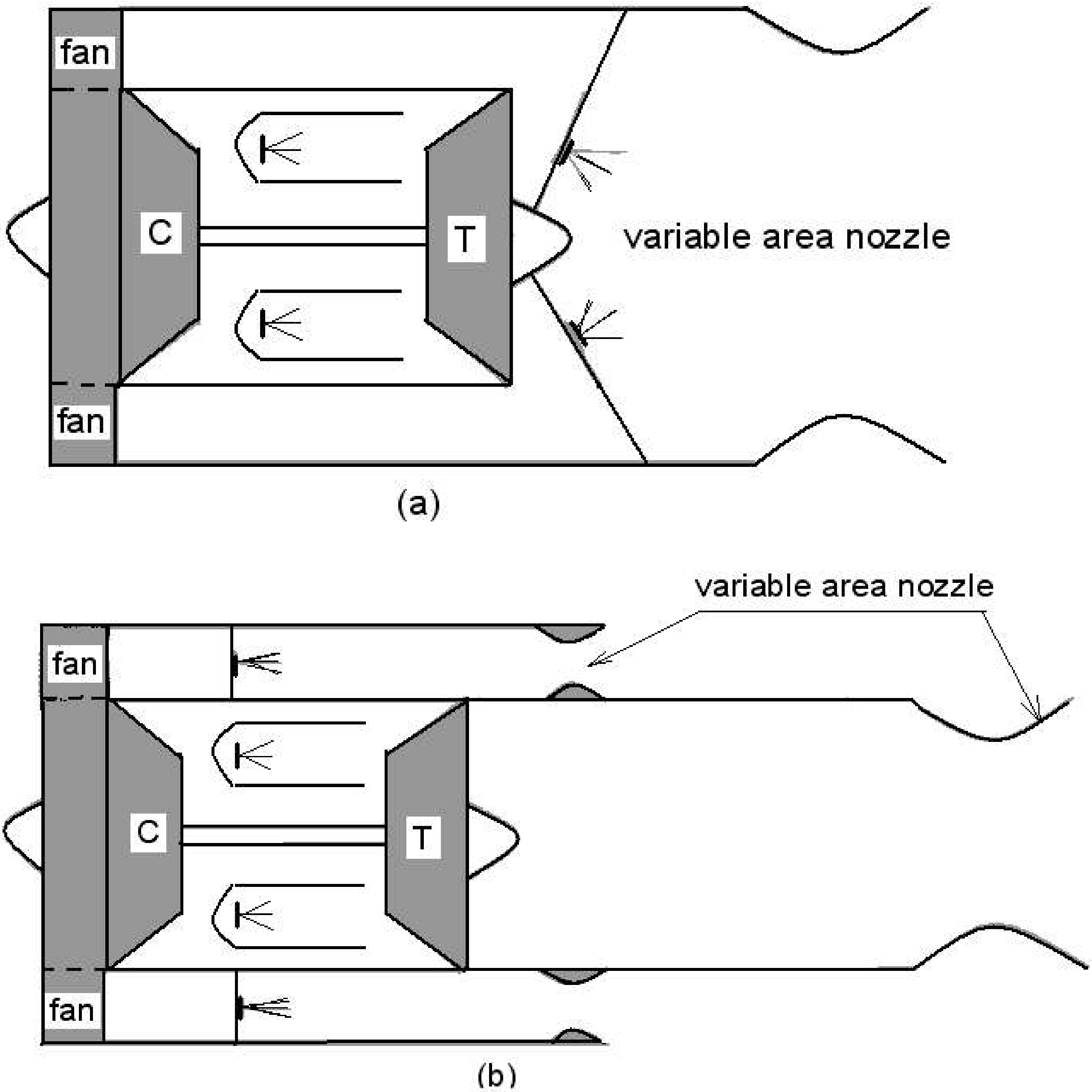

Figure 3.9. Mixed (a) and nonmixed (b) turbofan engines with afterburner

It should also be noted that the maximum gas temperature leaving the afterburner can be several hundred degrees above the maximum temperature permissible at the primary burner exit. This is explained by the fact that the turbine stators limit the primary burner temperature, whereas the liner cooling air limits the average afterburner temperature.

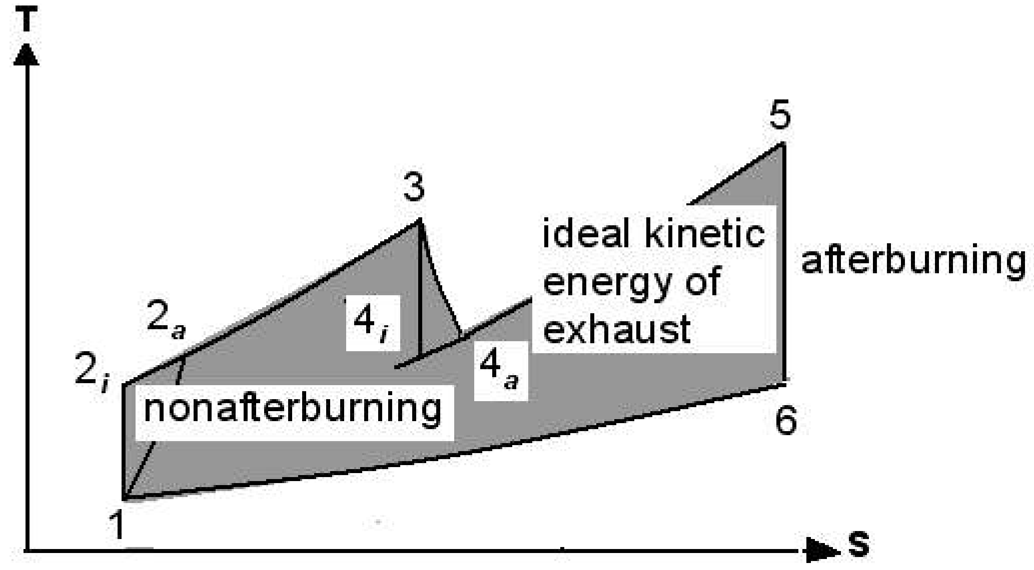

Figure 3.10. Enthalpy-entropy diagram for afterburning and nonafterburning turbojet engines

3.5 Turboprop Engine with Regenerator

It may be noted that one way to improve the specific fuel consumption of an engine was to add a regenerator. This normally is not done with gas turbines used for propulsion because of the added weight.

One known exception was the U.S. Navy’s contract with Allison Division of General Motors for the development of the T78 turboprop engine. This was an attempt to build a turboprop engine with much lower fuel consumption at partial power. The engine was to be used in antisubmarine and airborne early warning aircraft to allow them to spend more time on a search mission before returning to be refueled.

The regenerator was situated aft of the turbine to minimize the frontal area. This arrangement meant that the air, upon leaving the compressor, had to be ducted to the rear of the regenerator. It then flowed back toward the combustion chamber, absorbing energy from the exhaust gases.

The regenerator was a fixed, tubular, cross-counter-flow-type heat exchanger with the cold air inside the tubes, the hot gases outside.

Topic 4 PRINCIPAL PARAMETERS OF ENGINE PERFORMANCE

4.1 Thermal Efficiency

It is important at this point to introduce the concept of thermal efficiency of a heat engine. Thermal efficiency is defined as the fraction of the gross heat input to a system during a cycle that is converted into net work output, or

$$\eta_{th} = L_{net}/Q_{in}, \tag{4.1}$$

where
L_{net} – energy sought, J/kg;
Q_{in} – energy that costs, J/kg.

Another form for thermal efficiency that is often quite used:

$$\eta_{th} = \frac{Q_{in} - Q_{out}}{Q_{in}} = 1 - \frac{Q_{out}}{Q_{in}}, \tag{4.2}$$

where
Q_{in} - inlet total heat transfer, J/kg;
Q_{out} - outlet total heat transfer, J/kg.

4.2 General Thrust Equation

A fairly general equation for evaluating thrust from an air-breathing jet engine can be derived from the conservation of momentum and mass equation. Fig. 4.1 will be used in the derivation of a general equation from the position of an observer riding with the thrust-producing device.

In Fig. 4.1, the control surface is shown by the dashed line. Surface *A* is far upstream, where the pressure and velocity may be assumed uniform over the entire surface. Surface *B* is at the exit from the thrust-producing device. The side control surfaces are parallel to the upstream velocity and are far removed from the thrust-producing device. The derivation of the thrust equation will be for the time-invariant (steady-flow) condition.

Values that are known are shown in Fig. 4.1. The fuel is assumed to be added at a right angle to the direction of flow. The velocity and pressure at the inlet are assumed to be uniform, and the mass of air entering the thrust-producing device, G_a, enters the device through an area F_i. The products of combustion leave the thrust-producing device through an area F_e with a velocity C_e, a mass rate of flow $\boldsymbol{G_e}$, and at a pressure $\boldsymbol{p_e}$. The velocity and pressure of the air that passes around the thrust-producing device are $\boldsymbol{C}$ and $\boldsymbol{p_a}$, respectively. A rate of air, $\boldsymbol{G_s}$, leaves through the side control surface.

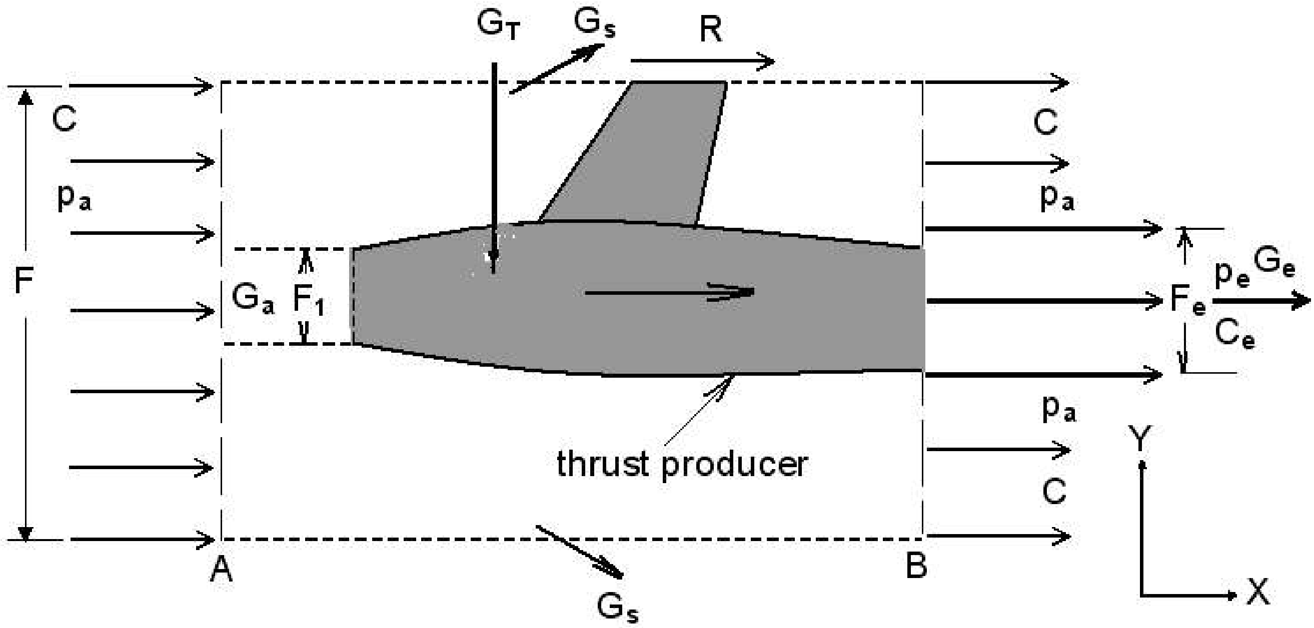

Figure 4.1. Generalized thrust-producing device

The thrust, $\boldsymbol{R}$, developed by the generalized thrust producer may be derived from Newton's second law.

For the engine (thrust producer)

$$\boldsymbol{R}=(\boldsymbol{p_e}-\boldsymbol{p_a})\, \boldsymbol{F_e} + \boldsymbol{G_a}\, \boldsymbol{C_e}\, (1+\boldsymbol{q_f}) - \boldsymbol{G_a}\, \boldsymbol{C}, \tag{4.3}$$

where

$\boldsymbol{q_f}=\boldsymbol{G_f}/\boldsymbol{G_a}$ - relative fuel mass rate. (4.4)

The specific thrust, $\boldsymbol{R_{sp}}$, is defined as the thrust produced when a unit mass of air per second enters the device, or

$$\boldsymbol{R_{sp}}= \boldsymbol{R} /\boldsymbol{G_a}. \tag{4.5}$$

4.3 Combustion Process Parameters

Further the net work and thermal efficiency must be considered when the actual working fluid is used. The solution to this problem involves calculating the amount of excess air that must be supplied to give the specified temperature leaving the combustion chamber. Once this is known, the analysis of the product gas expanding through the turbine will be known.

Calculating the percent excess air (and therefore the products leaving the combustion chamber) that must be supplied involves writing an energy balance around the combustion chamber. Since the temperature leaving the combustion chamber currently is approx. 1900 K or lower, complete combustion may be assumed.

Consider the combustion of a hydrocarbon fuel with dry air. The temperature and pressure of the air entering the combustion chamber will be known from the compressor calculations, and the temperature and phase of the fuel will be known. The temperature of the products leaving the combustion chamber is a controlled (specified) quantity. From the steady-flow energy equation, assuming no heat loss from the combustion chamber, the equality of the total enthalpies of products and reactions is evident.

Combustion efficiency takes into account the fact that there will be some heat loss due to radiation and conduction and that incomplete combustion might occur. *Combustion efficiency* is the actual thermal energy Q_1 added to the working fluid divided by the thermal energy Q_0 that should have been released when all the combustible constituents of the fuel have been completely oxidized in an adiabatic combustor. This means that the combustor (burner) efficiency is

$$\eta_B = Q_1/Q_0. \tag{4.6}$$

Once the gas analysis leaving the combustion chamber is known, the work produced by the turbine may be calculated. The composition of the product gas expanding through the turbine will be assumed to remain constant.

First, consider the isentropic expansion through the gas generator turbine. The pressure, temperature, and composition entering the gas generator turbine are known, along with the turbine efficiency and the actual work required. Therefore, by trial and error, the pressure and temperature at the turbine exit may be determined. This means that the conditions entering the power turbine will be known and, since the power turbine efficiency is known, the work developed by the power turbine may be determined.

A factor that is commonly used in judging the performance of a gas turbine is the *specific fuel consumption* (SFC).

The specific fuel consumption is the mass of fuel required per hour (G_{fh}) per kilowatt of net power output (W_{net}). This reduces to

$$SFC = G_{fh}/W_{net}, \text{ kg/kW.h.} \tag{4.7}$$

Two more performance parameters are of interest for thrust-producing devices. These are the propulsion efficiency, and the overall efficiency.

4.4 Propulsion Efficiency

One measure of the performance of a thrust-producing device is the ratio of the thrust power to the jet power. The thrust power is defined as the product of the thrust and the flight velocity. The jet power is the change of kinetic energy of the gases passing through the device.

Neglecting the pressure term and assuming that the fuel-air ratio (q_f) is zero yields

$$\eta_p = \frac{C(C_e - C)}{C_e^2/2 - C^2/2} = \frac{2C/C_e}{1 + C/C_e}. \qquad (4.8)$$

Examination of this equation indicates that the maximum propulsion efficiency occurs when the velocity leaving the thrust-producing device equals the velocity of the device. This, however, is not practical, since when $C=C_e$ the thrust is zero.

4.5 Overall Efficiency

The overall efficiency is defined as the product of the thermal efficiency and the propulsion efficiency, or

$$\eta_o = \eta_{th}\eta_p. \qquad (4.9)$$

Topic 5 IDEAL AND REAL CYCLES. PARAMETERS DISTRIBUTION IN TURBOJET ENGINE

5.1 General

The gas turbine engine is essentially a heat engine using air as a working fluid to provide thrust. To achieve this, the air passing through the engine has to be accelerated; this means that the velocity or kinetic energy of the air is increased. To obtain this increase, the pressure energy is first of all increased, followed by the addition of heat energy, before final conversion back to kinetic energy in the form of a high velocity jet efflux.

5.2 Basic Working Cycle

The basic (simplest) gas turbine engine is shown in Fig. 5.1. The engine consists of a compressor where air is compressed adiabatically, a combustion chamber where the fuel is burned with air, resulting in the maximum cycle temperature occurring at state 3. The product of combustion then expand adiabatically in the turbine (or turbines), part of the work developed in the turbine being used to drive the compressor, the remainder being delivered to equipment external to the gas turbine. This basic gas turbine engine results by adding additional turbine wheels (or a separate power turbine) behind the gas generator so that the gases can expand back to (or nearly so) the pressure at the compressor inlet. The basic gas turbine engine illustrated in Fig. 5.1 has a separate power turbine (PT).

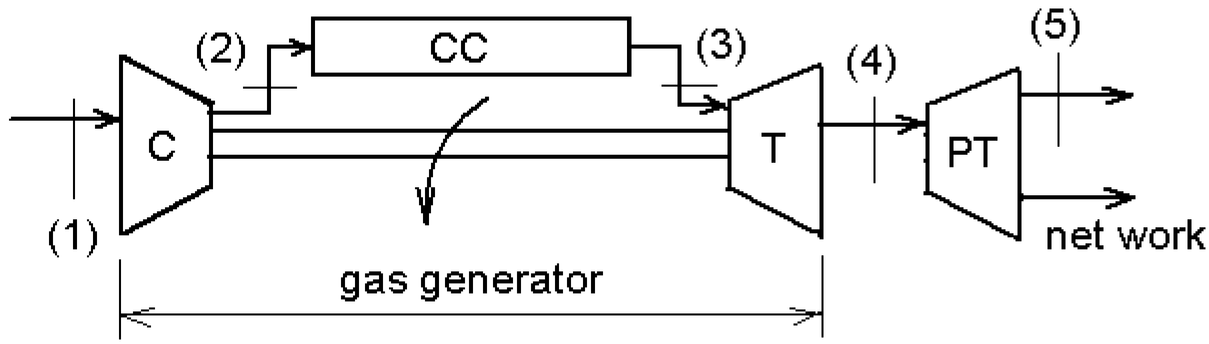

Figure 5.1. Basic gas turbine engine

This section is concerned with the air standard cycle. In an air standard cycle, air is assumed to be the working fluid throughout, the combustion chamber being replaced by a heat addition process. The cycle, which may be considered a close cycle, is completed by a heat rejection process.

The *ideal air standard cycle* assumes that the compression and expansion processes are adiabatic and reversible (isentropic), that there is no pressure drop during the heat addition process, and that the pressure leaving the turbine is equal to the pressure entering the compressor. The temperature-entropy (***T-s***) and pressure-specific volume (***p-v***) diagrams for the ideal air standard basic cycle are shown in Fig. 5.2.

Each component of the gas turbine engine operates in a steady-flow manner. Therefore, for an ideal cycle, the compressor work, the heat added, the turbine work, and the net work become, respectively, L_C, q_{in}, : L_T and L_{net}.

Note: Remembering that the compressor and turbine are assumed to operate adiabatically and reversibly, but that the heat process is one of zero work, and that the total (stagnation) enthalpy is the sum of the static enthalpy and the kinetic energy.

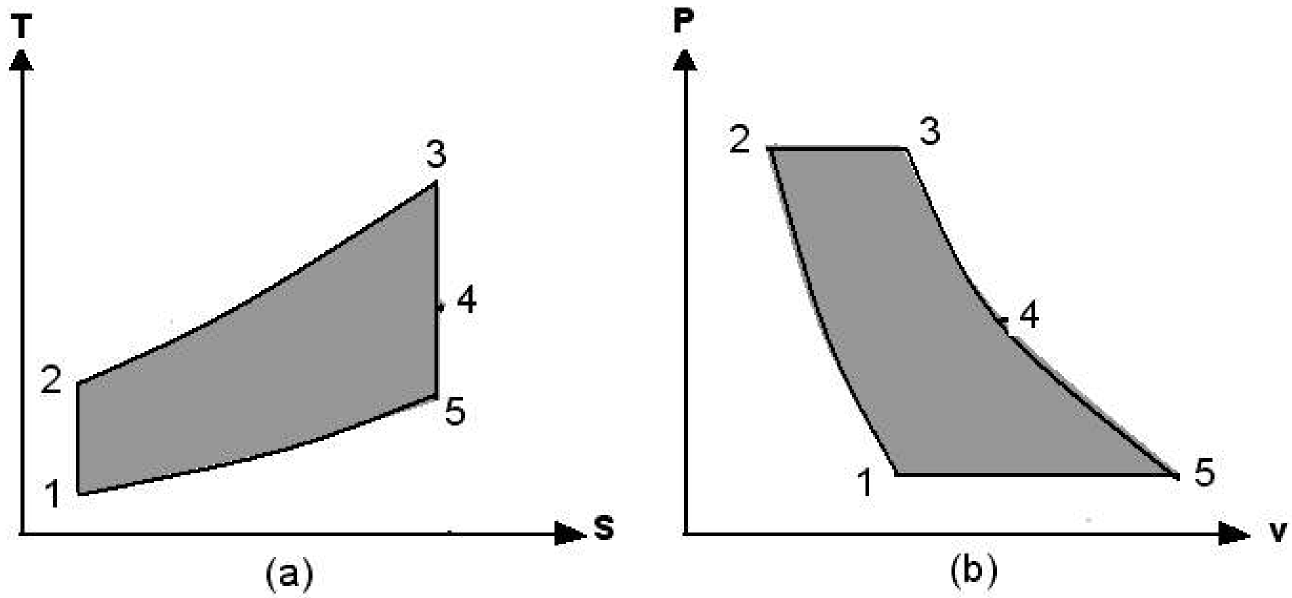

Figure 5.2. Temperature-entropy (a) and pressure-specific volume (b) diagrams for an ideal air standard basic cycle

For a heat engine, the cycle thermal efficiency is useful. For an ideal cycle, the thermal efficiency increases with increasing pressure ratio. This conclusion was found for constant specific heats, but the trend also applies for variable specific heats if operating under ideal conditions.

The pressure ratio for maximum net work increases with increasing turbine inlet temperature and a fixed compressor inlet temperature. This conclusion was also derived for constant specific heats, and the same trend applies for variable specific heats. The compressor inlet temperature is fixed by the atmospheric conditions, whereas the turbine inlet temperature is limited by the maximum temperature the turbine materials can withstand.

5.3 Basic Cycle with Friction

In a simplest actual gas turbine engine, the compressor and turbine may be assumed to operate adiabatically but not isentropically. The heat addition process will have a pressure drop, and the pressure at the exit from the turbine will be above the pressure of the air entering the compressor.

The *isentropic compressor efficiency* (sometimes referred to as the adiabatic compressor efficiency) is the ratio of the isentropic work of compression to the actual work of compression when both are compressed to the same final pressure. Care must be taken, as sometimes the final static pressure is used and sometimes the final total pressure is used. In all cases, the inlet total state is used. In this text, the isentropic compressor efficiency will be at the distance from the inlet total state to the same final total pressure, often referred to as the total *compressor efficiency*.

The *isentropic turbine efficiency* (in some cases called the adiabatic turbine efficiency) is the ratio of the actual turbine work divided by the isentropic turbine work when both expand from the same initial state to the same final pressure. Once again, the final state can be the same final static pressure or the same final total pressure. In this text, it is assumed to be the same total pressure. Pressure loss, in general, is expressed as the drop in total pressure divided by the inlet total pressure.

The working cycle of the gas turbine engine is similar to that of the four-stroke piston engine.

However, in the gas turbine engine, combustion occurs at a constant pressure, whereas in the piston engine it occurs at a constant volume. In both instances there are induction, compression, combustion and exhaust.

These processes are intermittent in the case of the piston engine whilst they occur continuously in the gas turbine. In the piston engine only one stroke is utilized in the production of power, the others being involved in the charging, compressing and exhausting of the working fluid. In contrast, the turbine engine eliminates the three “idle” strokes, thus enabling more fuel to be burnt in a shorter time; hence it produces a greater power output for a given size of engine.

Due to the continuous action of the turbine engine and the fact that the combustion chamber is not an enclosed space, the pressure of the air does not rise, like that of the piston engine, during combustion but its volume does increase. This process is known as heating at constant pressure. Under these conditions there is no peak or fluctuating pressures to be withstood, as is the case with the piston engine with its peak pressures in excess of 6900 kPa. It is these peak pressures which make it necessary for the piston engine to employ cylinders of heavy construction and to use high octane fuels, in contrast to the low octane fuels and the light fabricated combustion chambers used on the turbine engine.

The working cycle upon which the gas turbine engine functions is, in its simplest form, represented by the cycle shown on the pressure-volume diagram in Fig. 5.3.

Point A represents air at atmospheric pressure that is compressed along the line AB. From B to C heat is added to the air by introducing and burning fuel at constant pressure, thereby considerably increasing the volume of air. Pressure losses in the combustion chambers are indicated by the drop between B and C. From C to D the gases resulting from combustion expand through the turbine and jet pipe back to atmosphere. During this part of the cycle, some of the energy in the expanding gases is turned into mechanical power by the turbine; the remainder, on its discharge to atmosphere, provides a propulsive jet.

Because the turbojet engine is a heat engine, the higher the temperature of combustion the greater is the expansion of the gases. The combustion temperature, however, must not exceed a value that gives a turbine gas entry temperature suitable for the design and materials of the turbine assembly.

The use of air-cooled blades in the turbine assembly permits a higher gas temperature and a consequently higher thermal efficiency.

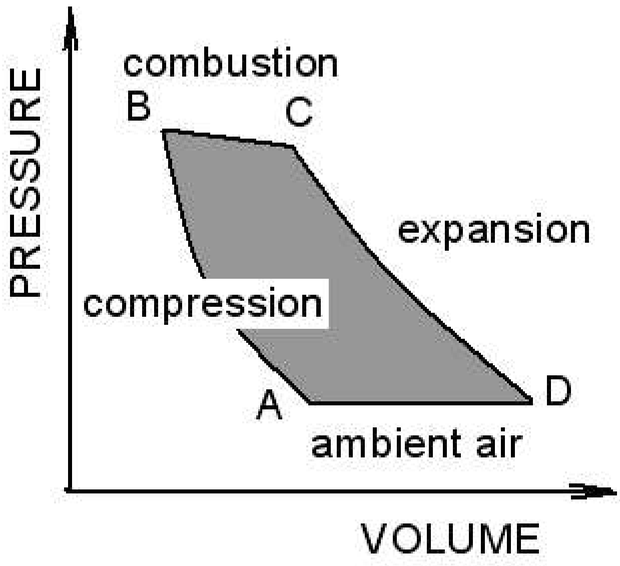

Figure 5.3. Working cycle on a pressure-volume diagram

5.4 The Relations between Pressure, Volume and Temperature

During the working cycle of the turbine engine, the airflow or “working fluid” receives and gives up heat, so producing changes in its pressure, volume and temperature. These changes as they occur are closely related, for they follow a common principle that is embodied in a combination of the laws of Boyle and Charles. Briefly, this means that the product of the pressure and the volume of the air at the various stages in the working cycle are proportional to the absolute temperature of the air at those stages. This relationship applies for whatever means are used to change the state of the air. For example, whether energy is added by combustion or by compression, or is extracted by the turbine, the heat change is directly proportional to the work added or taken from the gas.

There are three main conditions in the engine working cycle during which these changes occur. During compression, when work is done to increase the pressure and decrease the volume of the air, there is a corresponding rise in the temperature. During combustion, when fuel is added to the air and burnt to increase the temperature, there is a corresponding increase in volume whilst the pressure remains almost constant. During expansion, when work is taken from the gas stream by the turbine assembly, there is a decrease in temperature and pressure with a corresponding increase in volume.

Changes in the temperature and pressure of the air can be traced through an engine by using the airflow diagram in Fig. 5.4. With the airflow being continuous, volume changes are shown up as changes in velocity.

The efficiency with which these changes are made will determine to what extent the desired relations between the pressure, volume and temperature are attained. For the more efficient the compressor, the higher the pressure generated for a given work input; that is, for a given temperature rise of the air. Conversely, the more efficiently the turbine uses the expanding gas, the greater the output of work for a given pressure drop in the gas.

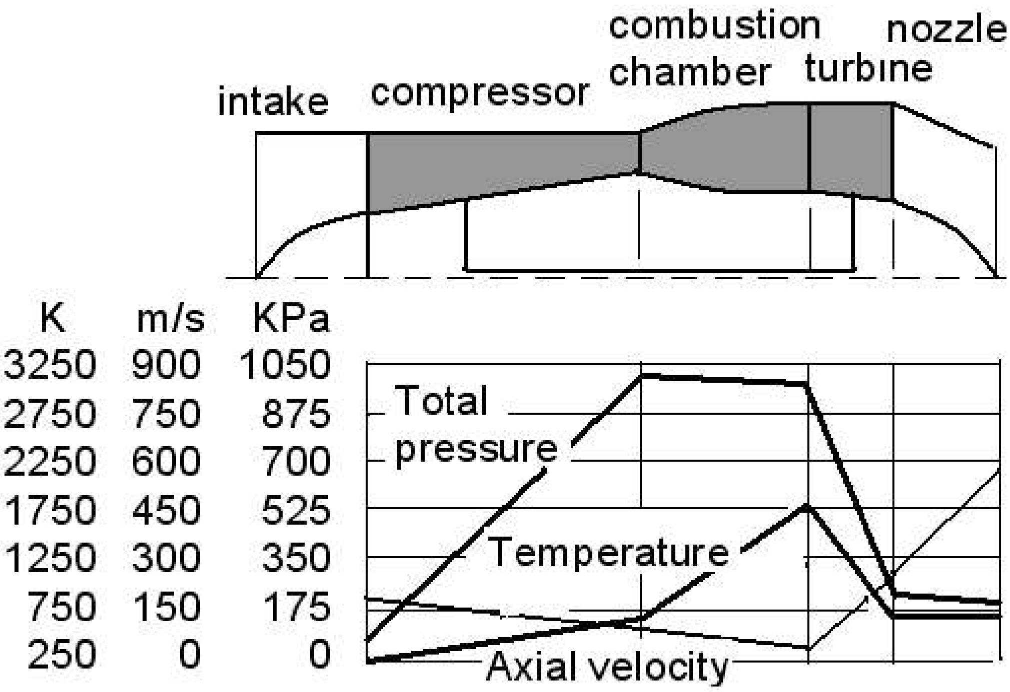

Figure 5.4. Airflow diagram

When the air is compressed or expanded at 100% efficiency, the process is said to be adiabatic. Since such a change means there is no energy losses in the process, either by friction, conduction or turbulence, it is obviously impossible to achieve in practice; 90% is a good adiabatic efficiency for the compressor and turbine.

5.5 Changes in Velocity and Pressure

During the passage of the air through the engine, aerodynamic and energy requirements demand changes in its velocity and pressure. For instance: during compression, a rise in the pressure of the air is required and not an increase in its velocity. After the air has been heated and its internal energy increased by combustion an increase in the velocity of the gases is necessary to force the turbine to rotate. At the propelling nozzle a high exit velocity is required, for it is the change in the momentum of the air that provides the thrust on the aircraft. Local decelerations of airflow are also required, as for instance, in the combustion chambers to provide a low velocity zone for the flame to burn.

These various changes are effected by means of the size and shape of the ducts through which the air passes on its way through the engine. Where a conversion from velocity (kinetic) energy to pressure is required, the passages are divergent in shape. Conversely, where it is required to convert the energy stored in the combustion gases to velocity energy, a convergent passage or nozzle is used. These shapes apply to the gas turbine engine where the airflow velocity is subsonic or sonic, i.e. at the local speed of sound. Where supersonic speeds are encountered such as in the propelling nozzle of the rocket, ramjet and some jet engines, a convergent-divergent nozzle is used to obtain the maximum conversion of the energy in the combustion gases to kinetic energy.

The design of the passages and nozzles is of great importance, for upon their good design will depend the efficiency with which the energy changes are effected. Any interference with the smooth airflow creates a loss in efficiency and could result in component failure due to vibration caused by eddies or turbulence of the airflow.

Topic 6 TURBOMACHINES: COMPRESSOR AND TURBINE

Efficient compression of large volumes of air is essential for a successful gas turbine engine. This has been achieved in two types of compressors, the axial-flow compressor and the centrifugal-or radial-flow compressor.

Compressors designed for maximum efficiency would not be difficult in operation were districted to a single operating condition. However, compressors must have good efficiency over a wide range of operating points. The object of a good compressor design is to obtain the most air through a given diameter compressor with a minimum number of stages while retaining relatively high efficiencies and aerodynamic stability over the operating range. The designer’s freedom is usually restricted by mechanical, geometric, cost, and time constraints. The compatibility of the compressor shaft speed with that of a good turbine design must also be considered.

Most of the gas turbine engines designed and/or built in the 1940s and early 1950s used centrifugal-flow compressors. Today, gas turbine engines are built with axial-flow compressors, centrifugal-flow compressors, and combinations of one or more axial-flow stages followed by a centrifugal-flow compressor.

Fig. 6.1 illustrates typical flow paths for axial-flow and centrifugal-flow compressors. The flow path in an axial-flow compressor is essentially parallel to the axis of rotation. Each stage includes a row of rotating blades where energy is added to the fluid. This rotor is followed by a row of fixed blades commonly referred to as a stator. Several stages are required in an axial-flow compressor to obtain the desired high pressure ratios.

In a centrifugal-flow compressor, the fluid enters at the center of the compressor and is turned radially outward. The rotating component of the centrifugal-flow compressor is followed by a diffusing passage, which may or may not incorporate stationary vanes or blades.

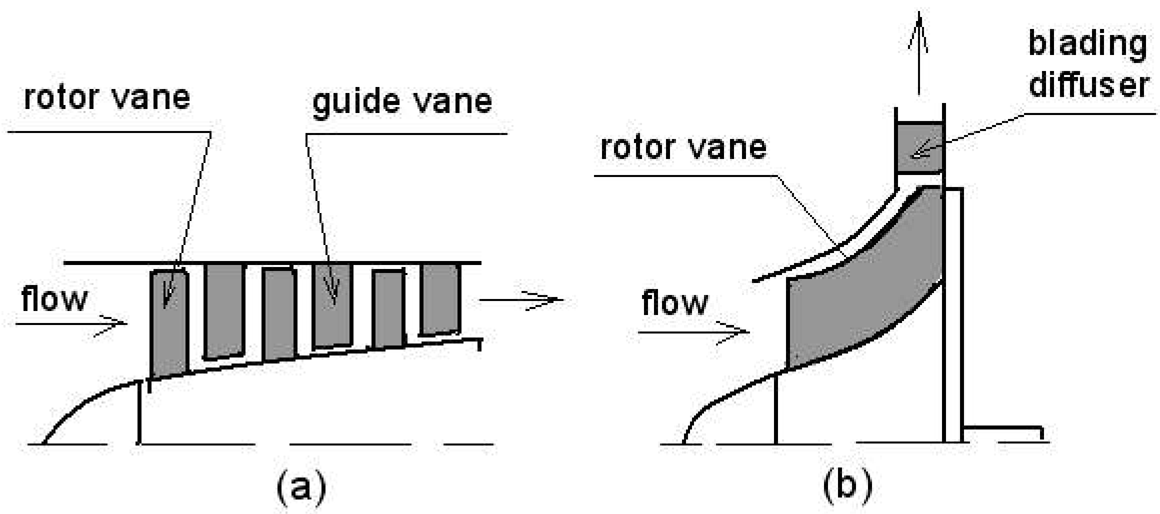

Figure 6.1. Flow path for axial-flow (a) and centrifugal-flow (b) compressors

The advantages of the axial-flow compressor over the centrifugal-flow compressor are

1. Smaller frontal area for a given mass rate of flow.

2. Flow direction at discharge more suitable for multistaging.

3. May use cascade experiment research in developing compressor.

4. Somewhat higher efficiency at high pressure ratios.

The advantages of the centrifugal-flow compressor over the axial-flow compressor are

1. Higher stage pressure ratio.

2. Simplicity and ruggedness of construction.

3. Less drop in performance with the adherence of dust to blades.

4. Shorter length for the same overall pressure ratio.

5. Flow direction of discharge air that is convenient for the installation of an intercooler and/or heat exchanger.

6. Wider range of stable operation between surging and choking limits at a given rotational speed.

In a turbine a stage consists of a stationary and a rotating member, the stationary row, commonly called a nozzle, and which precedes the rotating (blade) row.

There are two general types of turbines, the axial-flow and radial-flow turbines.

Axial-flow turbine (Fig. 6.2a) is almost always used in gas turbine engines. An axial-flow turbine may consist of one or more stages, each stage consisting of a nozzle row and a rotor row. The relative velocities in an axial-flow turbine are, in general, substantially higher than occur in axial-flow compressors, with a greater change in enthalpy per stage. In the nozzle row, the tangential velocity is increased in the direction of rotation with a corresponding drop in the static pressure. In the rotor row, the tangential velocity is decreased. Considerably fewer stages are needed in an axial-flow turbine than in an axial-flow compressor because in an axial-flow compressor the flow is decelerating (diffusing) in the passageways with a corresponding rise in pressure, whereas the gas is accelerating in a turbine. The diffusing action of a compressor allows only moderate changes in the compressor passageways to avoid separation.

The radial-flow turbine (Fig. 6.2b) is similar to a centrifugal-flow compressor except that flow is inward instead of outward.

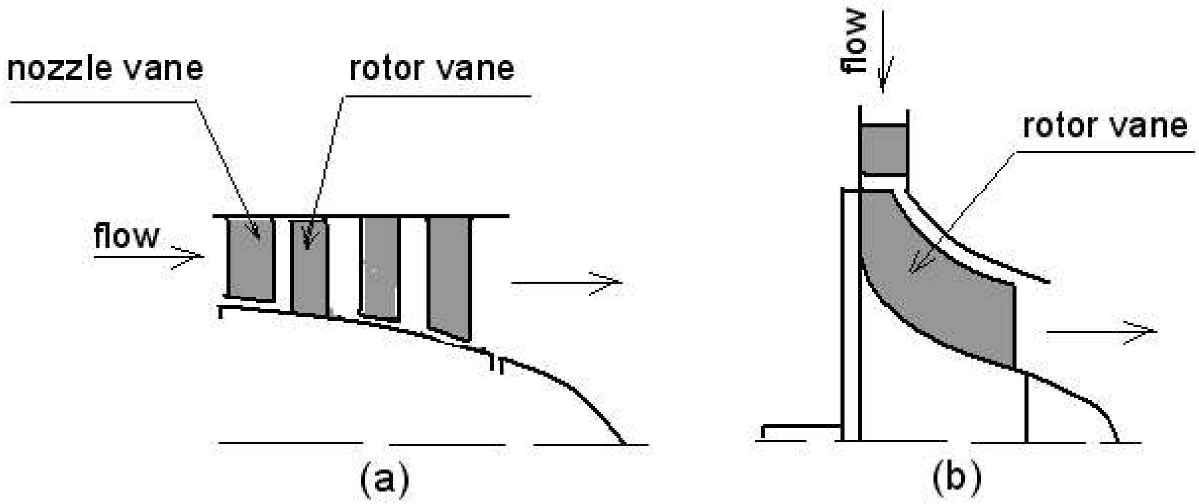

Figure 6.2. Flow path for axial-flow (a) and radial -flow (b) turbines

Radial-flow turbines are used only for extremely low powers or where compactness is more important than performance.

The problems of axial turbomachines design and operation are considered in the special course “Theory of Turbomachines”.

Topic 7 COMBUSTION CHAMBER

7.1 General

The combustion chamber (Fig. 7.1) has the difficult task of burning large quantities of fuel, supplied through the fuel spray nozzles, with extensive volumes of air, supplied by the compressor, and releasing the heat in such a manner that the air is expanded and accelerated to give a smooth stream of uniformly heated gas at all conditions required by the turbine. This task must be accomplished with the minimum loss in pressure and with the maximum heat release for the limited space available.

The amount of fuel added to the air will depend upon the temperature rise required. However, at present the maximum temperature is limited to within the range of 1125 to 1975 K by the materials from which the turbine blades and nozzles are made. The air has already been heated to between 475 and 825 K by the work done during compression. Since the gas temperature required at the turbine varies with engine thrust, and in the case of the turboprop engine upon the power required, the combustion chamber must also be capable of maintaining stable and efficient combustion over a wide range of engine operating conditions.

Efficient combustion has become increasingly important because of the rapid rise in commercial aircraft traffic and the consequent increase in atmospheric pollution, which is seen by the general public as exhaust smoke.

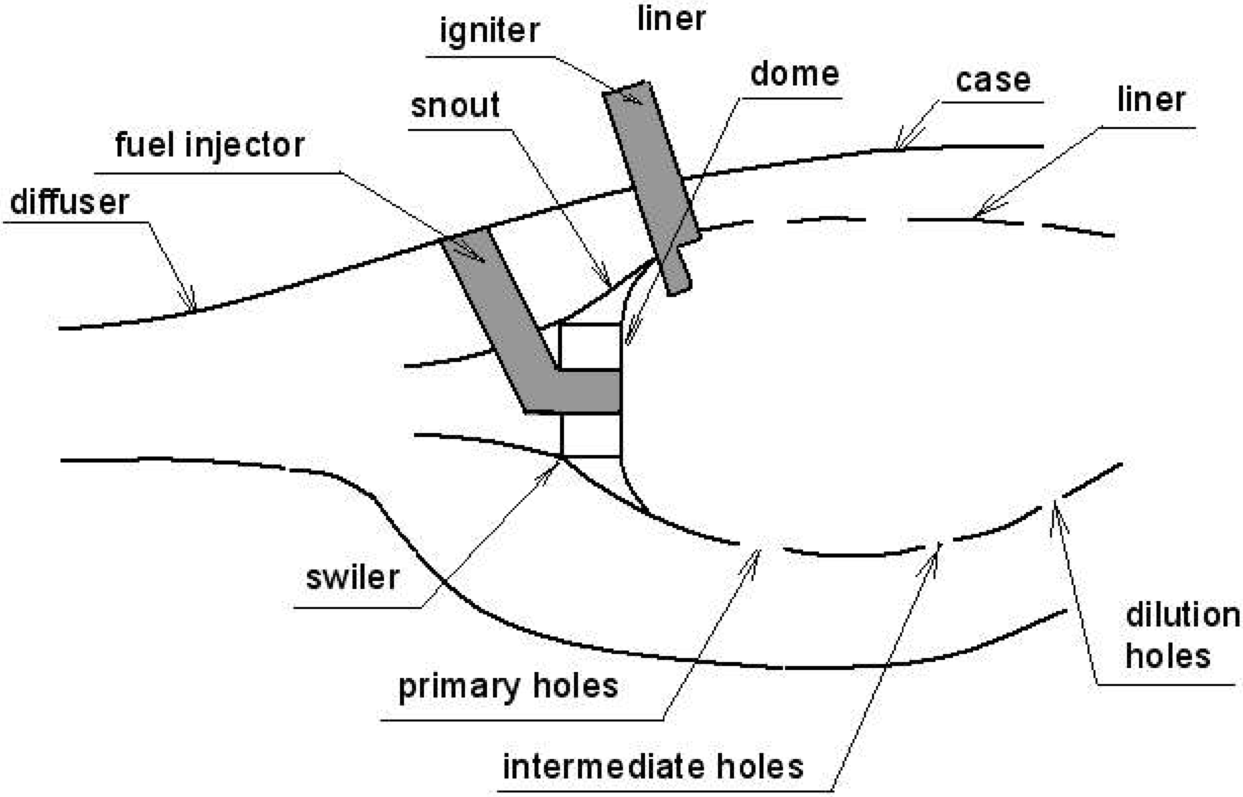

Figure 7.1. Combustion chamber

7.2 Combustion Process

Air from the engine compressor enters the combustion chamber at a velocity up to approx. 150 m/s, but because at this velocity the air speed is far too high for combustion, the first thing that the chamber must do to diffuse it, i.e. decelerate it and raise its static pressure. Since the speed of burning kerosene at normal mixture ratios is only a few meters per second, any fuel ignited even in the diffused air stream, which now has a velocity of about 25 m/s, would be blown away. A region of low axial velocity has therefore to be created in the chamber, so that the flame will remain alight throughout the range of engine operating conditions.

In normal operation, the overall air/fuel ratio of a combustion chamber can vary between 45:1 and 130:1. However, kerosene will only burn efficiently at, or close to, a ratio of 15:1, so the fuel must be burned with only part of the air entering the chamber, in what is called a primary combustion zone. This is achieved by means of a flame tube (combustion liner) that has various devices for metering the airflow distribution along the chamber.

Approximately 20% of the air mass flow is taken in by the snout or entry section. Immediately downstream of the snout are swirl vanes and a perforated flare, through which air passes into the primary combustion zone. The swirling air induces a flow upstream of the centre of the flame tube and promotes the desired recirculation. The air not picked up by the snout flows into the annular space between the flame tube and the air casing.

Through the wall of the flame tube body, adjacent to the combustion zone, are a selected number of secondary holes through which a further 20% of the main flow of air passes into the primary zone. The air from the swirl vanes and that from the secondary air holes interacts and creates a region of low velocity recirculation. This takes the form of a toroidal vortex, similar to a smoke ring, which has the effect of stabilizing and anchoring the flame. The recirculating gases hasten the burning of freshly injected fuel droplets by rapidly bringing them to ignition temperature.

Combustion chamber is arranged so that the conical fuel spray from the nozzle intersects the recirculation vortex at its centre. This action, together with the general turbulence in the primary zone, greatly assists in breaking up the fuel and mixing it with the incoming air.

The temperature of the gases released by combustion is about 2075 to 2275 K, which is far too hot for entry to the nozzle guide vanes of the turbine. The air not used for combustion, which amounts to about 60% of the total airflow, is therefore introduced progressively into the flame tube. Approximately a third of this is used to lower the gas temperature in the dilution zone before it enters the turbine and the remainder is used for cooling the walls of the flame tube. This is achieved by a film of cooling air flowing along the inside surface of the flame tube wall, insulating it from the hot combustion gases. A recent development allows cooling air to enter a network of passages within the flame tube wall before exiting to form an insulating film of air; this can reduce the required wall cooling airflow by up to 50%. Combustion should be completed before the dilution air enters the flame tube, otherwise the incoming air will cool the flame and incomplete combustion will result.

An electric spark from an igniter plug initiates combustion and the flame is then self-sustained.

The design of a combustion chamber and the method of adding the fuel may vary considerably, but the airflow distributions used to effect and maintain combustion is always very similar to that described.

7.3 Fuel Supply

Fuel is supplied to the air stream by one of two distinct methods. The most common is the injection of a fine atomized spray into the recirculating air stream through spray nozzles. The second method is based on the pre-vaporization of the fuel before it enters the combustion zone.

In the vaporizing method the fuel is sprayed from feed tubes into vaporizing tubes which are positioned inside the flame tube. These tubes turn the fuel through 180 degrees and, as they are heated by combustion, the fuel vaporizes before passing into the flame tube. The primary airflow passes down the vaporizing tubes with the fuel and also through holes in the flame tube entry section which provide "fans" of air to sweep the flame rearwards. Cooling and dilution air is metered into the flame tube in a manner similar to the atomizer flame tube.

7.4 Types of Combustion Chamber

There are three main types of combustion chamber (Fig. 7.2) in use for gas turbine engines. These are the multiple chamber, the tubo-annular chamber and the annular chamber.

Multiple combustion chamber

This type of combustion chamber is used on centrifugal compressor engines and the earlier types of axial flow compressor engines. It is a direct development of the early type of combustion chamber. The major difference is that chamber had a reverse but, as this created a considerable pressure loss, the straight-through multiple chambers was developed.

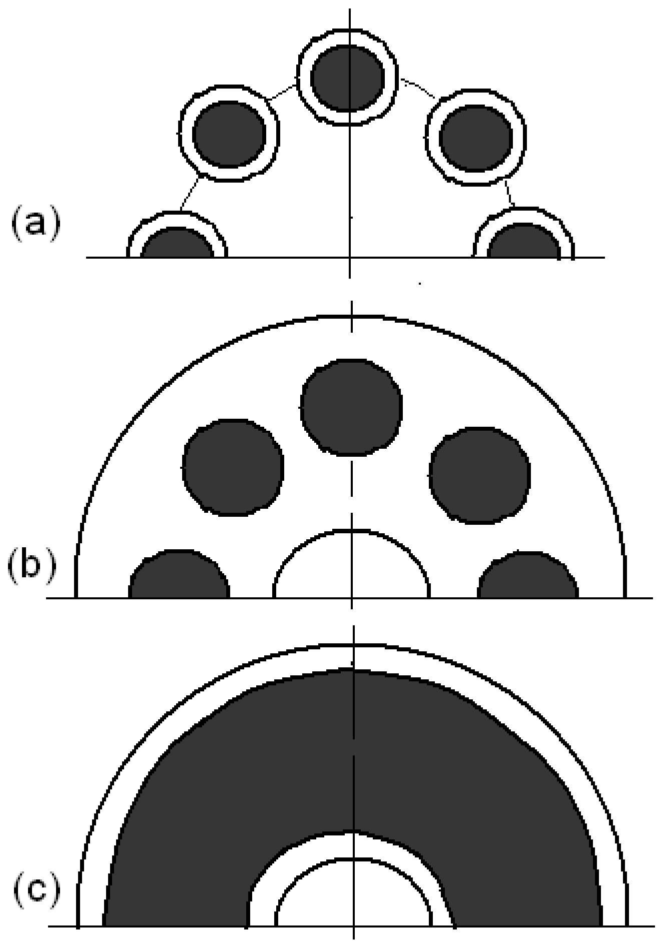

Figure 7.2. Types of combustion chamber

The chambers are disposed around the engine (Fig. 7.2a) and compressor delivery air is directed by ducts to pass into the individual chambers. Each chamber has an inner flame tube around which there is an air casing. The air passes through the flame tube snout and also between the tube and the outer casing.

The separate flame tubes are all interconnected. This allows each tube to operate at the same pressure and also allows combustion to propagate around the flame tubes during engine starting.

Tubo-annular combustion chamber

The tubo-annular combustion chamber bridges the evolutionary gap between the multiple and annular types. A number of flame tubes are fitted inside a common air casing (Fig. 7.2b). The airflow is similar to that already described. This arrangement combines the ease of overhaul and testing of the multiple systems with the compactness of the annular system.

Annular combustion chamber

This type of combustion chamber consists of a single flame tube, completely annular in form, which is contained in an inner and outer casing (Fig. 7.2c). The airflow through the flame tube is similar to that already described, the chamber being open at the front to the compressor and at the rear to the turbine nozzles.

The main advantage of the annular chamber is that, for the same power output, the length of the chamber is only 75% of that of a tubo-annular system of the same diameter, resulting in considerable saving of weight and production cost.

Another advantage is the elimination of combustion propagation problems from chamber to chamber.

In comparison with a tubo-annular combustion system, the wall area of a comparable annular chamber is much less; consequently the amount of cooling air required to prevent the burning of the flame tube wall is less, by approximately 15%. This reduction in cooling air raises the combustion efficiency to virtually eliminate unburnt fuel, and oxidizes the carbon monoxide to non-toxic carbon dioxide, thus reducing air pollution.

The introduction of the air spray type fuel spray nozzle to this type of combustion chamber also greatly improves the preparation of fuel for combustion by aerating the over-rich pockets of fuel vapors close to the spray nozzle; this results in a large reduction in initial carbon formation.

7.5 Combustion Chamber Performance

As it was mentioned, a combustion chamber must be capable of allowing fuel to burn efficiently over a wide range of operating conditions without incurring a large pressure loss. In addition, if flame extinction occurs, then it must be possible to relight. In performing these functions, the flame tube and spray nozzle atomizer components must be mechanically reliable.

The gas turbine engine operates on a constant pressure cycle; therefore any loss of pressure during the process of combustion must be kept to a minimum. In providing adequate turbulence and mixing, a total pressure loss varying from about 3 to 8% of the air pressure at entry to the chamber is incurred.

7.6 Combustion Properties

Combustion intensity

The heat released by a combustion chamber or any other heat generating unit is dependent on the volume of the combustion area. Thus, to obtain the required high power output, a comparatively small and compact gas turbine combustion chamber must release heat at exceptionally high rates.

For example, at take-off conditions a Rolls- Royce RB211 engine will consume 9 360 kg of fuel per hour. The fuel has a calorific value of approximately 43 000 kJ/kg, expressed in another way this is an expenditure of potential heat at a rate equivalent to approximately 112 000 kW.

Combustion efficiency

The combustion efficiency (see Chapter 4) of most gas turbine engines at sea-level take-off conditions is almost 100%, reducing to 98% at altitude cruise conditions.

Combustion stability

The combustion stability means smooth burning and the ability of the flame to remain alight over a wide operating range.

Range of air/fuel ratio

For any particular type of combustion chamber there is both a rich and weak limit to the air/fuel ratio, beyond which the flame is extinguished. Extinction is most likely to occur in flight during a glide or dive with the engine idling, when there is a high airflow and only a small fuel flow, i.e. very weak mixture strength.

The range of air/fuel ratio between the rich and weak limits is reduced with an increase of air velocity, and if the air mass flow is increased beyond a certain value, flame extinction occurs. The operating range defined by the stability loop must obviously cover the air/fuel ratios and mass flow of the combustion chamber.

The ignition process has weak and rich limits. The ignition loop, however, lies within the stability loop since it is more difficult to establish combustion “undercold” conditions than to maintain normal burning.

7.7 Materials

The containing walls and internal parts of the combustion chamber must be capable of resisting the very high gas temperature in the primary zone. In practice, this is achieved by using the best heat-resisting materials available, the use of high heat resistant coatings and by cooling the inner wall of the flame tube as insulation from the flame.

The combustion chamber must also withstand corrosion due to the products of the combustion, creep failure due to temperature gradients and fatigue due to vibration stresses.

Topic 8 ALTERNATIVE FUEL

8.1 General

The last hundred years have seen a rapid increase in global population. At the same time, per capita energy use has increased. The resultant overall increase in energy use is enormous. To date, energy has been provided mainly by combustion: the burning of wood, coal, oil, natural gas. The combustion of these fuels liberates CO_2, a “greenhouse gas”: 3 kg of CO_2 are produced for 1 kg of oil burned. As a result, the concentration of atmospheric CO_2 has been rising dramatically since the beginning of the industrial revolution.

Aircraft fuel (kerosene) is produced from crude oil. Proven crude oil reserves economically exploitable with current technology will last for another approximately forty – fifty years. Thereafter, kerosene made from alternative fossil energy sources will only be available at higher prices. Fossil energy reserves will inevitably be exhausted sooner or later, and we will be compelled to seek new fuels.

The earth’s atmosphere, absolutely vital to life, is a razor-thin layer. This sensitive atmosphere layer serves as the “rubbish dump” for power plants, industry, house fires, traffic… Although air traffic accounts only 3% of world energy use, and produces correspondingly few emissions, the high-tech aviation industry is taking up the challenge of working against further damage to our atmosphere.

8.2 Aircraft Fuel

Hydrogen (liquid LH_2 or gaseous) and liquid natural gas (LNG) are the most perspective power supplies now.

There are many advantages of hydrogen as an aviation fuel.

LH_2 easily exhales and fast diffuses on all volume of the combustion chamber, promoting a quick engine start. Minor energy and wide range of hydrogen-air mix ignition also promotes a quick engine start at different temperatures and altitudes. Hydrogen combustion gives a flame with low radiation and burns down without fouling, increasing safe life and reliability of engines. Hydrogen has a small corrosion activity. Engines on LH_2 don’t pollute environment. Hydrogen heat-absorption capacity is higher, than kerosene one in 30 times. So, hydrogen can be used in engine and aircraft cooling systems. Increasing of turbine cooling efficiency allows increasing the temperature at the turbine inlet, and a compressor pressure ratio. Other results are decreasing specific fuel consumption (by 15-20 %) and increasing a specific thrust of the engine. Smaller weight of aircraft reduces a wing load and a wing size. So noise in an airport will be decreased. LH_2 allows creating compact combustion chambers with more constant temperature field. Because of high hydrogen thermal capacity, temperature in a turbine inlet will be lower, etc.

Flight performances of LH_2 aircraft tend to optimization at Mach number about 6. So the more airplane weight and speed the more expediently hydrogen using. The majority disadvantages of LH_2 as an aviation fuel are its very low density (63-70 kg/м^3) and low boiling temperature (20 К). So, aircraft tanks should be large and have the configuration with minimized relation of a surface to a volume to minimize evaporation losses and isolation weight. Also, some materials become friable in LH_2.

Broadly, the problems connected with integrating the liquid hydrogen tanks are that hydrogen requires about four times as much volume as kerosene, and that the cryogenic temperatures require special, highly insulated pressure tanks. However, LH_2 has an advantage too: it weights only about one third as much as kerosene, which means either a lower take-off weight – or an increased pay load.

For volume considerations, it is absolutely necessary to keep hydrogen and methane fuels in the liquid state. However, both fuels reach this state only at cryogenic temperatures (liquid hydrogen: -253^{0}C; liquid methane: -160^{0}C).

8.3 Cryogenic Aircraft

There are four types of cryogenic engines.

1. Bifuel (hybrid) engines are created on the basis of existing engines. Only a hydrogen system is added. Such engine has two fuel systems and can run, using any available fuel (but not both fuels at the same time).

2. Cryogenic GT (Gas Turbines). Such engines are created for hydrogen use only, but kerosene GT is their prototype. The hydrogen system is similar to bifuel engines. Cryogenic GTE will be used with subsonic planes, and also hypersonic ones as a low-speed engine (take off–landing).

3. Hypersonic engines of external burning. Their fuel system is like cryogen GT one, but all the rest devices are completely different. The hypersonic engine has not rotating parts (compressor, turbine). Fuel goes directly on an aircraft surface, and burns down without initial ignition, because of high air temperature.

4. Rocket engines. Use hydrogen and oxygen from cryogenic tanks.

It will be possible to develop the engine for hydrogen-fueled aircraft on the base of existing engines. Most importantly, a new combustion chamber design will be necessary to achieve minimum emissions of oxides of nitrogen in operation with the new fuel. In particular, liquid hydrogen tanks on top of the fuselage offer the most favorable configuration for this aircraft type.

Materials currently used in cryogenics are too heavy to be used as tank materials. Therefore, new materials like fiber composites or special aluminum-lithium alloys are being considered. To date however, only little experience has been gained with these materials at cryogenic temperatures. Tanks and fuel ducts must be highly insulated to prevent evaporation loss.

8.4 Cryogenic Aviation History

There were early hydrogen aviation developments from 1950s (NASA, Lockheed, Boeing, Pratt & Whitney etc.). Turbojet J-57 (Pratt & Whitney) for the Lockheed reconnaissance, engine “304” (Pratt & Whitney), engine J-65 for

B-57 aircraft and engine J-85 for Space Shuttle Program, subsonic bomber "Canberra", hybrid power plant were designed and partially tested.

The higher result in the real hydrogen aviation was obtained in the Tupolev Corporation (Russia). "Hydrogen" aircraft was built and successfully tested without any serious incidents. It was preceded by a long-term program of bench and ground tests intended for testing functioning of new systems (such systems were more than 30 on the aircraft) and mainly for providing safe operation.

Tu-155 aircraft was built on the basis of serial Tu-154B aircraft.

To use cryogenic fuel airframe and some standard systems were modified, cryogenic fuel charging, storage and feeding systems were installed that ensured fire/explosion safety, and data acquisition and recording system as well.

Cryogenic fuel resource was kept in fuel tank of 17.5 m^3 capacity installed in special compartment in rear portion of passenger cabin. Experimental hydrogen–powered NK-88 engine is located in the rightside nacelle. Creation of the aircraft was accompanied by serious scientific and research works and elaboration of large amount of regulatory documentation.

15 April, 1988 the aircraft performed its maiden flight (21 min duration) using liquid hydrogen. Tu-155lab was the First Hydrogen Flying Test Laboratory with Test Hydrogen Turbofan Engine NK-88.

If to speak about near future tomorrow task is to introduce LNG as aviation fuel. That's why Tupolev flying laboratory having status of experimental Tu-155 aircraft was modified to use not only liquid hydrogen but also to use liquefied natural gas. This is how the first in the world cryogenic aircraft was built.

Natural gas is supplied to substantially each airfield via pipelines i.e. transportation issues have been practically solved now. Its high energy capacity, huge cooling capacity makes it possible to build aircraft with significantly high performance in comparison with aircraft using kerosene.

When using LNG potential emission of toxic agents will be decreased as follows: carbon monoxide – 1 – 10 times, hydrocarbons – 2.5 – 3 times, nitrogen oxides – 1.5 – 2 times, polycyclic aromatic hydrocarbons including benzopyrene – 10 times.

Upon flight testing and development 18 January, 1989 Tu-155 aircraft performed its first flight on liquefied natural gas. Large flight testing program was fulfilled, several international flight demonstrations were made including those to Bratislava (Slovakia), Nice (France), Berlin and Gannover (Germany) etc.

Appearance of Tu-155 aircraft changed dramatically scope of tasks for creation of cryogenic aviation. It was demonstrated in reality that using existing technical aids power plant has been built which allowed operating LH_2 or LNG-powered aircraft with the same safety level than those working on kerosene.

Experience of ensuring fire/explosion safety of cryogenic aircraft is unique. Principles and technical approaches that were developed when solving this problem (for example gas test system newly applied) will be used on all future cryogenic aircraft. The same is about power plant of the aircraft which main technical approaches are quite new. Engine scheme and cryogenic components, fuel pumps, pressure maintenance system and cryogenic fuel tanks - all of these could be utilized in future developments.

Topic 9 INLETS AND NOZZLES

9.1 Inlets

9.1.1 Subsonic Inlets

The purposes of any aircraft gas turbine engine inlet is to provide a sufficient air supply to the compressor with as low a loss in total pressure as possible and with as small a drag force on the airplane as possible. It also has the purpose of being a diffuser that reduces the velocity of the entering air as efficiently as possible.

It was mentioned earlier that there are a number of different types of air inlet ducts for aircraft turbine engines. With turboprop engines, the inlet design is complicated by the propeller and gear box at the inlet to the engine. Aircraft engines may be located under the wings of the aircraft, at the base of the vertical stabilizer, or in the fuselage of the aircraft with the inlet located in the root of the wing or under the fuselage. Each of these installations can cause problems associated with subsonic inlets, namely, distortion at the compressor inlet and total pressure losses. Inlets also may be classified as single-entrance, such as occurs with engines installed beneath the wings of an aircraft, or divided entrance, and such as occurs in fighter aircraft when the inlets are located at the roots of the aircraft wings. The divided-entrance configuration may lead to distortion (pressure and/or temperature variation) at the inlet to the engine.

Subsonic inlets are of fixed design, although inlets for some high by-pass ratio engines have been designed with blow-in doors, which are spring-loaded parts installed in the perimeter of the inlet duct designed to deliver additional air to the face of the gas turbine engine at high power output and low aircraft forward speed. The internal surface of a subsonic inlet is a diffusing section ahead of the compressor. The air pattern at the inlet at zero aircraft forward speed is shown in Fig. 9.1a (static operation), at low forward speed is shown in Fig. 9.1b, and for a high forward speed in Fig. 9.1c. It is essential that the inlet be designed so that boundary layer separation does not occur.

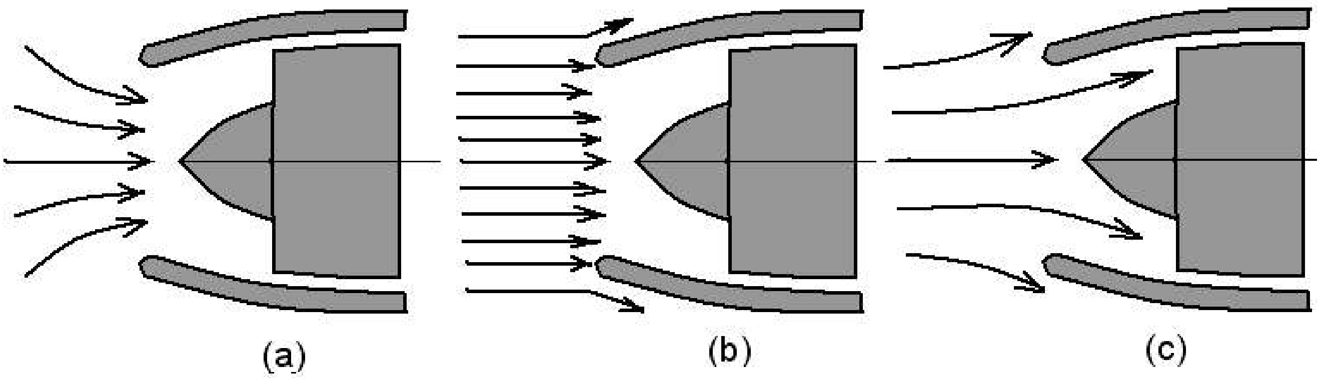

Figure 9.1. Air flow pattern for several forward speeds. (a) Static operation. (b) Low forward speed. (c) High forward speed

9.1.2 Supersonic Inlets

Supersonic inlets are much more difficult to design than are subsonic inlets. The inlet used on a supersonic aircraft is the design that optimizes performance for the mission for which the aircraft is designed. The inlet must provide adequate subsonic performance, good pressure distribution at the compressor inlet, high pressure recovery ratios, and must be able to operate efficiently at all ambient pressures and temperatures during take-off, subsonic flight, and at its supersonic design condition.

Effect of angle of attack on inlet air flow pattern is presented in Fig. 9.2.

Both axisymmetric and two-dimensional inlets have been designed and used. Variable geometry center bodies, which change the inlet geometry for better off-design operation, and boundary layer bleed have been incorporated into supersonic inlets.

The inlet performance characteristics that are used to assess the performance of supersonic inlets and that have the largest influence on aircraft performance and range are:

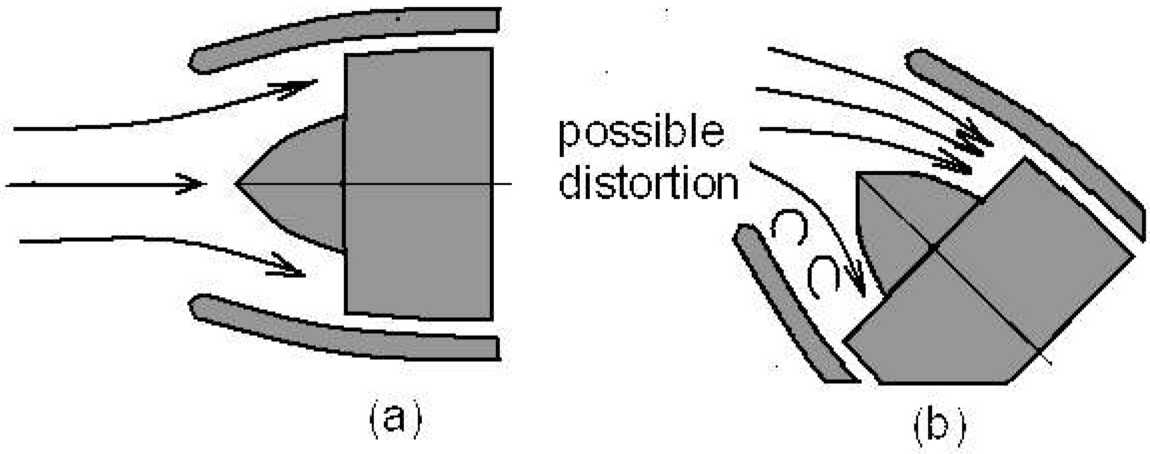

Figure 9.2. Effect of angle of attack on inlet air flow pattern.
(a) Normal operation. (b) High angle of attack

1. Total pressure recovery

2. Cowl drag

3. Boundary-layer bleed flow

4. Capture-area ratio (mass flow ratio)

5. Weight

Supersonic inlets are usually classified by their percent of internal compression. Internal compression refers to the amount of supersonic area change that takes place between the cowl lip and the throat. This is illustrated in Fig. 9.3, which identifies the center body, cowl lip, capture area, and throat for a supersonic inlet.

The total supersonic area change is the difference between the capture area and the minimum (throat) area. The area change that occurs in front of the cowl lip is called external compression; the amount of *supersonic* area change that occurs between the cowl lip and the throat is called internal compression. Inlets are usually classified as external compression inlets, mixed compression inlets, and internal compression inlets.

The *external* compression inlet completes the supersonic diffusion process outside the covered portion of the inlet. The normal shock where the flow changes from supersonic to subsonic and the throat are ideally located at the cowl lip is illustrated in Fig. 9.4.

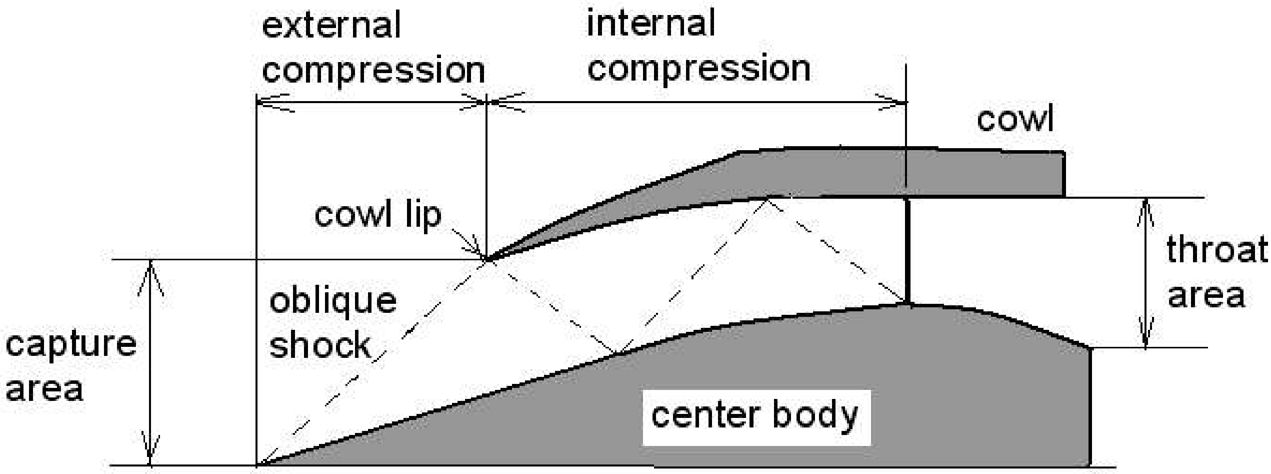

Figure 9.3. Supersonic inlet

The fixed external compression inlet is designed so that the oblique shock or shocks intersect the cowl lip as shown in Fig. 9.5. The normal shock is located at the cowl lip and is referred to as critical flow.

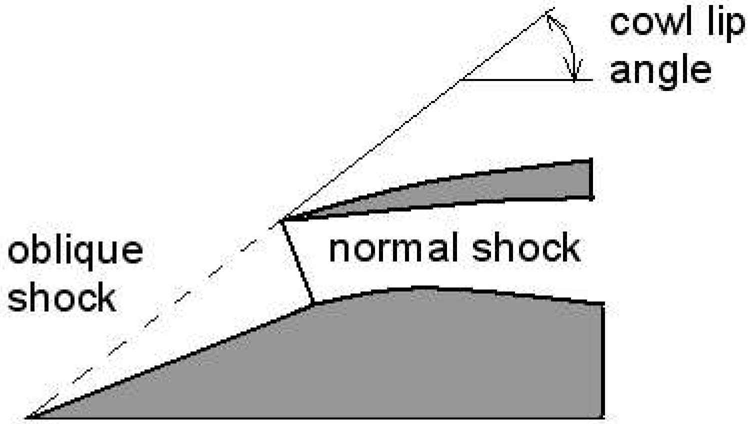

Figure 9.4. Supersonic inlet, all external compression

At air velocities below the design velocity or for high internal flow resistance, the normal shock occurs in front of the cowl lip (Fig. 9.6a). This is referred to as subcritical flow, and flow spillage occurs. At air velocities above design, the oblique shock may change as shown in Fig. 9.6b, with the shock impinging inside the cowl lip and the normal shock moving to the diverging section. This type of operation is referred to as supercritical operation. Ways to eliminate these problems are discussed later.

The fact is that as the Mach number of the air entering the supersonic diffuser decreases, the angle of the oblique shock (for a fixed wedge angle) changes. This means that for a fixed geometry and center body location, the capture area changes. This was illustrated in Fig. 9.6.

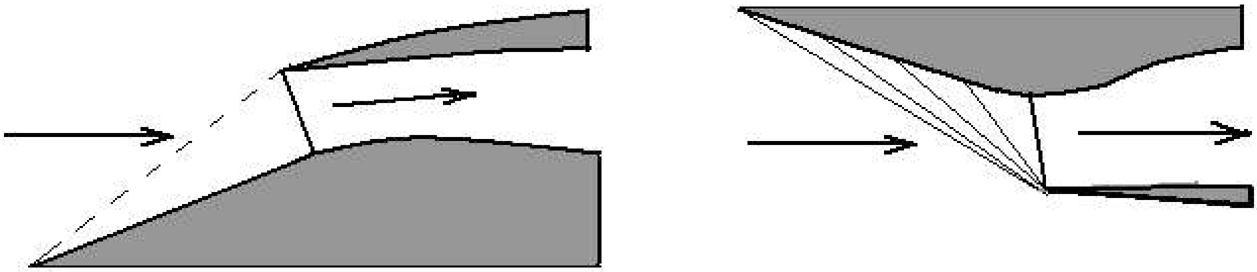

Figure 9.5. External compression inlets with critical flow

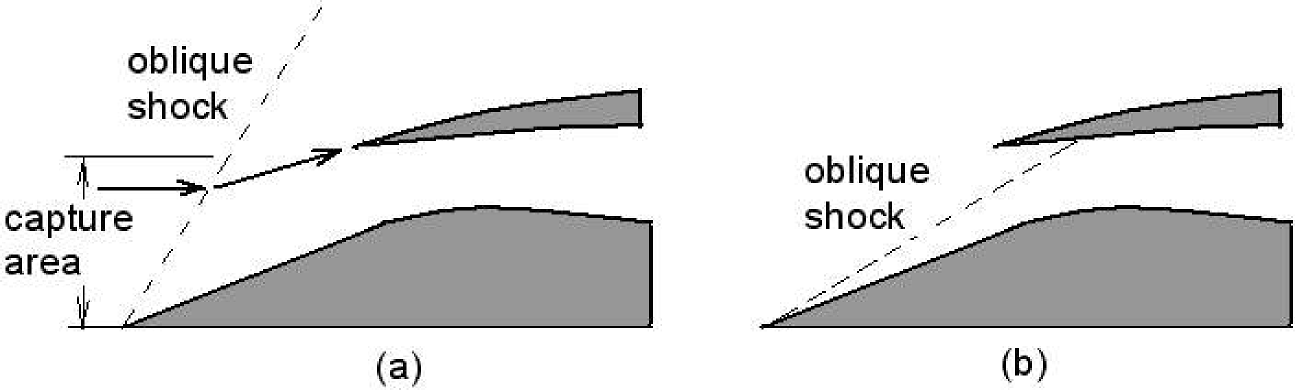

Figure 9.6. Subcritical and supercritical operation of a supersonic inlet.
(a) Subcritical operation. (b) Supercritical operation

If an external compression diffuser uses a single oblique shock followed by a normal shock, the ratio of the total pressure after the normal shock to the total pressure before the oblique shock is quite low.

The external compression inlet uses oblique shocks for a portion of the supersonic flow diffusion. The oblique shocks are followed by a normal shock across which the flow change from supersonic to subsonic.

The external compression inlet has a center body that extends forward of the cowl lip. The center body may be a cone or wedge that produces a single oblique shock as illustrated in Fig. 9.7a; a double cone or wedge that produces two oblique shocks that intersect at the cowl lip as illustrated in Fig. 9.7b; or a multiangle center body that produces a series of weak shocks as illustrated in Fig. 9.8. A diffuser that uses an infinite number of infinitesimal oblique shocks to reduce the supersonic flow to sonic flow would be an isentropic compression process and would be the most efficient.

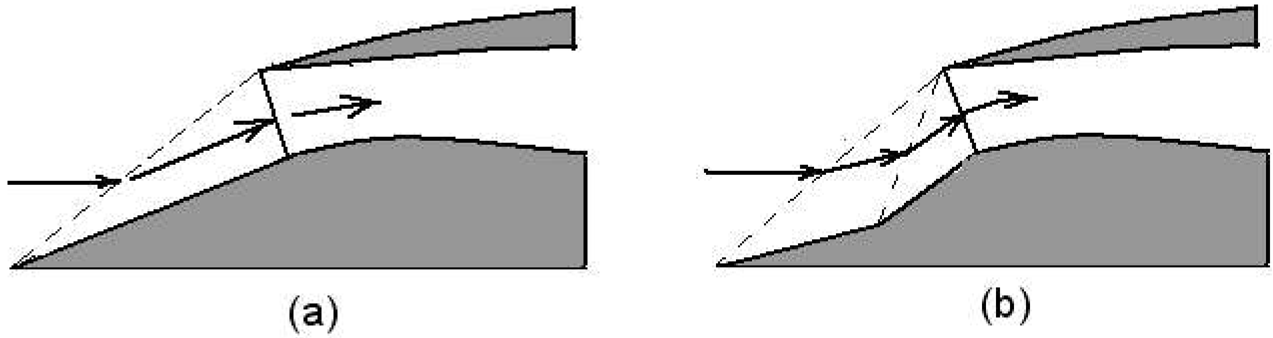

Figure 9.7. Single wedge (a) and double cone or wedge (b) external compression supersonic inlet

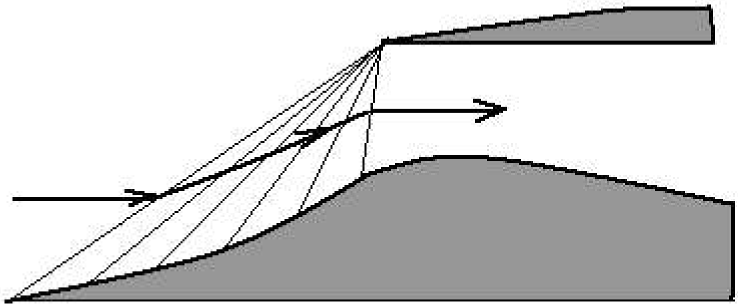

Figure 9.8. Isentropic compression supersonic inlet

The internal compression inlet locates all of the shocks within the covered passage-way as illustrated in Fig. 9.9. The terminal (normal) shock is usually located near the throat of the inlet. The inlet is said to be operating in the critical mode when the terminal shock is located at the throat, in subcritical operation when the normal shock is located upstream of the throat, and in supercritical operation when the terminal shock is located downstream of the throat.

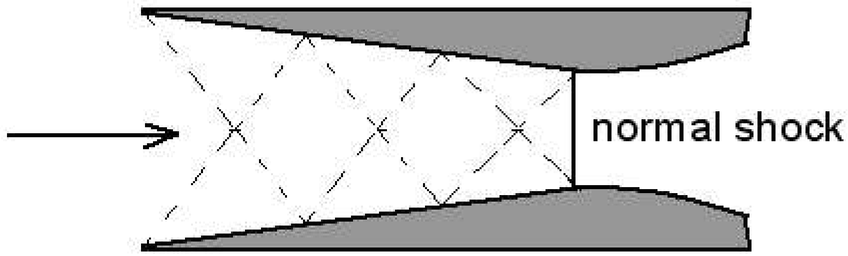

Figure 9.9. Internal compression supersonic inlet

The terminal shock, for subcritical operation, is located in the converging section of the inlet and is unstable. It will move ahead of the cowl lip, producing a condition called unstart. When this happens the pressure recovery of the inlet drops and flow spills over the cowl, producing a high drag and aircraft control problems. For this reason, supersonic inlets are designed to operate supercritically instead of critically. This is done to maintain a margin of stability in the event of a sudden change in the inlet flow. It must be remembered that the highest total pressure recovery occurs when the terminal shock is located at the throat.

The *mixed compression* inlet uses a combination of external and internal compression. This type of inlet is illustrated in Fig. 9.10.

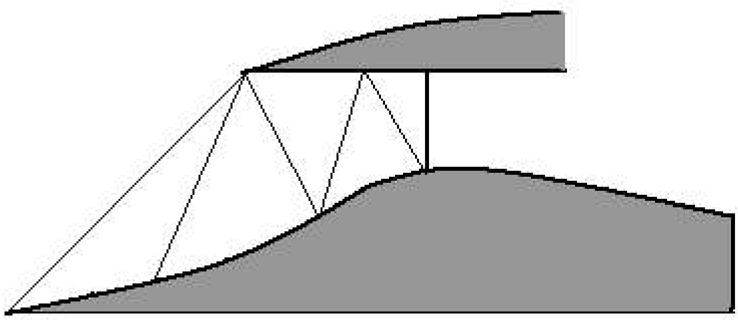

Figure 9.10. Mixed compression inlet

The type of inlet selected for an aircraft designed to operate at supersonic speeds depends on the aircraft mission – that is, what fraction of the flying time will be at supersonic speeds, the cruise Mach number, and cruise altitude.

Most, if not all; supersonic inlets currently in use are either external compression or mixed compression inlets.

Inlets are designed for one flight condition but must provide stable operation and an adequate match with the compressor during off-design operation. Several ways of controlling the inlet flow to provide stable operation and a proper match between the inlet and engine are to use variable geometry and boundary layer bleed and to incorporate flow by-pass into the inlet design.

Variable inlet geometry may involve the use of a translating center body, use of a cowl where the lip angle may be varied, variable ramp angles, and/or variable throat areas. Varying the geometry allows the inlet to operate with the optimum shock location, thereby providing optimum off-design performance. Using variable geometry inlets requires the use of sensors, which adds complexity and weight to the inlet.

Bleeding involves the extraction of air from the low-momentum boundary layer. This reduces the possibility of flow separation in the diffuser and improves engine matching. This is normally extracted at the points where the shock wave reflects from the wall.

By-pass (dump) doors are also used. They allow excess inlet air to spill in cases of engine throttling or shut-down.

9.2 Nozzles

9.2.1 Converging Nozzles

A schematic diagram of a converging nozzle is shown in Fig. 9.11a. The cross-sectional area at the inlet to the nozzle is assumed to be large so that the velocity is negligibly small. This means that the pressure and temperature at the inlet are the stagnation pressure and temperature, respectively. It is assumed that the inlet conditions are held constant and that the pressure in the discharge region may be varied.

If the pressure at the minimum cross-sectional area (throat of the nozzle) is equal the pressure in the discharge region, the pressure at all points in the nozzle is the same and there is no flow. As the discharge pressure is lowered slightly, flow occurs with the equality of discharge and throat pressures and the velocity increases from the inlet to the throat, reaching its maximum value at the throat.

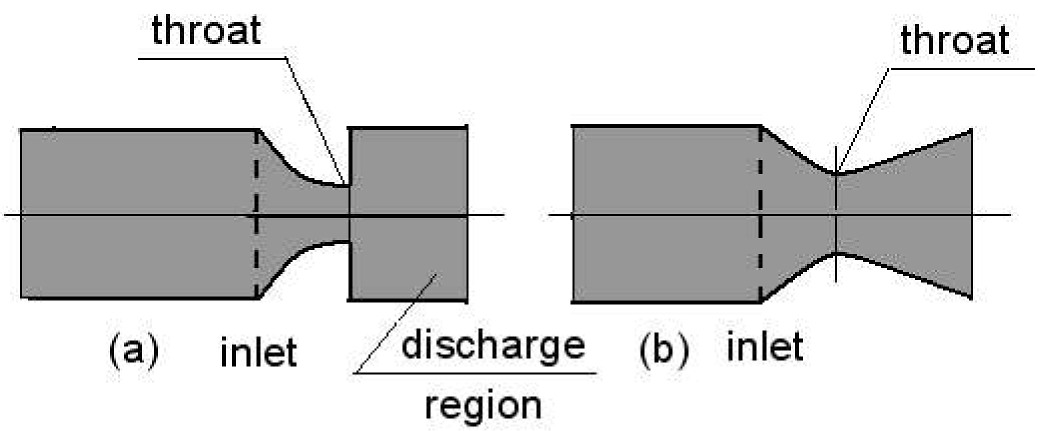

Figure 9.11. Converging (a) and converging- diverging (b) nozzles

9.2.2 Converging-Diverging Nozzles

In order to accelerate a fluid from a subsonic to a supersonic velocity, a diverging section must be added to the converging nozzle. This is shown in Fig. 9.11b.

For situations when the velocity at the throat does not reach sonic velocity, the diverging section acts as a diffuser and the velocity never reaches sonic velocity.

As the discharge region pressure is lowered, the velocity at the throat will reach sonic velocity. What happens in the discharge region depends on the discharge pressure.

At some point, theory predicts a normal shock occurring with a resulting increase in pressure and decrease in velocity, the velocity going from supersonic to subsonic across the normal shock. After the shock, the pressure increases and the velocity decreases with an increase in area.

9.2.3 Exhaust Nozzles

Designing an exhaust system for a subsonic commercial airplane usually involves use of a fixed-area converging nozzle, whereas designing an exhaust system for a supersonic airplane usually involves an exhaust system with variable geometry. The exhaust system selected for a supersonic aircraft is a compromise among weight, complexity, and performance.

Many types of nozzles have been used in aircraft designs. These include:

1. Fixed-area converging nozzles

2. Fixed-area converging-diverging nozzles

3. Variable-area converging-diverging nozzles

4. Plug nozzles

5. Two-dimensional nozzles

The exhaust nozzle on an aircraft gas turbine engine should:

1. Be matched to the other engine components for all engine operating conditions

2. Provide the optimum expansion ratio

3. Have minimum losses at both design and off-design conditions

4. Have low drag

5. Provide reverse thrust if necessary

6. Be able to incorporate noise-absorbing material

The simplest exhaust system is the converging nozzle with fixed area. This type of exhaust system has no moving parts, needs no control mechanism, and usually is used on subsonic commercial aircraft. Almost any smooth contour in the converging region will provide good performance because of the favorable pressure gradient in this region.

An additional thrust may be achieved if a converging-diverging nozzle is used. A fixed-area nozzle is designed for one expansion ratio and mass flow rate. For all other expansion ratios, the nozzle, will either overexpand or underexpand. A fixed-area converging-diverging nozzle adds weight, length, and possibly drags to the exhaust system, which may offset any additional thrust that would be achieved by using a fixed-area converging-diverging nozzle.

Many aircraft gas turbine engines, including all afterburning engines, require an exhaust system where the throat area of the nozzle may be varied. Some kind of iris mechanism with necessary controls must be used to achieve the area variation. One way of achieving this area variation is shown schematically in Fig. 9.12a.

Another type of nozzle that has been investigated extensively when a variable-area nozzle is required is the plug nozzle, which is shown schematically in Fig. 9.12b. The throat area, in a plug nozzle, may be varied by an axial translation of the plug, the outer casing or by an iris arrangement.

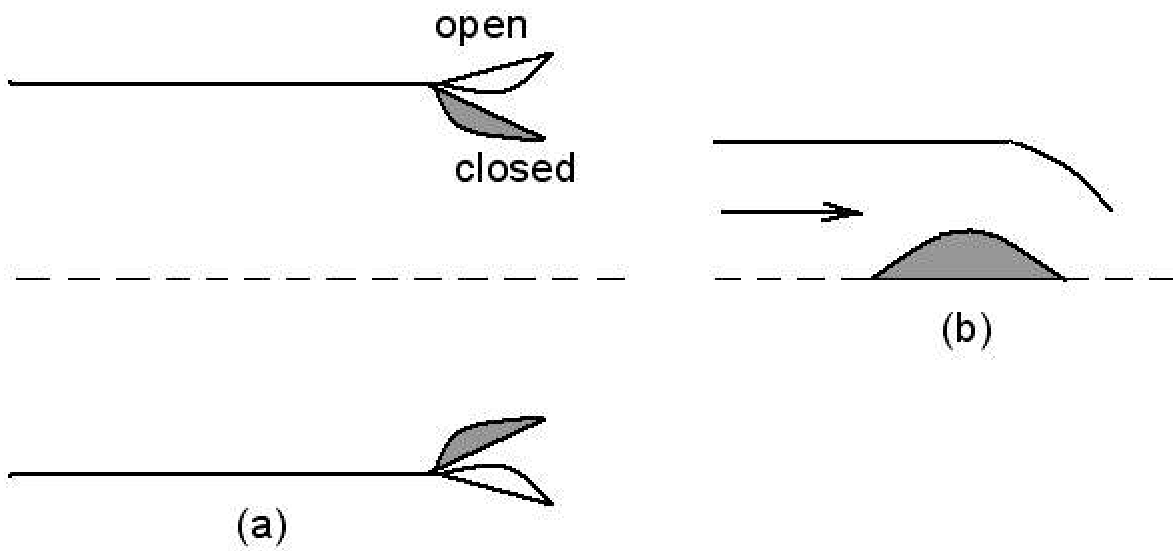

Figure 9.12. Schematic diagrams of a variable-area converging-diverging (a) and plug (b) nozzle

The major disadvantages of a plug nozzle are that it requires cooling and it is structurally weak.

The nozzles discussed above all are axisymmetric nozzles. Another type of nozzle that has been considered is the two-dimensional nozzle. Two-dimensional nozzles have the potential for reduced cruise drag and thrust vectoring for increased maneuver ability and are simpler to produce. Design problems inherent with two-dimensional nozzles are that they are heavier and have lower nozzle efficiency at subsonic velocities.

The choice of the exhaust system to be used in a given airplane requires an extensive analysis of the aircraft mission and compromises between minimum weight and maximum performance.

Topic 10 ENGINE PERFORMANCES

10.1 General

The performance requirements of an engine are obviously dictated to a large extent by the type of operation for which the engine is designed. The power of the turbojet engine is measured in thrust, produced at the propelling nozzle or nozzles, and that of the turboprop engine is measured in shaft power (SP) produced at the propeller shaft. However, both types are in the main assessed on the amount of thrust or SP they develop for a given weight, fuel consumption and frontal area.

Since the thrust or SP developed is dependent on the mass of air entering the engine and the acceleration imparted to it during the engine cycle, it is obviously influenced by such variables as the forward speed of the aircraft, altitude and climatic conditions, These variables influence the efficiency of the air intake, the compressor, the turbine and the jet pipe; consequently, the gas energy available for the production of thrust or SP also varies.

In the interest of fuel economy and aircraft range, the ratio of fuel consumption to thrust or SP should be as low as possible. This ratio, known as the above mentioned specific fuel consumption (SFC), is determined by the thermal and propulsion efficiency of the engine. In recent years considerable progress has been made in reducing SFC and weight.

Whereas the thermal efficiency is often referred to as the internal efficiency of the engine, the propulsion efficiency is referred to as the external efficiency. This latter efficiency explains why the pure jet engine is less efficient than the turboprop engine at lower aircraft speeds leading to development of the by-pass principle and, more recently, the “propfan” designs.

The thermal and the propulsion efficiency also influence, to a large extent, the size of the compressor and turbine, thus determining the weight and diameter of the engine for a given output.

These and other factors are presented in curves and graphs, calculated from the basic gas laws, and are proved in practice by bench and flight testing, or by simulating flight conditions in a high altitude test cell. To make these calculations, specific symbols are used (Fig. 10.1.) to denote the pressures and temperatures at various locations through the engine:

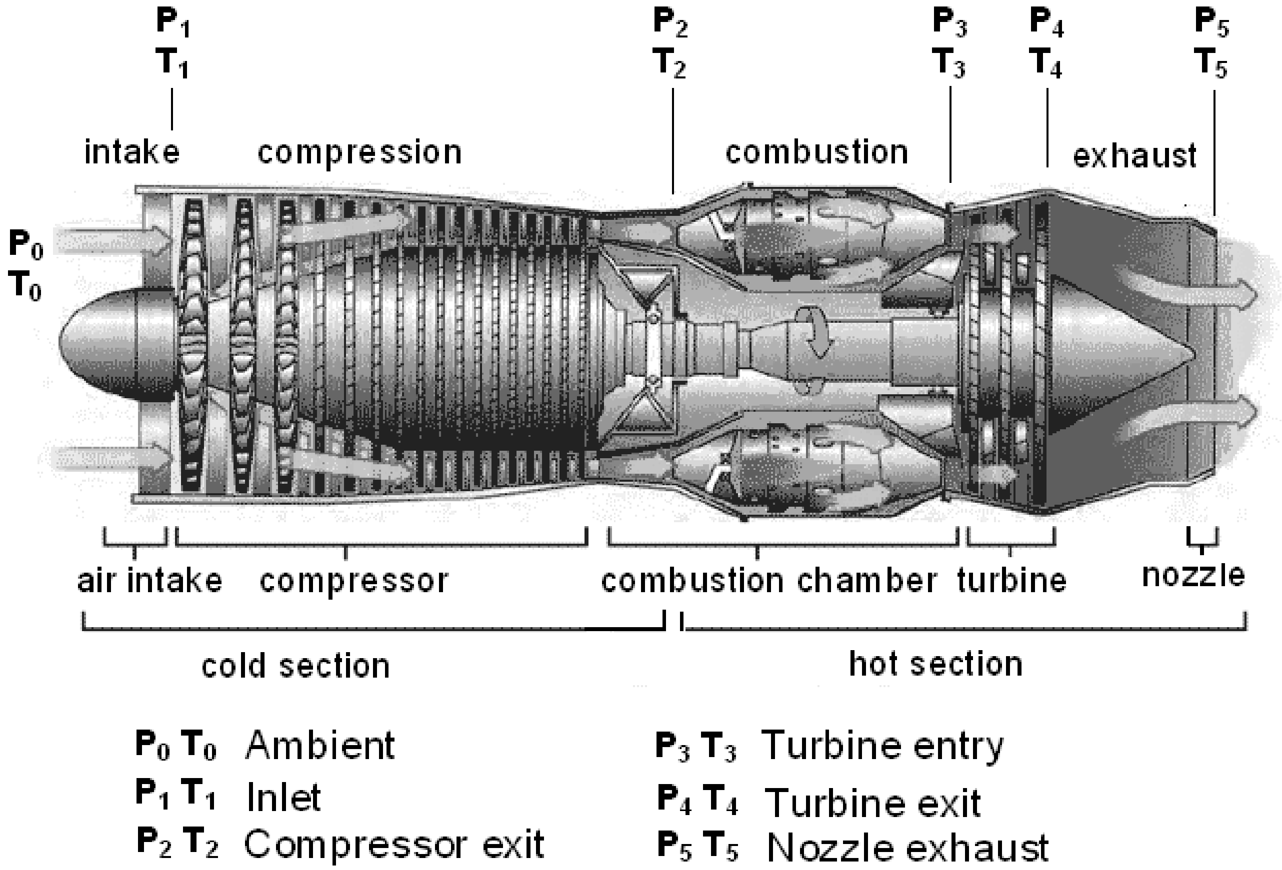

Figure 10.1. Test cross-section notations

For instance, using these symbols shown the overall compressor pressure ratio is $\boldsymbol{p_2/p_1}$. These symbols vary slightly for different types of engine; for instance, with high by-pass ratio engines, and also when afterburning is incorporated, additional symbols are used.

To enable the performance of similar engines to be compared, it is necessary to standardize in some conventional form the variations of air temperature and pressure that occur with altitude and climatic conditions. There are in use several different definitions of standard atmospheres, the one in most common use being the International Standard Atmosphere (I.S.A.). This is based on a temperature lapse rate of approximately 1.98 K per 300 m, resulting in a fall from 288.15 K (15^0C) at sea level to 216.65 K (-56.5^0C) at 11 000 m (the tropopause). Above this altitude the temperature is constant up to 20 000 m.

The I.S.A. standard pressure at sea level is 101 kPa falling to 23 kPa at the tropopause.

10.2 Propulsion Efficiency vs Aircraft Speed

Performance of the jet engine is not only concerned with the thrust produced, but also with the efficient conversion of the heat energy of the fuel into kinetic energy, as represented by the jet velocity, and the best use of this velocity to propel the aircraft forward, i.e. the efficiency of the propulsion system.

The efficiency of conversion of fuel energy to kinetic energy is termed thermal or internal efficiency and, like all heat engines, is controlled by the cycle pressure ratio and combustion temperature. Unfortunately, this temperature is limited by the thermal and mechanical stresses that can be tolerated by the turbine. The development of new materials and techniques to minimize these limitations is continually being pursued.

The efficiency of conversion of kinetic energy to propulsion work is termed the propulsion or external efficiency and this is affected by the amount of kinetic energy wasted by the propelling mechanism.

Assuming an aircraft speed (***V***) of 600 km/h and a jet velocity ($\boldsymbol{V_j}$) of 1980 km/h, the efficiency of a turbojet is:

$$\frac{2x600}{600+1980} = approx. 47\ per\ cent\ .$$

On the other hand, at an aircraft speed of 965 km/h the efficiency is:

$$\frac{2x965}{965+1980} = approx. 66\ per\ cent\ .$$

Propeller efficiency at these values of ***V*** is approximately 82 and 55 per cent, respectively, and from reference to Fig.10.2 it can be seen that for aircraft designed to operate at sea level speeds below approximately 645 km/h it is more effective to absorb the power developed in the jet engine by gearing it to a propeller instead of using it directly in the form of a pure jet stream. The disadvantage of the propeller at the higher aircraft speeds is its rapid fall off in efficiency, due to shock waves created around the propeller as the blade tip speed approaches Mach 1.0. Advanced propeller technology, however, has produced a multi-bladed, swept back design capable of turning with tip speeds in excess of Mach 1.0 without loss of propeller efficiency. By using this design of propeller in a contra-rotating configuration, thereby reducing swirl losses, a "propfan" engine, with very good propulsion efficiency capable of operating efficiently at aircraft speeds in excess of 805 km/h at sea level, can be produced.

To obtain good propulsion efficiencies without the use of a complex propeller system, the by-pass principle is used in various forms. With this principle, some part of the total output is provided by a jet stream other than that which passes through the engine cycle and this is energized by a fan or a varying number of LP compressor stages. This by-pass air is used to lower the mean jet temperature and velocity either by exhausting through a separate propelling nozzle, or by mixing with the turbine stream to exhaust through a common nozzle.

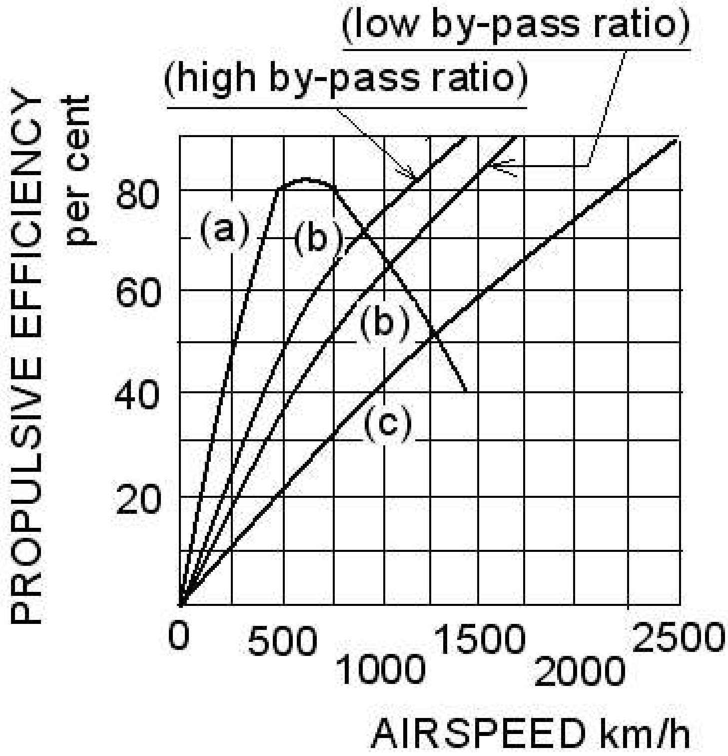

Figure 10.2. Propulsion efficiencies vs. aircraft speed. (a) Turboprop (b) Turbofan (c) Pure turbojet

The propulsion efficiency for a high by-pass ratio engine exhausting through separate nozzles is given below.

By calculation, substituting the following values, which will be typical of a high by-pass ratio engine of triple-spool configuration, it will be observed that a propulsion efficiency of approximately 85 per cent results:

V = 938 km/h; G_1 = 223 kg/s; G_2 = 45 kg/s; V_{j1} = 1 257 km/h;
V_{j2} = 1 307 km/h.

Propulsion efficiency can be further improved by using the rear mounted contra-rotating fan configuration of the by-pass principle. This gives very high by-pass ratios in the order of 15:1, and reduced "drag" results due to the engine core being “washed” by the low velocity aircraft slipstream and not the relatively high velocity fan efflux.

The improved propulsion efficiency of the by-pass system bridges the efficiency gap between the turboprop engine and the pure turbojet engine.

10.3 Fuel Consumption and Power-to-Weight Relationship

Primary engine design considerations, particularly for commercial transport duty, are those of low specific fuel consumption and weight. Considerable improvement has been achieved by use of the by-pass principle, and by advanced mechanical and aerodynamic features, and the use of improved materials. With the trend towards higher by-pass ratios, in the range of 15:1, the triple-spool and contra-rotating rear fan engines allow the pressure and by-pass ratios to be achieved with short rotors, using fewer compressor stages, resulting in a lighter and more compact engine.

SFC is directly related to the thermal and propulsion efficiencies; that is, the overall efficiency of the engine. Theoretically, high thermal efficiency requires high pressures which in practice also mean high turbine entry temperatures. In a pure turbojet engine this high temperature would result in a high jet velocity and consequently lower the propulsion efficiency. However, by using the by-pass principle, high thermal and propulsion efficiencies can be effectively combined by by-passing a proportion of the LP compressor or fan delivery air to lower the mean jet temperature and velocity. With advanced technology engines of high by-pass and overall pressure ratios, a further pronounced improvement in SFC is obtained.

The turbines of pure jet engines are heavy because they deal with the total airflow, whereas the turbines of by-pass engines deal only with part of the flow; thus the HP compressor, combustion chambers and turbines, can be scaled down. The increased power per kg of air at the turbines, to take advantage of their full capacity, is obtained by the increase in pressure ratio and turbine entry temperature. It is clear that the by-pass engine is lighter, because not only has the diameter of the high pressure rotating assemblies been reduced but the engine is shorter for a given power output. With a low by-pass ratio engine, the weight reduction compared with a pure jet engine is in the order of 20 per cent for the same air mass flow.

With a high by-pass ratio engine of the triple- spool configuration, a further significant improvement in specific weight is obtained. This is derived mainly from advanced mechanical and aerodynamic design, which in addition to permitting a significant reduction in the total number of parts, enables rotating assemblies to be more effectively matched and to work closer to optimum conditions, thus minimizing the number of compressor and turbine stages for a given duty. The use of higher strength light-weight materials is also a contributory factor.

For a given mass flow less thrust is produced by the by-pass engine due to the lower exit velocity. Thus, to obtain the same thrust, the by-pass engine must be scaled to pass a larger total mass airflow than the pure turbojet engine. The weight of the engine, however, is still less because of the reduced size of the HP section of the engine. Therefore, in addition to the reduced specific fuel consumption, an improvement in the power-to-weight ratio is obtained.

Topic 11 EMISSIONS AND NOISE

11.1 General

One last area concerning the gas turbine engine will be considered, this being a brief examination of gas turbine air and noise emissions and engine modifications that can be made to reduce the quantity of air emissions and noise emitted by a gas turbine engine.

Air pollution episodes and problems existed long before any ambient levels or gas turbine exhaust stream measurements were conducted. Most of the early air pollution episodes were associated with particulate and sulfur dioxide emissions and occurred in valleys or during atmospheric inversions in highly industrialized areas.

Engine emissions may vary for a number of reasons. Manufacturing differences, aging characteristics for individual engines, operational and atmospheric conditions, and changes in fuel contents all have parts to play.

Aircraft noise, since the introduction of jet-powered commercial airplanes, has been of concern. A continuous effort has been made to develop the technology to design a quiet gas turbine engine. One indication of the public concern about aircraft noise is the large number of major airports around the world that have noise restrictions in the various forms. The major problem with aircraft noise, in terms of number of people exposed and the frequency, with which they are exposed, occurs in the vicinity of airports.

11.2 Emissions

11.2.1 Aircraft Emissions

There would not appear to be any practical alternatives to kerosene-based fuels for commercial jet aircraft for the next several decades. The unwanted pollutants which are found in the gas turbine exhaust gases are created within the combustion chamber.

Of the pollutants generated in any combustion process, only carbon monoxide (CO), hydrocarbons (HC), nitrogen oxides (NO_x), and smoke have created the most concern in aircraft gas turbine engines and for which emission standards have been issued.

The ways of expressing gaseous emissions include:

1. Pollutant concentration.

2. Ratio of mass of pollutant emitted to mass of fuel consumed.

3. Total mass of pollutants emitted over the landing-takeoff cycle.

Nitric oxide is the only mainly formed in the combustion chamber. Factors that influence the amount of NO formed are:

1. Maximum temperature.

2. Percent excess air.

3. Pressure.

4. Time at the maximum temperature.

5. Fuel-bound nitrogen.

11.2.2. Aircraft Emission Reduction

General combustor characteristics showed that virtually all of the total HC and CO emissions are generated at low power, primary at engine idle, and that NO_x emissions are lowest at engine idle and increase as the gas turbine power increases. To reduce NO_x emissions, the maximum flame temperature must be reduced and/or the residence time of the combustion gases at the maximum temperature must be decreased.

The maximum temperature occurs when the fuel is burned with the stoichiometric (theoretical) amount of air. The higher the temperature of the air at the inlet to the combustion chamber the higher the resulting equilibrium adiabatic flame temperature.

Burning the fuel with excess air lowers the maximum temperature but increases the availability of oxygen and nitrogen in the products of combustion.

Increasing the combustion chamber pressure increases the equilibrium adiabatic flame temperature but decreases the amount of NO formed.

There would not appear to be any practical alternatives to kerosene-based fuels for a long time. Other fuel options, such as hydrogen, may be viable in the long term, but would require new aircraft designs and new infrastructure for supply. Hydrogen fuel would eliminate emissions of carbon dioxide from aircraft, but would increase those of water vapor. The overall environmental impacts and the environmental sustainability of the production and use of hydrogen or any other alternative fuels have not been determined.

11.3 Noise

11.3.1 Engine Noise

Airport regulations and aircraft noise certification requirements, all of which govern the maximum noise level aircraft are permitted to produce, have made jet engine noise suppression one of the most important fields of research.

The principal noise source is the engine.

To understand the problem of engine noise suppression, it is necessary to have a working knowledge of the noise sources and their relative importance. The significant sources originate in the fan or compressor, the turbine and the exhaust jet or jets. These noise sources obey different laws and mechanisms of generation, but all increase, to a varying degree, with greater relative airflow velocity. Exhaust jet noise varies by a larger factor than the compressor or turbine noise, therefore a reduction in exhaust jet velocity has a stronger influence than an equivalent reduction in compressor and turbine blade speeds.

Jet exhaust noise is caused by the violent and hence extremely turbulent mixing of the exhaust gases with the atmosphere and is influenced by the shearing action caused by the relative speed between the exhaust jet and the atmosphere. The small eddies created near the exhaust duct cause high frequency noise but downstream of the exhaust jet the larger eddies create low frequency noise. Additionally, when the exhaust jet velocity exceeds the local speed of sound, a regular shock pattern is formed within the exhaust jet core. A reduction in noise level occurs if the mixing rate is accelerated or if the velocity of the exhaust jet relative to the atmosphere is reduced. This can be achieved by changing the pattern of the exhaust jet.

Compressor and turbine noise results from the interaction of pressure fields and turbulent wakes from rotating blades and stationary vanes, and can be defined as two distinct types of noise; discrete tone (single frequency) and broadband (a wide range of frequencies). Discrete tones are produced by the regular passage of blade wakes over the stages downstream causing a series of tones and harmonics from each stage. The wake intensity is largely dependent upon the distance between the rows of blades and vanes. If the distance is short then there is an intense pressure field interaction which results in a strong tone being generated. With the high by-pass engine, the low pressure compressor (fan) blade wakes passing over downstream vanes produce such tones, but of a lower intensity due to lower velocities and larger blade/vane separations.

Broadband noise is produced by the reaction of each blade to the passage of air over its surface, even with a smooth airstream. Turbulence in the airstream passing over the blades increases the intensity of the broadband noise and can also induce tones.

With the pure jet engine the exhaust jet noise is of such a high level that the turbine and compressor noise is insignificant at all operating conditions, except low landing-approach thrusts. With the by-pass principle, the exhaust jet noise drops as the velocity of the exhaust is reduced but the low pressure compressor and turbine noise increases due to the greater internal power handling.

The introduction of a single stage low pressure compressor (fan) significantly reduces the compressor noise because the overall turbulence and interaction levels are diminished. When the by-pass ratio is in excess of approximately 5:1, the jet exhaust noise has reduced to such a level that the increased internal noise source is predominant.

Listed amongst the several other sources of noise within the engine is the combustion chamber. It is a significant but not a predominant source, due in part to the fact that it is 'buried' in the core of the engine. Nevertheless it contributes to the broadband noise, as a result of the violent activities which occur within the combustion chamber.

11.3.2 Methods of Suppressing Noise

Noise suppression of internal sources is approached in two ways; by basic design to minimize noise originating within or propagating from the engine, and by the use of acoustically absorbent linings. Noise can be minimized by reducing airflow disruption which causes turbulence. This is achieved by using minimal rotational and airflow velocities and reducing the wake intensity by appropriate spacing between the blades and vanes. The ratio between the number of rotating blades and stationary vanes can also be advantageously employed to contain noise within the engine.

As previously described, the major source of noise on the pure jet engine and low by-pass engine (Fig. 11.1) is the exhaust jet, and this can be reduced by inducing a rapid or shorter mixing region. This reduces the low frequency noise but may increase the high frequency level. Fortunately, high frequencies are quickly absorbed in the atmosphere and some of the noise which does propagate to the listener is beyond the audible range, thus giving the perception of a quieter engine. This is achieved by increasing the contact area of the atmosphere with the exhaust gas stream by using a propelling nozzle incorporating a corrugated or lobe-type noise suppressor.

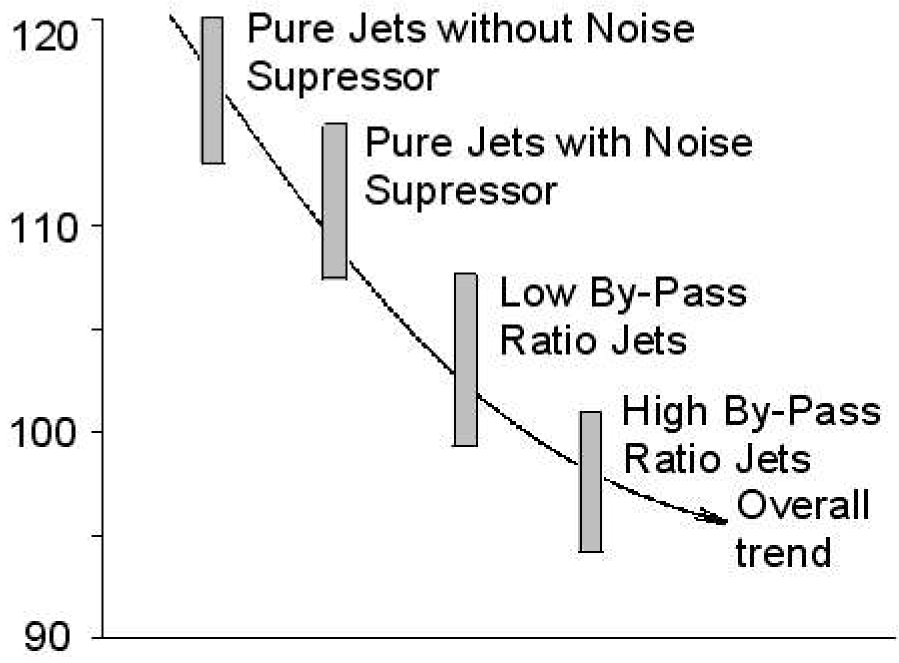

Figure 11.1. Level of noise in jet engines

In the corrugated nozzle, freestream atmospheric air flows down the outside corrugations and into the exhaust jet to promote rapid mixing. In the lobe-type nozzle, the exhaust gases are divided to flow through the lobes and a small central nozzle. This forms a number of separate exhaust jets that rapidly mix with the air entrained by the suppressor lobes. This principle can be extended by the use of a series of tubes to give the same overall area as the basic circular nozzle.

Deep corrugations, lobes, or multi-tubes, give the largest noise reductions, but the performance penalties incurred limit the depth of the corrugations or lobes and the number of tubes. For instance, to achieve the required nozzle area, the overall diameter of the suppressor may have to be increased by so much that excessive drag and weight results. A compromise which gives a noticeable reduction in noise level with the least sacrifice of engine thrust, fuel consumption or addition of weight is therefore the designer's aim.

The high by-pass engine has two exhaust streams to eject to atmosphere. However, the principle of jet exhaust noise reduction is the same as for the pure or low by-pass engine, i.e. minimize the exhaust jet velocity within overall performance objectives. High by-pass engines inherently have a lower exhaust jet velocity than any other type of gas turbine, thus leading to a quieter engine, but further noise reduction is often desirable. The most successful method used on by-pass engines is to mix the hot and cold exhaust streams within the confines of the engine and expel the lower velocity exhaust gas flow through a single nozzle.

In the high by-pass ratio engine the predominant sources governing the overall noise level are the fan and turbine. Research has produced a good understanding of the mechanisms of noise generation and comprehensive noise design rules exist. As previously indicated, these are founded on the need to minimize turbulence levels in the airflow, reduce the strength of interactions between rotating blades and stationary vanes, and the optimum use of acoustically absorbent linings.

Noise absorbing ‘lining’ material converts acoustic energy into heat. The absorbent linings normally consist of a porous skin supported by a honeycomb backing, to provide the required separation between the facesheet and the solid engine duct. The acoustic properties of the skin and the liner depth are carefully matched to the character of the noise, for optimum suppression. The disadvantage of liners is the slight increase in weight and skin friction and hence a slight increase in fuel consumption. They do however; provide a very powerful suppression technique.

Topic 12 SHAFT-POWER GAS TURBINES

12.1 General

A gas turbine is a rotary engine that extracts energy from a flow of combustion gas. Energy is extracted in the form of shaft power, compressed air and thrust, in any combination, and used to power aircraft, trains, ships, generators, and even tanks.

Gas turbine industrial applications are presented below.

12.2 Gas Turbine with Regenerator

It is important to investigate ways of improving the thermal efficiency and/or the net work of the gas turbine. The single improvement that gives the greatest increase in the thermal efficiency is the addition of a device for transferring energy (a heat exchanger) from the hot turbine exhaust gas to the air leaving the compressor. Fig. 12.1 illustrates the regenerative gas turbine power plant.

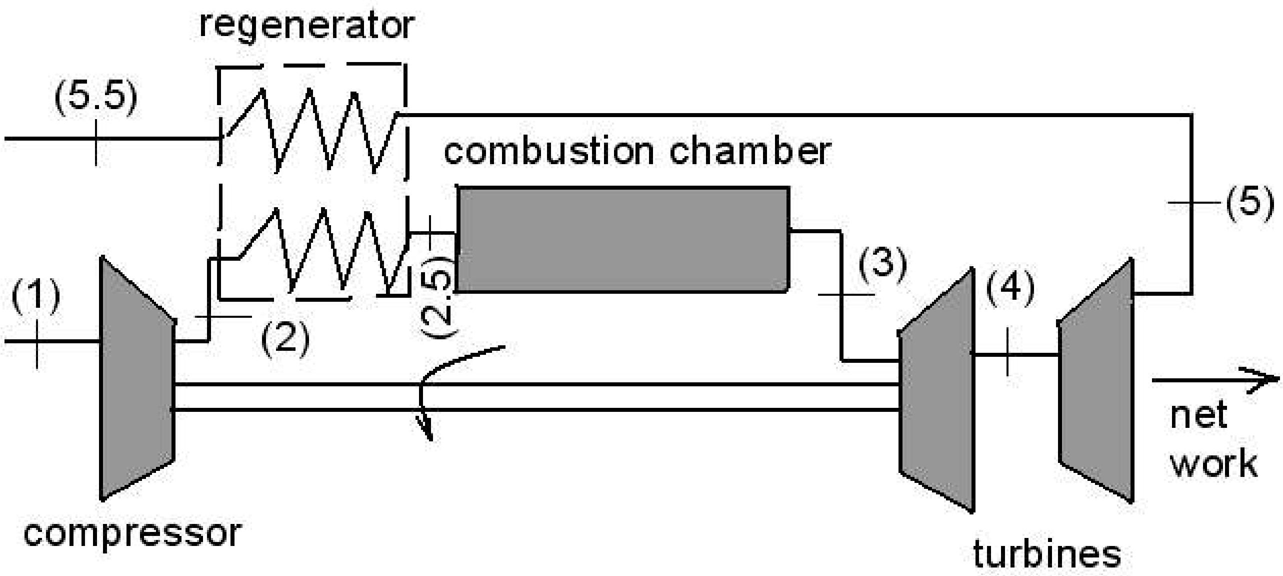

Figure 12.1. Regenerative gas turbine power plant

There two main types of regenerators currently being used, the recuperator and the rotating matrix regenerator. The two most commonly used configurations of rotating regenerators are the disk and drum types.

When comparing the two types of heat exchangers mentioned above, it must be remembered that each has its own advantages and disadvantages. One must remember that the increased cycle efficiency that can be gained through the use of a regenerator or recuperator must always be weighed against the disadvantages of increased service problems, cost, size, and weight.

Ideally, there is no pressure drop in the regenerator. Fig. 12.2 shows the increase in the temperature of the high-pressure air and the resulting decrease in the temperature of the low-pressure gas; all the energy removed from the low-pressure gas is transferred to the high pressure.

The regenerator effectiveness is defined as

$$\eta_{reg} = \text{Actual heat transfer/Maximum heat that can be transferred.} \qquad (12.1)$$

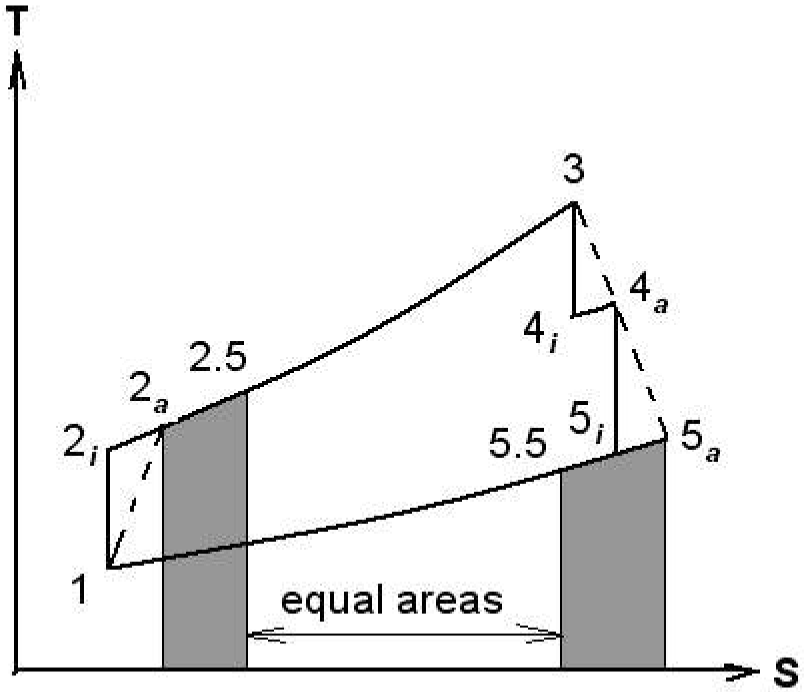

Figure 12.2. Temperature-entropy diagram for a regenerative air standard gas turbine engine

For the air standard cycle where the mass rate of flow through the turbine equals the mass rate of flow through the compressor, and for constant specific heats,

$$\eta_{reg} = \frac{T_{2.5} - T_{2a}}{T_{5a} - T_{2a}}. \qquad (12.2)$$

In an actual gas turbine with a regenerator, there will be a pressure drop in the regenerator. Effectiveness, pressure drop, and leakage are the three heat exchanger performance parameters that influence the efficiency of a gas turbine engine with exhaust heat recovery.

12.3 Steam-Injected Gas Turbine Cycle

A modification to the basic cycle, which has been used for power generation since the mid-1980s, is the steam-injected gas turbine. The schematic diagram for the steam-injected gas turbine is shown in Fig. 12.3. The exhaust gases from the power turbine are used as the energy source in a heat recovery steam generator (HRSG) where energy is transferred from the exhaust gases to boiler feedwater. The water at the exit from the HRSG is high-pressure steam, which is injected along with fuel in the combustion chamber. It is also possible to inject a portion of the high pressure steam into the gas turbine between the gas generator turbine and the power turbine.

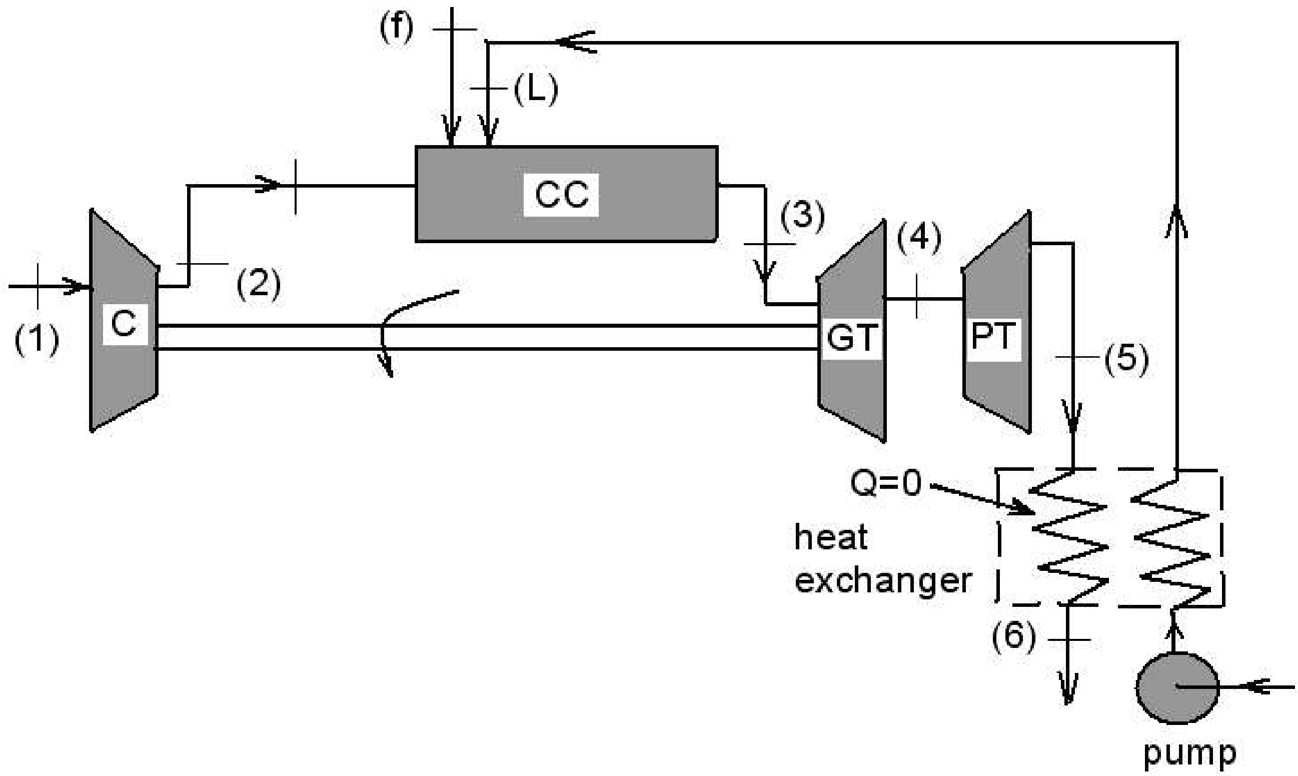

Figure 12.3. Schematic diagram for a steam-injected gas turbine cycle

The advantages of the steam-injected gas turbine are:

1. Able to place into service quickly.

2. Helps reduce NO_x emissions.

3. Increases the net power output compared with a comparable simple cycle gas turbine.

4. Has a lower heat rate (and a higher thermal efficiency) than a comparable simple cycle gas turbine.

5. Lower capital cost than for a combined cycle.

6. Can be converted to a combined cycle.

Disadvantages of the steam injected gas turbine are:

1. A constant supply of purified water is required since the steam after being injected into the gas turbine cycle is exhausted to the atmosphere.

2. A heat recovery steam generator must be added.

3. Care must be taken to avoid injected compressed liquid (water slugs) into the combustion chamber as this will cause a partial or complete extinction of the flame in the combustion chamber.

The amount of water injected into the discharge area between the compressor discharge and combustion chamber is usually between 2% and 3% of the compressor discharge air, although flow can be as high as 5% of the compressor discharge air flow.

The injected steam must be superheated under all conditions and at a pressure higher than the compressor discharge pressure.

12.4 Evaporative-Regenerative Gas Turbine Cycles

Another way to increase the specific power output of the simple gas turbine cycle is the evaporative-regenerative gas turbine cycle, which is shown schematically in Fig. 12.4. In this cycle, water is sprayed into the air as the water leaves the compressor. As the water evaporates, it lowers the temperature of the air, resulting in a large temperature difference between the temperature of the air leaving the power turbine (state 5) and the air leaving the evaporator (state 2.3), thereby justifying the addition of a regenerator/recuperator between the evaporator exit (state 2.3) and the combustion chamber inlet (state 2.5).

The mass of water added and the fact water has a higher specific heat than air results in a gas turbine with a larger specific power and a lower heat rate when compared to a gas turbine operating on a simple cycle.

Care must be taken to be certain all of the water sprayed into the air in the evaporator is evaporated. The maximum amount is limited in theory to that which results in the air being saturated at the evaporator exit (state 2.3) or when the temperature at the evaporator exit is equal to the temperature of the water being sprayed into the evaporator.

12.5 Gas Turbine with Intercooling

Two other improvements can be made in the gas turbine cycle. These are intercooling and reheat. Both of these improvements increase the net work obtained from the cycle, intercooling decreasing the compressor work without changing the turbine work, reheat increasing the turbine work without changing the compressor work. The following discussion of intercooling and reheat assumes the same compressor inlet temperature, turbine inlet temperature, and same overall pressure ratio as the basic gas turbine cycle. This section discusses intercooling.

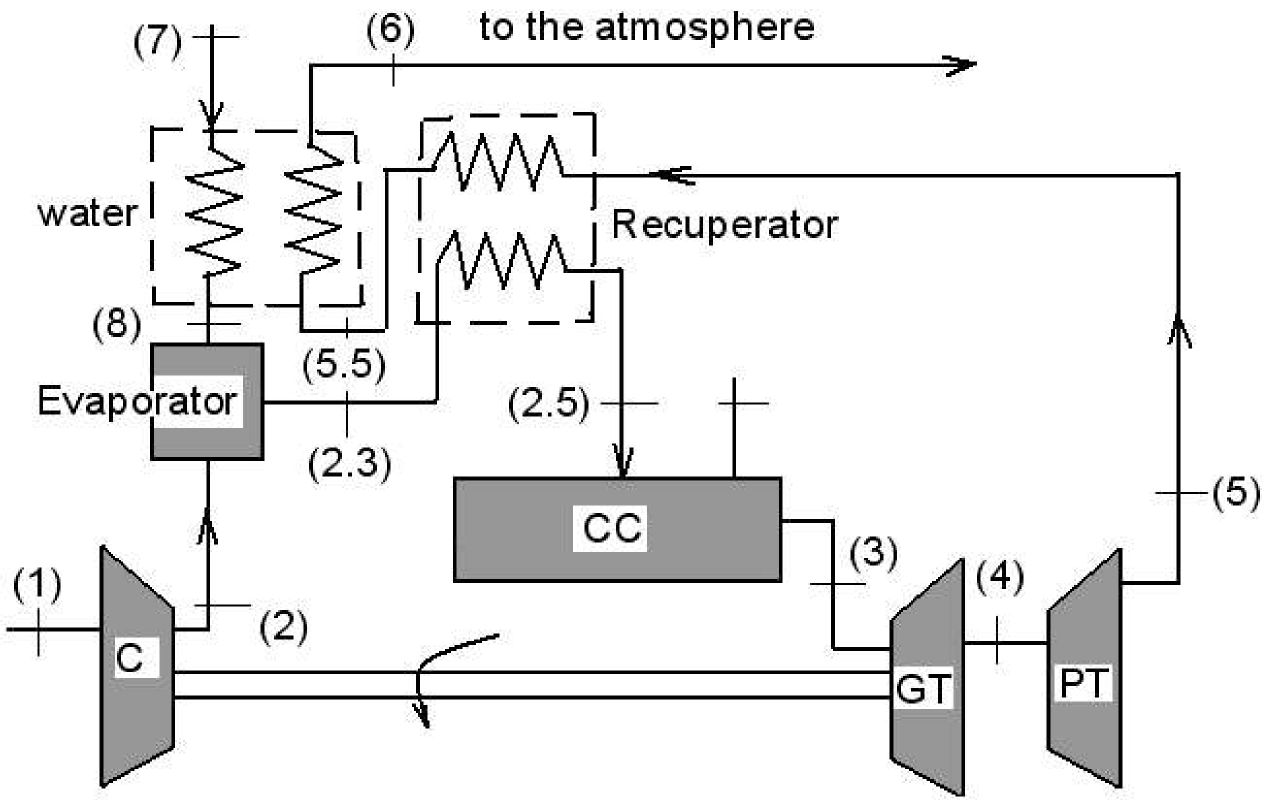

Figure 12.4. Evaporative-regenerative gas turbine cycle

Compressors, as used in gas turbines, operate adiabatically. Ideally, the compressor process is reversible, steady-flow.

Examination of the pressure-volume diagram (Fig. 12.5) shows that the work for an isothermal, steady-flow compression process from p_1 to p_2 is less than that for an isentropic compression process between the same two pressures, the decrease in work being represented by the shaded area.

It is impractical to construct a gas turbine compressor that would operate isothermally. The compression process can be made to approximate the isothermal compression process by intercooling, which involves the use of two or more compressors.

Fig. 12.6 illustrates the schematic diagram for a gas turbine engine with two compressors and intercooling between the compressors. In the engine illustrated in this figure, it is assumed that the gas generator turbine (GT) drives both compressors, the power turbine (PT) delivers power to an external device (load).

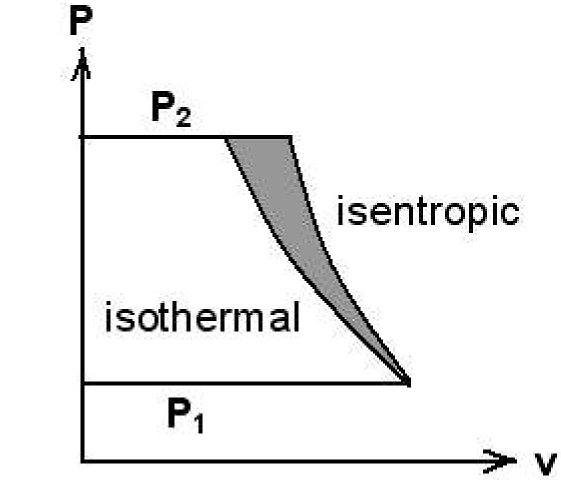

Figure 12.5. Pressure-specific volume diagram for an isothermal and isentropic compression process

The first compressor (C_1) compresses air from ambient pressure to some intermediate pressure. The second compressor (C_2) completes the compression process to the desired final pressure. Fig. 12.7 illustrates three ***p*** - ***v*** diagrams for different intermediate pressures. In all three parts of the diagrams, it is assumed that the temperature leaving the intercooler is the same as that entering the first compressor.

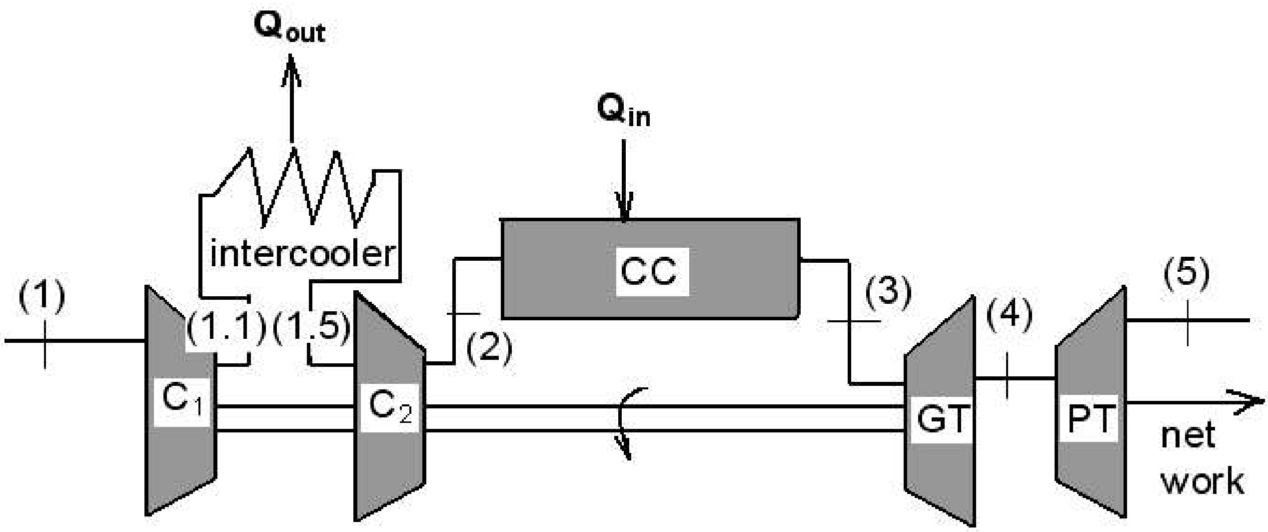

Figure 12.6. Gas turbine engine with intercooling

The three diagrams illustrate the work saved by using intercooling, this work saved being represented by the shaded area. Examination of Fig. 12.7 suggests that there should be an optimum intercooler pressure. It can be shown that this optimum intercooler pressure is (assuming no pressure drops in the intercooler and that the temperature leaving the intercooler is the same as the temperature entering the first compressor)

$$p_i=(p_1p_2)^{0.5}. \qquad (12.4)$$

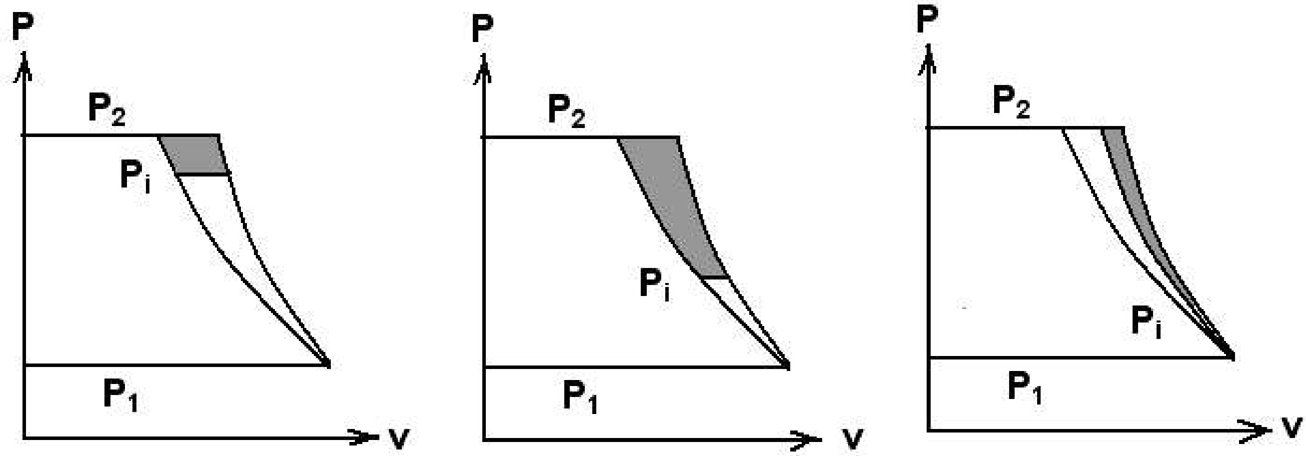

Figure 12.7. Pressure-specific volume diagram for compression process with intercooling. (a) High intercooler pressure. (b) Moderate intercooler pressure. (c) Low intercooler pressure

Because of the cost and complexity of adding an intercooler, no more than one intercooler is ever used in the design of a gas turbine engine.

12.6 Gas Turbine with Reheat

Fig. 12.8 is a schematic diagram for a gas turbine engine operating on the reheat cycle. The engine illustrated in this figure has turbine GT driving the compressor, the net work developed by the power turbine (PT). Fig. 12.9 illustrates the temperature-entropy diagram for a gas turbine with reheat.

The optimum intermediate pressure with reheat is the same as with intercooling or

$$p_{re}=(p_3p_5)^{0.5}. \qquad (12.5)$$

Adding reheat to a gas turbine engine increases the turbine work (therefore the net work) without changing the compressor work.

12.7 Gas Turbine with Intercooling, Reheat, and Regeneration

Intercooling and reheat, when are used, decrease the cycle thermal efficiency; therefore, they are seldom, if ever, used alone.

Fig. 12.10 illustrates temperature-entropy diagram for a gas turbine with intercooling-reheat-regeneration cycle that is usually used whenever reheat and/or intercooling are used. The engine illustrated in Fig. 12.10 uses turbine T_2 (HPT) to drive compressor C_2 (HPC) and turbine T_1 (LPT) to drive the compressor C_1 (LPC). The power turbine (PT) delivers work to external equipment.

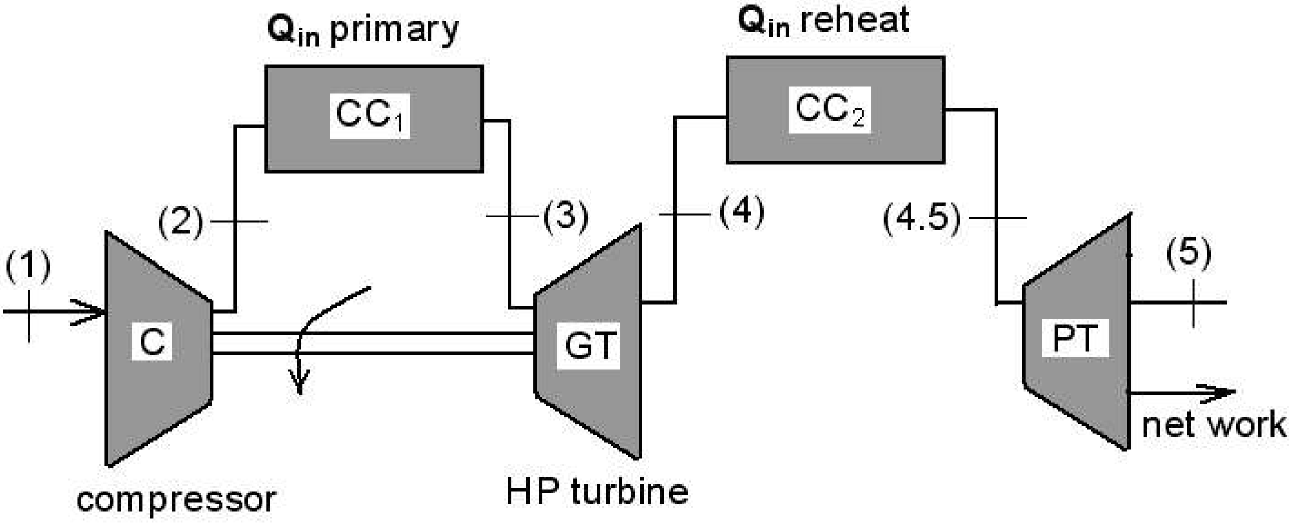

Figure 12.8. Schematic diagram of a gas turbine with reheat where the high-pressure turbine drives the compressor

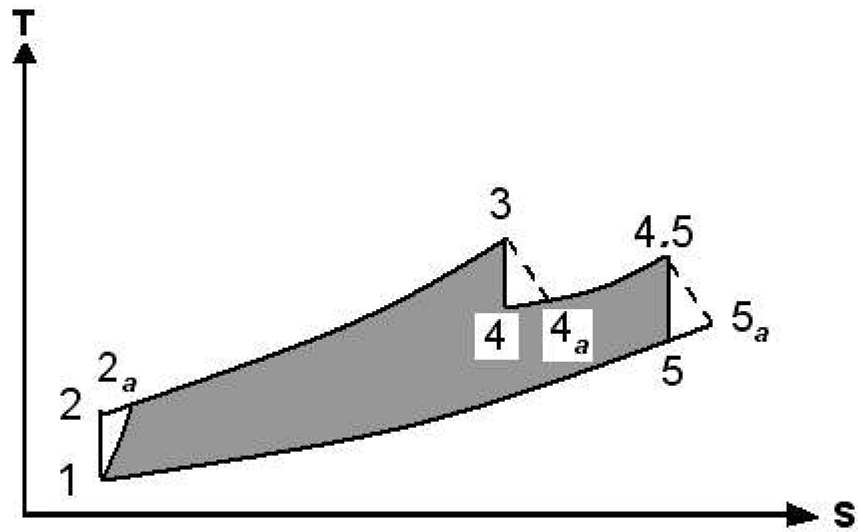

Figure 12.9. Temperature-entropy diagram for a gas turbine with reheat

12.8 Combined-Cycle Power Plant

Baseload units operate essentially all year long. These units are usually the most efficient and reliable units, with a major design consideration being units that give the lowest cost in mills per kilowatt.

Peaking units usually operate for no more than 5% to 6% of the time during the year. Units in this category must provide reliable standby reserve. Equipment used in this category is characterized with the lowest possible capital cost with little concern for operating efficiency.

The intermediate load is between the baseload units and the peaking units. It must be able to take the swing load, operates up to 75% of the time and must have a short startup time.

A review of the literature suggests many ways to combine two or more thermal cycles into a power plant. The only one that has been widely accepted is the combination of the gas turbine and a steam turbine.

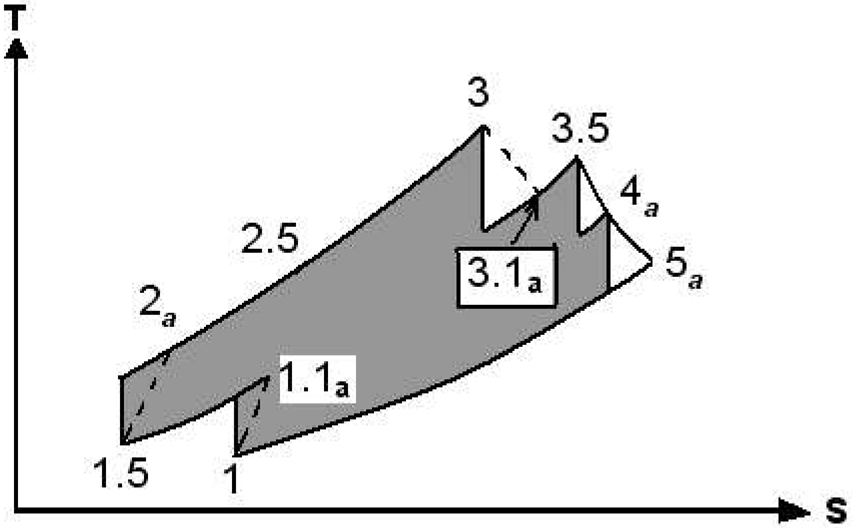

Figure 12.10. Temperature-entropy diagram of an intercooled- reheat regenerative gas turbine cycle

Reasons for considering a combined cycle are:

1. High cycle thermal efficiency or low heat rate.

2. Reduces need for cooling water since only the steam turbine portion of the cycle needs cooling water.

3. Able to build in stages. The gas turbine can be installed initially and operated as a simple cycle gas turbine. At a later date, the steam turbine can be added and then operated as a combined-cycle power plant.

4. Able to operate the gas turbine with the steam turbine idle if a diverter valve was installed so the gas turbine exhaust gas can by-pass the heat recovery steam generator.

5. Smaller unit size.

Studies have shown that a combined-cycle system using gas and steam turbines could provide an optimum plant for the intermediate and base load areas with thermal efficiencies well above 50%.

The combined-cycle plant usually is composed on one or more gas turbines exhausting into a HRSG. Some of the units have supplementary fuel added to the gas turbine exhaust gases; others do not use any additional fuel.

The amount of energy transferred from the exhaust gases to the water can be increased by decreasing the *pinch point* temperature (the minimum temperature difference between the gas stream and the water stream) differences. This requires an increase in the heat exchanger surface area.

In designing a combined-cycle power plant, one must be certain to make optimum use of the energy in the gases leaving the gas turbine in the HRSG.

12.9 Combined-Cycle Power Plant with Supplementary Firing

The preceding section examined the fundamentals of the single-pressure combined-cycle power plant with no additional energy being added to the gas turbine exhaust gases between the exit from the gas turbine power turbine and the inlet to the HRSG.

When additional heat is added between the gas turbine exit and the inlet to the heat recovery steam generator, the fraction of the total output produced by the gas turbine, decreases. Advantages of adding supplementary firing to the combined cycle include the ability to

1. Increase the total combined cycle output.

2. Control the temperature at the inlet to the HRSG. The gas temperature and air flow rate at the exit from the gas turbine and very dependent on the temperature at the inlet to the gas turbine compressor.

Comparing the results from example problems, one observes that

1. A higher fraction of the output results from the steam turbine portion of the combined cycle when supplementary firing is used.

2. The pinch point temperature difference is much higher for the combined-cycle with supplementary firing since the inlet and exit temperatures were assumed to be the same for both cycles.

3. The unfired combined-cycle power plant had a lower heat rate.

4. Decreasing the pinch point temperature difference for the cycle with supplementary firing to that of the unfired cycle would decrease the HRSG exit temperature for the exhaust gases and increase the fraction of power developed by the steam turbine.

One should recognize that there is a significant difference between a conventional steam power plant and the combined-cycle power plant. A conventional steam power plant achieves a higher efficiency if the feedwater is increased to a higher temperature by means of one or more open and/or closed feedwater heaters. The combined-cycle power plant achieves a higher efficiency with a low feedwater temperature.

For a combined cycle, a high steam pressure does not necessarily mean a high thermal efficiency (or low overall heat rate). A high steam pressure does increase the steam turbine cycle efficiency but decreases the rate of energy transferred from the exhaust gases.

12.10 Multipressure Combined-Cycle Power Plants

The preceding two sections assumed a single-pressure steam turbine cycle for the combined cycle. Designing combined-cycle steam turbines with steam supplied to the steam turbines at two different pressures increases the cycle thermal efficiency over that of a one-pressure steam turbine cycle, allows for better utilization of the energy in the exhaust gases and, when reheat is used, allows for good control of the quality of the steam at the exit from the turbines.

The major design parameters that influence the performance of the steam portion of the combined-cycle power plant are the

1. Steam pressure and temperature at the inlet to the high pressure turbine.

2. Steam pressure and temperature at the inlet to the low pressure turbine.

3. High pressure and low pressure pinch points.

4. Amount of superheat (if any) of the steam supplied to the low pressure turbine.

5. Fraction of the total steam flow that passes through both the high pressure and low pressure turbine.

6. Condenser pressure.

12.11 Closed-Cycle Gas Turbines

The merits of the closed-cycle gas turbines have long recognized for combined electrical power and the utilization of waste heat for industrial uses and urban district heating.

The advantages of the closed-cycle gas turbine gas turbine include:

1. Possibility of using a wide variety of medium gases such as air, carbon dioxide, helium, hydrogen, or nitrogen.

2. Possibility of using a wide variety of fuels such as coal, oil, gas, nuclear, solar, or stored energy.

3. Ability to operate with a much higher pressure. Increasing the pressure decreases the specific volume, thereby permitting more work per unit area. A decrease in specific volume also improves the heat transfer characteristics of the gas turbine medium.

4. Opportunity to vary the total pressure of the system. This offers an opportunity to control part-load output that is independent of the turbine inlet temperature. The total pressure may be varied by adding or withdrawing fluid from the gas turbine. This is accomplished by the use of pressurized storage tanks.

5. Working fluid (and components) is not contaminated by the combustion process.

6. Heat rejection characteristics are well suited to either dry cooling, wet-dry cooling, district heating, or utilization of a binary cycle. The cooling of the closed-cycle gas turbine exhaust gas may be achieved by a dry air-cooling tower because of the large difference between the exhaust gas temperature and the ambient air temperature.

12.12 Stationary Gas Turbines Emission Reduction

The oxides of nitrogen, NO_x, are the predominant emission from stationary gas turbine engines and the one that is controlled by standards. The most prevalent NO_x emissions are nitric oxide, NO, and nitrogen dioxide, NO_2.

It was illustrated in Chapter 11, that the higher the temperature and the longer the gases are at that temperature, the more nitric oxide is formed. NO_x is the main pollutant from stationary gas turbine engines. The amount of SO_2 emitted is limited by the amount of sulfur in the fuel since this is the only source of sulfur.

Prior to NO_x emissions controls, gas turbine engine combustion chambers were designed so that the fuel-air ratio in the primary zone was approximately the stoichiometric value; that is, the percent excess air in the primary zone was 0%. This resulted in maximum temperature.

The maximum temperature can be reduced by designing the combustion chamber so that the primary zone either operates fuel rich (insufficient air for complete combustion) or fuel lean (excess air). Both of these conditions can result in increased smoke (fuel rich) or increased carbon monoxide and total hydrocarbon emissions (fuel lean).

Several methods can be used to reduce NO_x emissions. These include

1. Water or steam injection.

2. Staged combustion.

3. Selective catalytic reduction.

The most commonly used method of controlling NO_x emissions is with water or steam injection into the primary zone of the combustion chamber. The water (or steam) injected acts as a heat sink, resulting in a lower maximum temperature, thereby reducing the amount of NO_x formed. The rate at which water is injected is approximately 50% of the fuel flow. Steam rates are usually 100-200% of the fuel flow.

NOMENCLATURE

Symbols

ρ - density of fluid, kg/m^3

η - efficiency

$\overline{M}$ - molecular weight of the gas, kg/mol

$\overline{R}$ - universal gas constant, J/kg.K

C - absolute velocity, m/s

c - constant specific heat, J/kg.K

F - cross-sectional area, m^2

G - mass rate of flow, kg/s

i - enthalpy, J/kg

k - specific heat ratio

Δ***KE -*** change in the kinetic energy, J/kg

Δ***PE -*** change in the potential energy, J/kg

L_{12} - work done by the system in going from state 1 to state 2, J/kg

m - mass, kg

p - pressure, Pa

Q, q - thermal energy, total heat transfer; , J/kg

q_f - relative fuel mass rate

R - gas constant, J/kg.K; thrust, N

s - entropy, J/kg.K

T - temperature, K

U - internal energy, J/kg

V - total volume, m^3; aircraft speed, jet velocity at propelling nozzle, km/h

v - specific volume of the fluid, m^3/kg

W - power output, kW

0; 1; 2; 3; 4; 5 etc – states

Subscripts

v - *volume*

a – axial; air; ambient

B – burner

e – exit

f – fuel

i – isentropic; inlet; intercooling

in – inlet

j - *jet*

o – overall

out – outlet

p - pressure; propulsion

re – reheat

reg – regenerator

s – side
sp – specific
th – thermal

Superscripts
0 – total (stagnation) parameter

Abbreviations
BPR - by-pass ratio
C – compressor; Celsius
CC – combustion chamber
CO - carbon monoxide
CO_2 – carbonic acid
EPN – effective perceived noise
ESP - equivalent shaft power
GT – gas turbine
GTE – gas turbine engine
HC - hydrocarbon
HP – high pressure
HPC – high pressure compressor
HPT – high pressure turbine
HRSG - heat recovery steam generator
LH_2 – liquid hydrogen
LNG – liquid natural gas
LP – low pressure
LPC – low pressure compressor
LPT – low pressure turbine
M – Mach number
NASA – National Aeronautics and Space Administration
NO - nitric oxide
NO_2 - nitrogen dioxide
NO_x – nitrogen oxide
PT – power turbine
rpm – revolutions per minute
SFC - specific fuel consumption, kg/kW.h
SO_2 – sulfur dioxide
SP – shaft power
T - turbine

Units
dB – decibel
h – hour
J - Joul
K – Kelvin

k – kilo
kg – kilogram
kJ – kiloJoul
kW – kilowatt
m – meter
N – Newton
Pa – Pascal
s – second
W – watt

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Навчальне видання

Незим Віталій Юрійович

ГАЗОТУРБІННІ ДВИГУНИ: ПРОБЛЕМИ ПРОЕКТУВАННЯ

(Англійською мовою)

Редактор Є.В. Пизіна

Технічний редактор Л.О. Кузьменко

Зв. план, 2012
Підписано до друку 17.04.2012
Формат 60x84 1/16. Папір офс. № 2. Офс. друк
Ум. друк. арк. 4,6. Обл.-вид. арк. 5,25. Наклад 50 пр. Замовлення 107.
Ціна вільна

Національний аерокосмічний університет ім. М.Є. Жуковського
«Харківський авіаційний інститут»
61070, Харків-70, вул. Чкалова, 17
http://www.khai.edu
Видавничий центр «ХАІ»
61070, Харків-70, вул. Чкалова, 17
izdat@khai.edu