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SCHOOL OF MECHANICAL ENGINEERING

DEPARTMENT OF AERONAUTICAL ENGINEERING

UNIT – I- FUNDAMENTALS OF GAS TURBINE ENGINES – SAE1302

I. Introduction

Propulsion

Propulsion is the action or process of pushing or pulling to drive an object forward. The term is derived from two Latin words: Pro, meaning forward; and pellere, meaning to drive. Propulsion is divided into 2 types: Aircraft Propulsion, Aerospace Propulsion.An aircraft propulsion system generally consists of an engine to generate thrust. Based on the Altitude, Range, Endurance and Speed of the aircraft, the Engine will be Turboprop, Turbo Shaft, Turbo Jet, Turbofan and Ram Jet Engine. Air breathing Engine is the engine which utilizes atmospheric Air For Combustion

History of Propulsion

Aeolipile

In 1st century AD when Hero of Alexandria built an engine called an aeolipile.He mounted a hollow metal globe with projecting tubes between two pipes so it could spin.Steam entered the globe through the pipes. As it escaped through the bent tubes, the jets of steam spun the globe.

Fig 1 : Aeolipile

Hero's machine illustrates a scientific principle that Sir Isaac Newton formulated in 1687. Newton's third law of motion states that for every action there is an equal and opposite reaction. In Hero's machine the jets of steam escaping from the tubes are the action, the spinning of the globe the reaction. The same principle applies to jet engines, and for this reason they are called reaction engines.

Branca's Stamping Mill

In 1629 an Italian engineer, Giovanni Branca, was probably the first to invent an actual impulse turbine. This device, a stamping mill was generated by a steam-powered turbine.

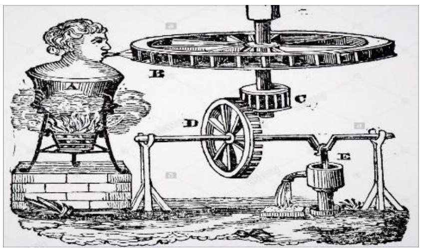

Fig 2: Branca's Stamping Mill

A jet nozzle directed steam onto a horizontally mounted turbine wheel, which then turned an arrangement of gears that operated his mill.

The Steam Wagon

In 1687 Isaac Newton attempted to put his newly formulated laws of motion to the test with his "steam wagon“.He tried to propel the wagon by directing steam through a nozzle pointed rearward. Steam was produced by a boiler mounted on the wagon. Due to lack of power from the steam, this vehicle didn't operate.

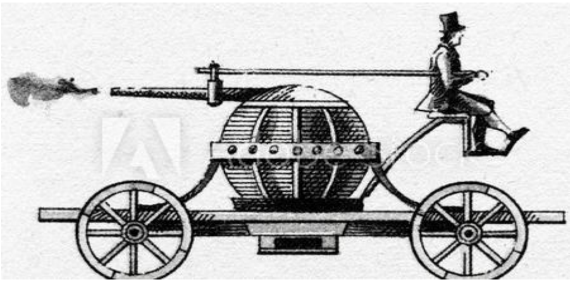

Fig 3: The Steam Wagon

Barber’s Instrument

In 1791 John Barber, an Englishman. was the first to patent a design that used the thermodynamic cycle of the modem gas turbine. His design contained the basics of the modem gas turbine it had a compressor, a combustion chamber, and a turbine. The main difference in his design was that the turbine was equipped with a chain-driven reciprocating type of compressor.

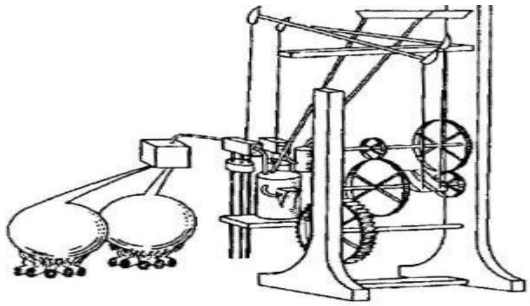

Fig 4: Barber’s Instrument

Modern Aviation

In January 1930 an Englishman, Frank Whittle, submitted a patent application for a gas turbine for jet propulsion. It wasn't until the summer of 1939 that the Air Ministry awarded Power Jets Ltd a contract to design a flight engine.

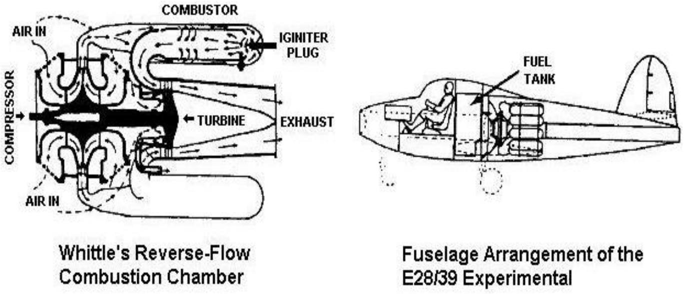

Fig 5: White’s Engine

In May 1941 the Whittle W1 engine made its first flight mounted in the Gloster Model E28/39 aircraft. This airplane would later achieve a speed of 370 MPH in level flight with 1000 pounds of thrust.The Germans, Hans von Ohain and Max Hahn, students at Gottingen, seemingly unaware of Whittle's work patented a jet propulsion engine in 1936.

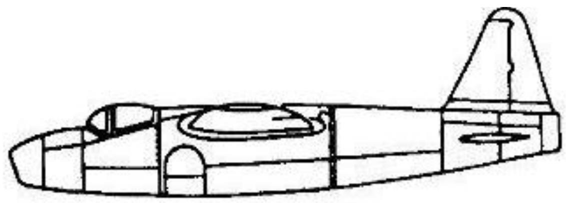

Fig 6: Heinkel Aircraft

Ernst Heinkel Aircraft Company adapted their ideas and flew the second aircraft engine of this development in an HE-178 aircraft on August 27, 1939. This was the first true jet-propelled aircraft. The engine, known as the Heinkel HES-36, developed 1100 pounds of thrust and hurled the HE-178 to speeds of over 400 MPH. This engine used a centrifugal flow compressor.

TYPES OF ENGINE

Thrust produced by increasing the kinetic energy of the air in the opposite direction of flight

Slight Acceleration of a Large Mass of Air and Engine Driving a Propeller-Turbo Prop, Turbo Shaft

Large Acceleration of a Small Mass of Air- Turbojet

Combination of Both-Turbofan

II. Turbojet Engine

The turbojet is an airbreathing jet engine, typically used in aircraft. It consists of a gas turbine with a propelling nozzle. The gas turbine has an air inlet, a compressor, a combustion chamber, and a turbine (that drives the compressor). The compressed air from the compressor is heated by burning fuel in the combustion chamber and then allowed to expand through the turbine. The turbine exhaust is then expanded in the propelling nozzle where it is accelerated to high speed to provide thrust.Two engineers, Frank Whittle in the United Kingdom and Hans von Ohain in Germany, developed the concept independently into practical engines during the late 1930s.

While the turbojet was the first form of gas turbine powerplant for aviation, it has largely been replaced in use by other developments of the original concept. In operation, turbojets typically generate thrust by accelerating a relatively small amount of air to very high supersonic speeds, whereas turbofans accelerate a larger amount of air to lower transonic speeds. Turbojets have been replaced in slower aircraft by turboprops because they have better specific fuel consumption. At medium speeds, where the propeller is no longer efficient, turboprops have been replaced by turbofans. The turbofan is quieter and has better range-specific fuel consumption than the turbojet. Turbojets can be highly efficient for supersonic aircraft.

Turbojets have poor efficiency at low vehicle speeds, which limits their usefulness in vehicles other than aircraft. Turbojet engines have been used in isolated cases to power vehicles other than aircraft, typically for attempts on land speed records. Where vehicles are "turbine-powered", this is more commonly by use of a turboshaft engine, a development of the gas turbine engine where an additional turbine is used to drive a rotating output shaft. These are common in helicopters and hovercraft. Turbojets were used on Concorde and the longer-range versions of the TU-144 which were required to spend a long period travelling supersonically. Turbojets are still common in medium range cruise missiles, due to their high exhaust speed, small frontal area, and relative simplicity. They are also still used on some supersonic fighters such as the MiG-25, but most spend little time travelling supersonically, and so employ turbofans and use afterburners to raise exhaust speed for supersonic sprints.

Turbojets are the oldest kind of general-purpose jet engines.Turbojets are rotary engines that extracts energy from a flow of combustion gasThey produce thrust by increasing the velocity of the air flowing through the engine and operate on Newton’s third law of motion " For every action there is an equal and opposite reaction”. Newton’s 2nd Law on motion F= Mass * Acceleration. Here Large acceleration with small mass of air

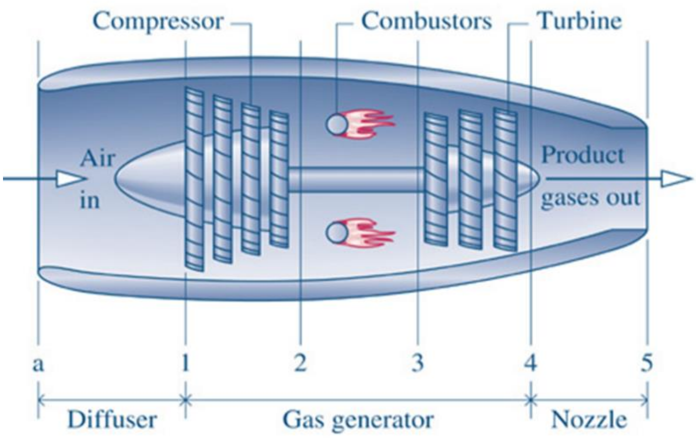

Fig 7: Turbojet Engine

a-1 Isentropic increase in pressure (diffuser)
1-2 Isentropic compression (compressor)
2-3 Isobaric heat addition (combustion chamber)
3-4 Isentropic expansion (turbine)
4-5 Isentropic decrease in pressure with an increase in fluid velocity (nozzle)

The operation of a turbojet is modelled approximately by the Brayton cycle. The efficiency of a gas turbine is increased by raising the overall pressure ratio, requiring higher-temperature compressor materials, and raising the turbine entry temperature, requiring better turbine materials and/or improved vane/blade cooling. It is also increased by reducing the losses as the flow progresses from the intake to the propelling nozzle. These losses are quantified by compressor and turbine efficiencies and ducting pressure losses. When used in a turbojet application, where the output from the gas turbine is used in a propelling nozzle, raising the turbine temperature increases the jet velocity. At normal subsonic speeds this reduces the propulsive efficiency, giving an overall loss, as reflected by the higher fuel consumption, or SFC. However, for supersonic aircraft this can be beneficial, and is part of the reason why the Concorde employed turbojets. Turbojet systems are complex systems therefore to secure optimal function of such system, there is a call for the newer models being developed to advance its control systems to implement the newest knowledge from the areas of automation, so increase its safety and effectiveness

Primary Parts of Turbojet Engine

1. Air Intake/ Inlet
2. Compressor
3. Combustion Chamber
4. Turbine
5. Nozzle

Secondary Parts of Turbojet Engine

1. After Burner
2. Auxiliary Power Unit

Construction

Intake

Intake refers to the capture area definition and attached ducting to an aircraft gas turbine engine. Air intake aims at bringing large amounts of surrounding air into the engine. A tube-shaped inlet, like one you would see on an airliner usually of cylindrical or conical design. Inlets come in many shapes and sizes depending on the aircraft. The intake has to supply air to the engine with an acceptably small variation in pressure (known as distortion) and having lost as little energy as possible on the way (known as pressure recovery). The ram pressure rise in the intake is the inlet's contribution to the propulsion system's overall pressure ratio and thermal efficiency.

Compressor

The compressor rotates at very high speed, adding energy to the airflow and at the same time squeezing it into a smaller space. Compressing the air increases its pressure and temperature. The compressor is driven by the turbine. Compressors used in turbojet engines are mainly classified as: Axial Flow Compressors and Centrifugal Compressors. Turbojets supply bleed air from the compressor to the aircraft for the environmental control system, anti-icing, and fuel tank pressurization, for example. The engine itself needs air at various pressures and flow rates to keep it running. This air comes from the compressor, and without it, the turbines would overheat, the lubricating oil would leak from the bearing cavities, the rotor thrust bearings would skid or be overloaded, and ice would form on the nose cone. The air from the compressor, called secondary air, is used for turbine cooling, bearing cavity sealing, anti-icing, and ensuring that the rotor axial load on its thrust bearing will not wear it out prematurely.

Combustion Chamber

The burning process in the combustor is significantly different from that in a piston engine. In a piston engine, the burning gases are confined to a small volume, and as the fuel burns, the pressure increases. In a turbojet, the air and fuel mixture burn in the combustor and pass through to the turbine in a continuous flowing process with no pressure build-up. Instead, a small pressure loss occurs in the combustor.The fuel-air mixture can only burn in slow-moving air, so an area of reverse flow is maintained by the fuel nozzles for the approximately stoichiometric burning in the primary zone.

Further compressed air is introduced which completes the combustion process and reduces the temperature of the combustion products to a level which the turbine can accept. Less than 25% of the air is typically used for combustion, as an overall lean mixture is required to keep within the turbine temperature limits.

Turbine

Hot gases leaving the combustor are allowed to expand through the turbine. Turbines are usually made up of high temperature alloys such as inconeL The turbine's rotational energy is used primarily to drive the compressor. and other accessories, like fuel, oil, and hydraulic pumps. In a turbojet almost two-thirds of all the power generated by burning fuel is used by the compressor to compress the air for the engine. In the first stage, the turbine is largely an impulse turbine (similar to a pelton wheel) and rotates because of the impact of the hot gas stream. Later stages are convergent ducts that accelerate the gas. Energy is transferred into the shaft through momentum exchange in the opposite way to energy transfer in the compressor. The power developed by the turbine drives the compressor and accessories, like fuel, oil, and hydraulic pumps that are driven by the accessory gearbox.

Nozzle

After the turbine, the gases are allowed to expand through the exhaust nozzle to atmospheric pressure, producing a high velocity jet in the exhaust plume. In a convergent nozzle, the ducting narrows progressively to a throat. After the turbine, the gases expand through the exhaust nozzle producing a high velocity jet. In a convergent nozzle, the ducting narrows progressively to a throat. The nozzle pressure ratio on a turbojet is high enough at higher thrust settings to cause the nozzle to choke.If, however, a convergent-divergent de Laval nozzle is fitted, the divergent (increasing flow area) section allows the gases to reach supersonic velocity within the divergent section. Additional thrust is generated by the higher resulting exhaust velocity.

After Burner

An afterburner or "reheat jet-pipe" is a device added to the rear of the jet engine. It provides a means of spraying fuel directly into the hot exhaust, where it ignites and boosts available thrust significantly; a drawback is its very high fuel consumption rate. An afterburner or "reheat jetpipe" is a combustion chamber added to reheat the turbine exhaust gases. The fuel consumption is very high, typically four times that of the main engine.

Merits

1. Very high power-to-weight ratio.
2. Compact than most reciprocating engines of the same power rating.
3. High operation speeds.
4. Low lubricating oil cost and consumption.

Demerits

1. Cost and Longer startup than reciprocating engines
2. Less responsive to changes in power demand

Application

Supersonic aircraft and Military aircraft.

III. Turbo-Prop Engine

Turboprop engines generate a substantial shaft power in addition to nozzle thrust.Turboshaft engines, generate only shaft power. These engines are used in helicopters. The shaft power is used to drive the main rotor blade. Both turboprops and turboshafts have applications at relatively lower speeds.Turboprops and turboshafts usually have a free-turbine or power turbine to drive the propeller or the main rotor blade (turboshafts). Stress limitations require that the large diameter propeller rotate at a much lower rate and hence a speed reducer is required.Turboprops may also have a thrust component due to the jet exhaust in addition to the propeller thrust.

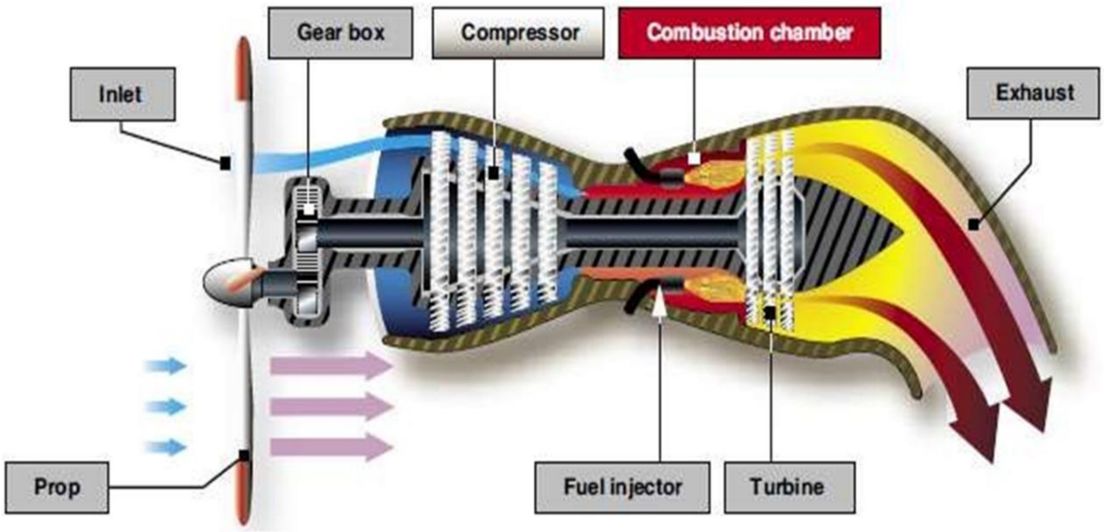

Fig 8: Turboprop Engine

Advantages

1. In dense air, i.e. lower levels, a propeller has a higher efficiency than jet exhaust;
2. generally turboprop aircraft can operate into shorter runways than turbo fan; mechanical Reliability due to relatively few moving parts
3. Light weight and simplicity of operation
4. High power per unit of weight

Disadvantages

1. Propellers lose efficiency at high altitudes;
2. Vibration levels can cause slight passenger discomfort;
3. En-route weather (icing/turbulence) can cause problems and additional passenger discomfort due to operating altitudes

Application

1. Passenger Aircraft of least range
2. Military Cargo Aircraft
3. Trainer Aircraft

IV. Turbo-Fan Engine

Propulsion efficiency is a function of the exhaust velocity to flight speed ratio. This can be increased by reducing the effective exhaust velocity. In a turbofan engine, a fan of a larger diameter than the compressor is used to generate a mass flow higher than the core mass flow. This ratio is called the bypass ratio.Turbofan engines have a higher propulsion efficiency as compared with turbojet engines operating in the same speed range. A turbofan engine is the most modern variation of the basic gas turbine engine. As with other gas turbines, there is a core engine, whose parts and operation are discussed on a separate page. In the turbofan engine, the core engine is surrounded by a fan in the front and an additional turbine at the rear.

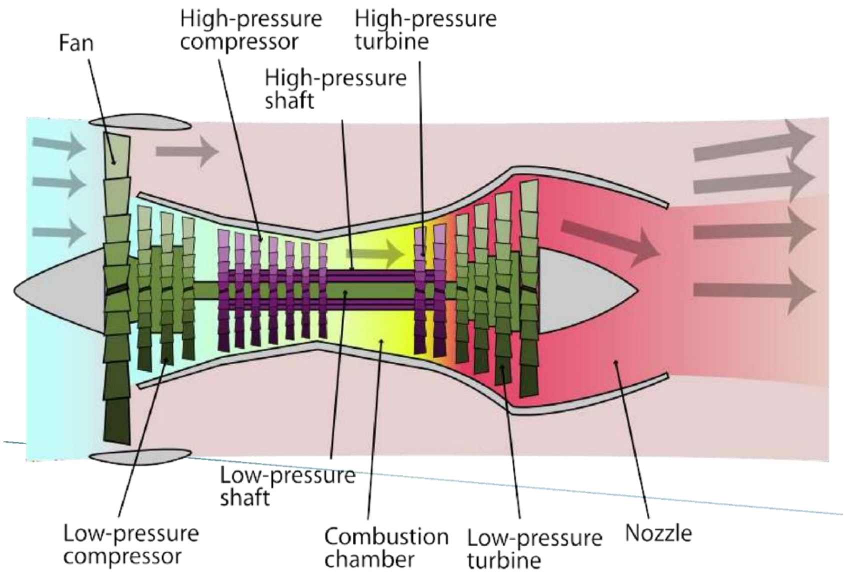

Fig 9: Turbo Fan Engine

Fan

The fan is responsible for producing the majority of the thrust generated by a turbofan engine and is easily visible when looking at the front of the engine.The fan is directly connected to the low pressure compressor (LPC) and the low pressure turbine (LPT) by way of a shaft known as the low pressure shaft. Turbofans were invented to circumvent the undesirable characteristic of turbojets being inefficient for subsonic flight. To raise the efficiency of a turbojet, the obvious approach would be to increase the burner temperature, to give better Carnot efficiency and fit larger compressors and nozzles. However, while that does increase thrust somewhat, the exhaust jet leaves the engine with even higher velocity, which at subsonic flight speeds, takes most of the extra energy with it, wasting fuel.Instead, a turbofan can be thought of as a turbojet being used to drive a ducted fan, with both of those contributing to the thrust. Whereas all the air taken in by a turbojet passes through the turbine (through the combustion chamber), in a turbofan some of that air bypasses the turbine.Because the turbine has to additionally drive the fan, the turbine is larger and has larger pressure and temperature drops, and so the nozzles are smaller. This means that the exhaust velocity of the core is reduced. The fan also has lower exhaust velocity, giving much more thrust per unit energy (lower specific thrust). The overall effective exhaust velocity of the two exhaust jets can be made closer to a normal subsonic aircraft's flight speed. In effect, a turbofan emits a large amount of air more slowly, whereas a turbojet emits a smaller amount of air quickly, which is a far less efficient way to generate the same thrustThe ratio of the mass-flow of air bypassing the engine core compared to the mass-flow of air passing through the core is referred to as the bypass ratio. The engine produces thrust through a combination of these two portions working together; engines that use more jet thrust relative to fan thrust are known as low-bypass turbofans, conversely those that have considerably more fan thrust than jet thrust are known as high-bypass. Most commercial aviation jet engines in use today are of the high-bypass type and most modern military fighter engines are low-bypass. Afterburners are not used on high-bypass turbofan engines but may be used on either low-bypass turbofan or turbojet engines.

Compressor

The purpose of compression is to prepare the air for combustion by adding energy in the form of pressure and heat.The compressor is divided into two portions: the low pressure compressor, mentioned above, and the high pressure compressor however, they interact with different parts of the turbofan engine.

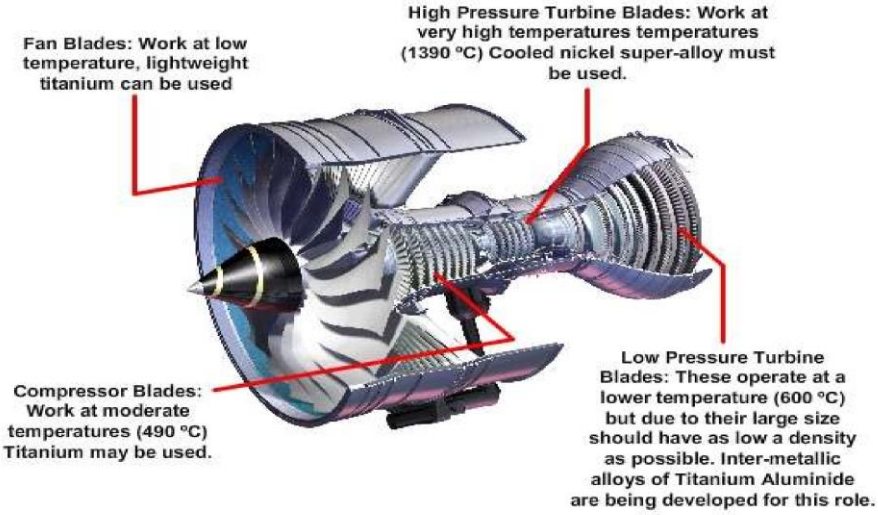

Fig 10: High Pressure and Low Pressure Blades

Combustion Chamber

Combustion occurs within the combustor, a stationary chamber within the core of the engine The combustor is directly downstream of the HPC and directly upstream of the high pressure . The purpose of the combustor is to add even more energy to the air flow by way of heat addition. Within the combustor, fuel is injected and mixed with the air. This fuel-air mixture is then ignited, creating a dramatic increase in temperature and energizing the flow, propelling it rearward towards the high pressure turbine.

Turbine

Expansion occurs within the high pressure and low pressure turbines. Similar in appearance to the compressors, the turbines have rows of blades which spin . The purpose of the turbines is to extract energy from the flow which is then used to spin the compressors and the fan. The spinning fan draws more air through the core of the engine which continues the entire process, and it pulls more bypass air around the engine, generating continuous thrust.

Nozzle

The exhaust nozzle is located directly downstream of the LPT and it is the last component that the air flow touches before exiting the engine. The purpose of the exhaust nozzle is to propel the core flow out of the engine, providing additional thrust. This is accomplished by way of its geometry or shape. The nozzle also helps regulate pressures within the engine to keep the other components functioning properly and efficiently.

V. RAMJET

Ramjet is simply a duct of a special shape, which faces the airflow caused by the forward motion and relies on the ram effect to collect the air, add heat to it by combusting suitable fuel and then exit through a nozzle at higher velocity and mass to create ever increasing thrust. There are no moving parts, no need for lubrication, and no energy losses in trying to run something. To put it simply, it is an 'Aerodynamic engine' or to make it sound more complicated we can call it Athodyd short for 'Aero- thermo-dynamic duct'. in which the injectors spray the mist of fuel into the ram compressed air stream and a spark ignites the mixture. The grill-type flame holder provides a type of barrier to the burning mixture while allowing, expanding hot gases to escape through the exhaust nozzle. The high-pressure air coming into the combustion chamber keeps the burning mixture from effectively reacting toward the intake end of the engine. It is important to note that ramjets will not function until enough air is coming through the intake to create a high-pressure flow. Otherwise, the expanding gases of the burning fuel-air mixture would be expelled from both ends of the engine.

Ram Effect

By a suitable design of intake the additional compression and therefore pressure rise can be achieved at the air intake which is called as ram effect. The ram effect increases with the increase in forward speed. At 1.0M the external compression caused by the ram effect in the engine intake is approximately equal to that of the engine.At higher Mach No the contribution of ram effect increases markedly as compared to the turbojet engine. Thus now we can dispense off the compressor and achieve the necessary compression by pure ram effect. The above three components viz. Ram effect, compressing the air, Engine, converting the chemical energy into heat and pressure energy and finally the jet converting heat and pressure energy into forward push make what is called as ram jets. Now compare the thrust obtained by the reaction engines, against the speed of the aircraft (TAS), it can be clearly seen that the trust without intake ram effect would be a straight line and will show a steady drop as the TAS increases and tends to equal the exhaust jet velocity. The ram effect however starts to increase as speed goes past 300 kts (500 kmph) and continues to increase the thrust till about 3.0M for the Turbojets. The subsequent drop in Turbojet thrust in this graph is again due to TAS approaching jet exhaust velocities (V) and the difference Ve-Vo reduces drastically (V being the TAS). This can be increased slightly by using Reheat augmentation engines,however 0 the turbojets have to now make way for ramjets to take over from here onwards.

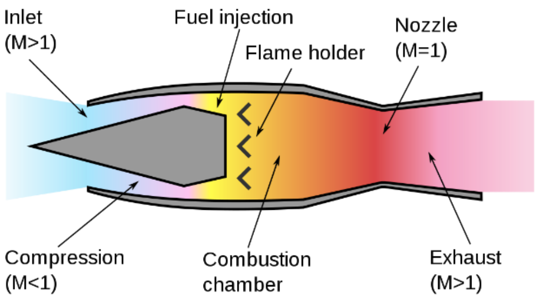

Fig 11: Ramjet Engine

Thrust Equation of the Turbojet Engine

Thrust = Momentum Thrust + Pressure Thrust

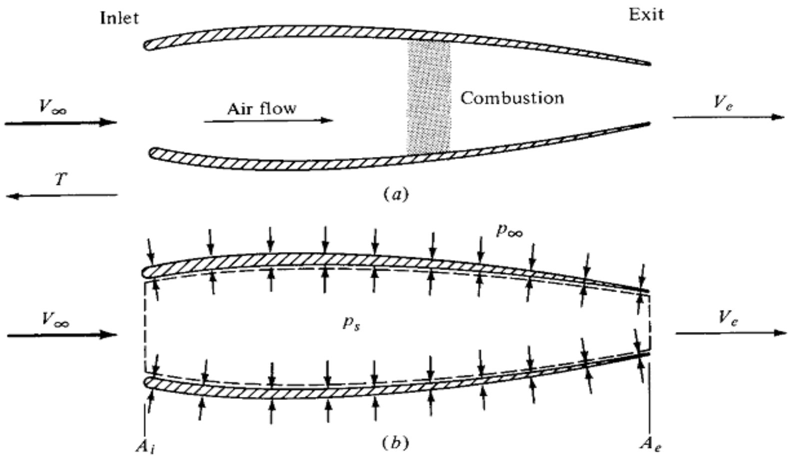

Fig 11: Engine thrust requirements

Pressure Thrust

P α = Ambient Pressure

Pe = Exit Pressure

V α = Ambient Velocity

Ve = Exit Velocity

T = Thrust

Ai, Ae = Inlet Area, Exit Area

Pressure = Force/ Area

Pα =Fα/A α

Fα = Pα * A α

Fα = Pα * dS α

Fe = Pe * dS e

X- Direction

Fα = (Pα * dS α) x

Fe = (Pe * dS e) x

$$T = \int (p_s \, dS)_x + \int (p_\infty \, dS)_x$$

$$\int (p_\infty \, dS)_x = p_\infty \int (dS)_x = p_\infty (A_i - A_e)$$

Pressure Thrust

$$T = \int (p_s dS)_x + p_\infty (A_i - A_e)$$

<u>Momentum Thrust</u>

Momentum = mass * velocity

M i = Mass of air * velocity

M e = Mass of gas * Velocity

$$F = (\dot{m}_{\text{air}} + \dot{m}_{\text{fuel}})V_e - \dot{m}_{\text{air}}V_\infty$$

$$F = (\dot{m}_{\text{air}} + \dot{m}_{\text{fuel}})V_e - \dot{m}_{\text{air}}V_\infty$$

$$(\dot{m}_{\text{air}} + \dot{m}_{\text{fuel}})V_e - \dot{m}_{\text{air}}V_\infty = p_\infty A_i + \int (p_s dS)_x - p_e A_e$$

$$\int (p_s dS)_x = (\dot{m}_{\text{air}} + \dot{m}_{\text{fuel}})V_e - \dot{m}_{\text{air}}V_\infty + p_e A_e - p_\infty A_i$$

$$\boxed{T = (\dot{m}_{\text{air}} + \dot{m}_{\text{fuel}})V_e - \dot{m}_{\text{air}}V_\infty + (p_e - p_\infty)A_e}$$

If the speed of the jet is equal to sonic velocity the nozzle is said to be "choked". If the nozzle is choked, the pressure at the nozzle exit plane is greater than atmospheric pressure, and extra terms must be added to the above equation to account for the pressure thrust. Cycle analysis is study of thermodynamic behavior of air as it flows through engine without regard for mechanical means used to affect its motion. The rate of flow of fuel entering the engine is very small compared with the rate of flow of air. In the Ideal Cycle and real cycle is shown in the figure 12.

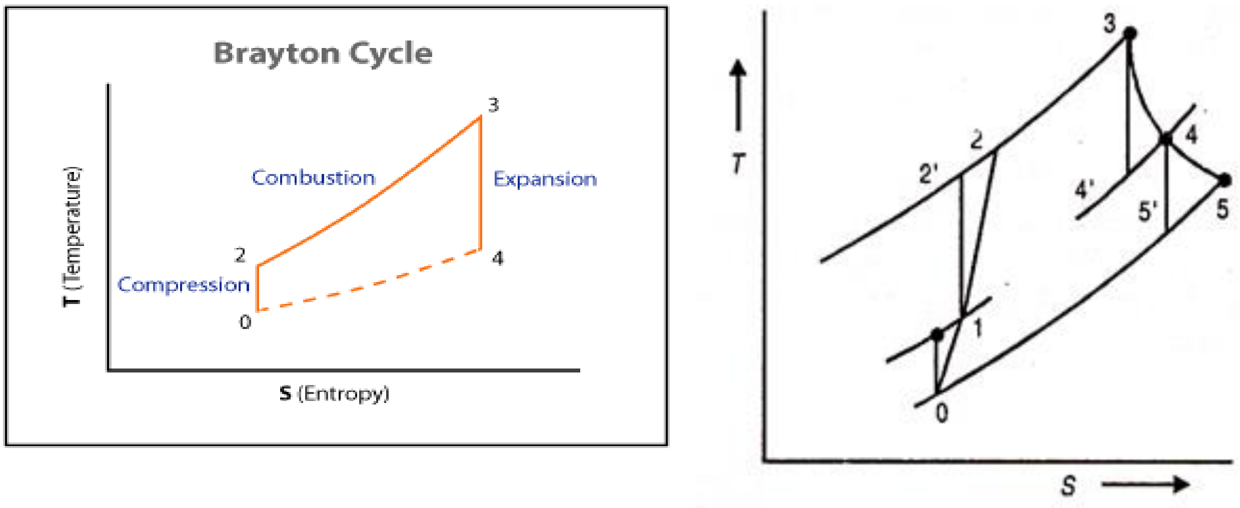

Fig 12: Brayton Cycle

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UNIT – II- INLETS AND NOZZLES – SAE1302

I. Inlets

Most modern passenger and military aircraft are powered by gas turbine engines, which are also called jet engines. There are several different types of gas turbine engines, but all turbine engines have some parts in common. All turbine engines have an inlet to bring free stream air into the engine. The inlet sits upstream of the compressor and, while the inlet does no work on the flow, there are some important design features of the inlet as described on a separate slide. As shown in the figures above, inlets come in a variety of shapes and sizes with the specifics usually dictated by the speed of the aircraft.

Subsonic Inlets

For aircraft that cannot go faster than the speed of sound (like large airliners), a simple, straight, short inlet works quite well. On a typical subsonic inlet, the surface of the inlet, from outside to inside, is a continuous smooth curve with some thickness from inside to outside. The very front (most upstream portion) of the inlet is called the highlight, or the inlet lip. A subsonic aircraft has an inlet with a relatively thick lip.

Supersonic Inlets

An inlet for a supersonic aircraft, on the other hand, has a relatively sharp lip. The inlet lip is sharpened to minimize the performance losses from shock waves that occur during supersonic flight. For a supersonic aircraft, the inlet must slow the flow down to subsonic speeds before the air reaches the compressor. Some supersonic inlets, like the one at the upper right, use a central cone to shock the flow down to subsonic speeds. Other inlets, like the one shown at the lower left, use flat hinged plates to generate the compression shocks, with the resulting inlet geometry having a rectangular cross section. This kind of inlet is seen on the F-14 and F-15 fighter aircraft. There are other, more exotic inlet shapes used on some aircraft for a variety of reasons.

Inlet Efficiency

An inlet must operate efficiently over the entire flight envelope of the aircraft. At very low aircraft speeds, or when just sitting on the runway, free stream air is pulled into the engine by the compressor. In England, inlets are called intakes, which is a more accurate description of their function at low aircraft speeds. At high speeds, a good inlet will allow the aircraft to maneuver to high angles of attack and sideslip without disrupting flow to the compressor. Because the inlet is so important to overall aircraft operation, it is usually designed and tested by the airframe company, not the engine manufacturer. But because inlet operation is so important to engine performance, all engine manufacturers also employ inlet aerodynamicists.

Major Design Variables/Choices

- Inlet total pressure ratio and drag at cruise
- Engine location on wing or fuselage
- Inlet total pressure ratio and distortion envelope
- Engine out flow and drag
- Integration of diffuser and fan flow path contour
- Integration of external nacelle contour with thrust reverser or accessories
- Flow field interaction with nacelle, pylon or wing
- Noise suppression requirements

The engine inlet of a turbine engine is designed to provide a relatively distortion-free flow of air, in the required quantity, to the inlet of the compressor. Many engines use inlet guide vanes (IGV) to help straighten the airflow and direct it into the first stages of the compressor. A uniform and steady airflow is necessary to avoid compressor stall (airflow tends to stop or reverse direction of flow) and excessive internal engine temperatures in the turbine section. Normally, the air-inlet duct is considered an airframe part and not a part of the engine. However, the duct is very important to the engine's overall performance and the engine's ability to produce an optimum amount of thrust.

Keep ducts as short as possible

reduces volume, reduces viscous losses

limits on turning flow without separation

Keep offset ducts long enough to prevent separation

Use the wing and fuselage to shield the inlet, reduce distortion

Watch proximity to ground

Nozzle Design Considerations

1. Accelerate the flow to high velocity with minimum total pressure loss
2. Match exit and atmospheric pressures as closely as desired
3. Permit afterburner operation without affecting main engine operation – requires variable-area nozzle
4. Allow for cooling of walls if necessary
5. Mix core and bypass streams of turbofan if necessary
6. Allow for thrust reversing if desired
7. Suppress jet noise and infrared radiation (IR) if desired
8. Thrust vector control if desired

II. MODES OF INLET OPERATION

Critical Inlet Operation

The condition when the inlet can accept the mass flow of air required to position the terminal shock just inside the cowl lip is called critical inlet operation

Sub - Critical Inlet Operation

The condition when the inlet is not matched to the engine, due to which the normal shock moves upstream and stays in front of cowl lip is called as sub-critical operation

Super - Critical Inlet Operation

The condition when the inlet can not capture the mass flow required by the engine and the terminal shock is sucked into the diffuser is called super – critical operation

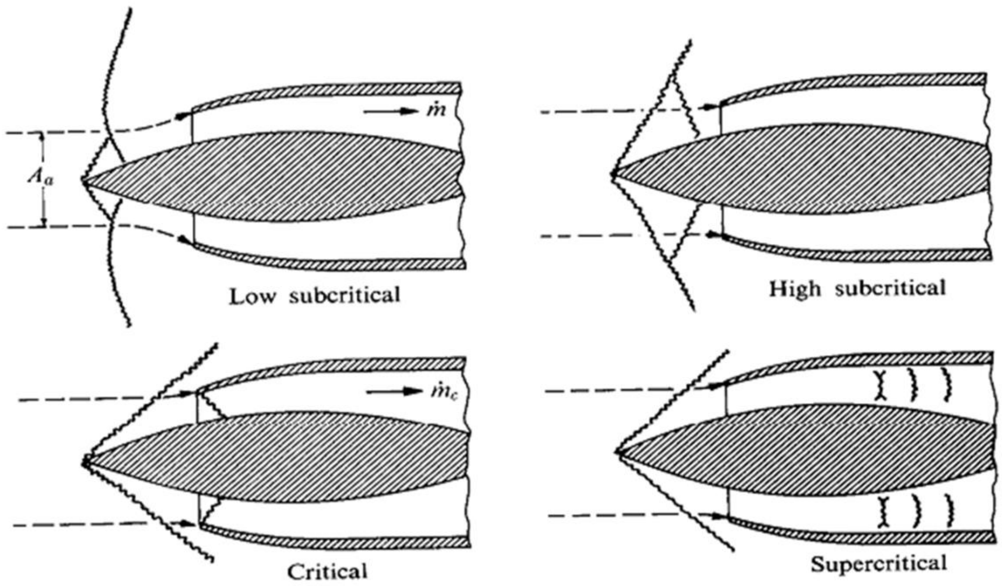

Fig 1. Modes of Inlet Operation

Starting ofthe Diffuser

If the flow arrives undisturbed at the inlet, it could only occupy a fraction of it, the rest of the flow into the frontal area A1 is required to be disposed of which is called "Spillage".This "Spillage" is accomplished by the detached normal shock; behind it the flow is subsonic and it can turn around the inlet. The shock at the full flight Mach number is very lossy, and it is not practical to simply force the plane to continue accelerating to the design condition What can be done is to manipulate the geometry to swallow the shock and reduce its strength. This is called "STARTING" THE DIFFUSER".To "START" THE DIFFUSER, means to pass the shock through the convergent portion, there should be an increase in the throat area until the normal shock is just at the lip. At that point, any further small increase in throat area causes the shock to jump rapidly to a position in the divergent part of the nozzle where the area is again A1.This rapid jumping of shock from converging portion to diverging portion takes place because the shock is unstable in the converging section, but stable in the divergent section.

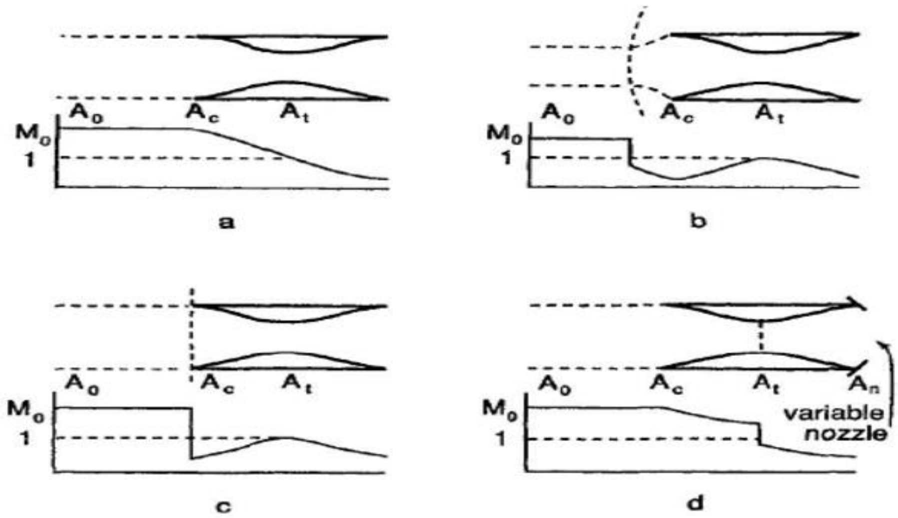

Fig 2. Starting of the Diffuser

III. Nozzles

Subsonic Aircraft: Usually a fixed area convergent nozzle is adequate. It can be more complex for noise suppression The Supersonic Aircraft: More complex, variable-area, convergent-divergent device. The two Primary Functions: Provide required throat area to match mass flow and exit conditions. The efficiently expand high pressure, high temperature gases to atmospheric pressure (convert thermal energy → kinetic energy)

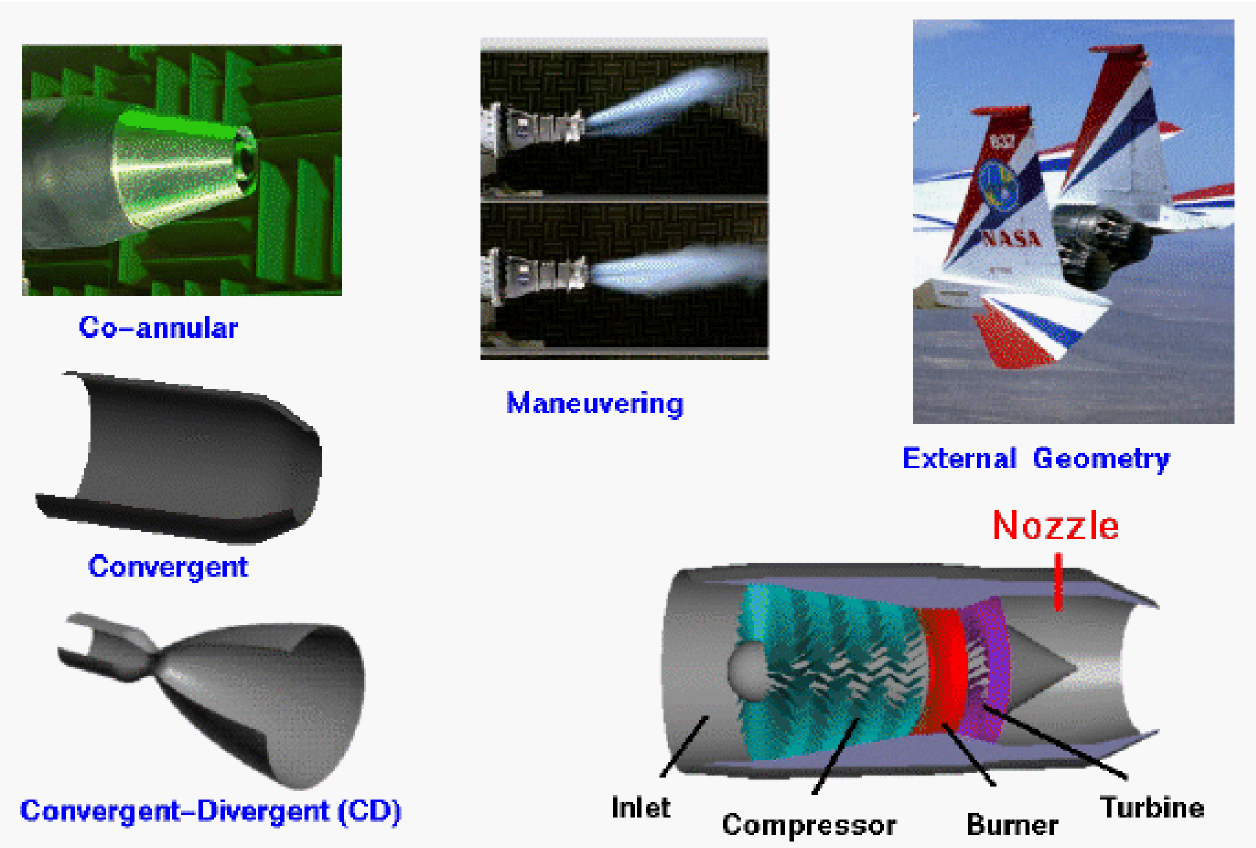

Fig 3. Nozzle

Operation of Converging-Diverging Nozzles

The flow through the nozzle when it is completely subsonic (i.e. nozzle isn't choked). The flow accelerates out of the chamber through the converging section, reaching its maximum (subsonic) speed at the throat. The flow then decelerates through the diverging section and exhausts into the ambient as a subsonic jet. Lowering the back pressure in this state increases the flow speed everywhere in the nozzle.Further lowering pb results in figure (b). The flow pattern is exactly the same as in subsonic flow, except that the flow speed at the throat has just reached Mach 1. Flow through the nozzle is now choked since further reductions in the back pressure can't move the point of M=1 away from the throat. However, the flow pattern in the diverging section does change as the back pressure is lowered further. As p_b is lowered below that needed to just choke the flow a region of supersonic flow forms just downstream of the throat. Unlike a subsonic flow, the supersonic flow accelerates as the area gets bigger. This region of supersonic acceleration is terminated by a normal shock wave. The shock wave produces a near-instantaneous deceleration of the flow to subsonic speed. This subsonic flow then decelerates through the remainder of the diverging section and exhausts as a subsonic jet. In this regime if the back pressure is lowered or raised the length of supersonic flow in the diverging section before the shock wave increases or decreases.

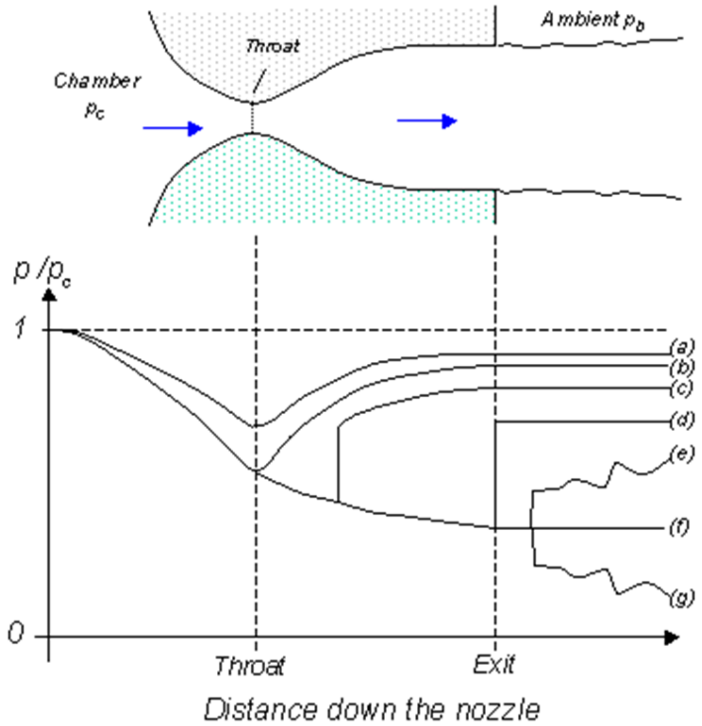

Fig 4.CD Nozzle

If pb is lowered enough the supersonic region may be extended all the way down the nozzle until the shock is sitting at the nozzle exit, figure (d). Because of the very long region of acceleration (the entire nozzle length) the flow speed just before the shock will be very large. However, after the shock the flow in the jet will still be subsonic.Lowering the back pressure further causes the shock to bend out into the jet, figure (e), and a complex pattern of shocks and reflections is set up in the jet which will now involve a mixture of subsonic and supersonic flow, or (if the back pressure is low enough) just supersonic flow. Because the shock is no longer perpendicular to the flow near the nozzle walls, it deflects it inward as it leaves the exit producing an initially contracting jet. We refer to this as over-expanded flow because in this case the pressure at the nozzle exit is lower than that in the ambient (the back pressure)- i.e. the flow has been expanded by the nozzle too much. A further lowering of the back pressure changes and weakens the wave pattern in the jet. Eventually, the back pressure will be lowered enough so that it is now equal to the pressure at the nozzle exit. In this case, the waves in the jet disappear altogether, figure (f), and the jet will be uniformly supersonic. This situation, since it is often desirable, is referred to as the ‘design condition’, Pe=Pa.

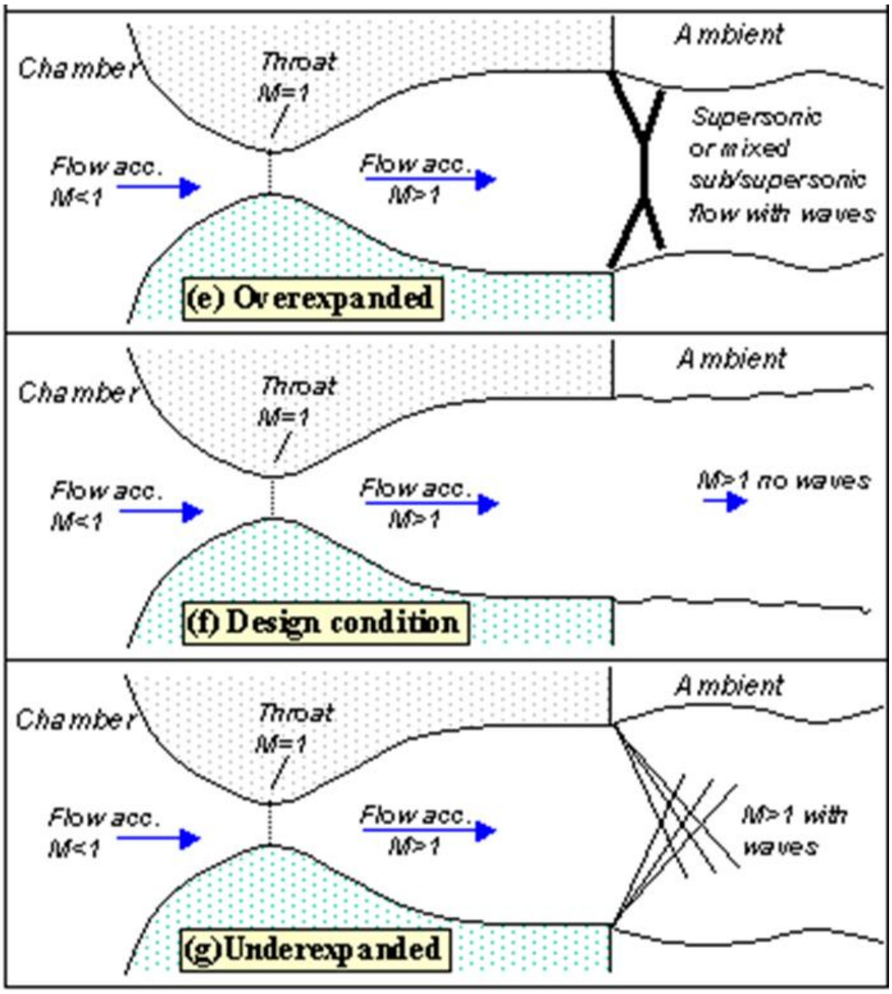

Fig 4.Design condition of the Nozzle

Finally, if the back pressure is lowered even further we will create a new imbalance between the exit and back pressures (exit pressure greater than back pressure), figure (g). In this situation, called under-expanded, expansion waves that produce gradual turning and acceleration in the jet form at the nozzle exit, initially turning the flow at the jet edges outward in a plume and setting up a different type of complex wave pattern.

Summary Points to Remember:

- When flow accelerates (sub or supersonically) pressure always drops
- Pressure rises instantaneously across a shock
- Pressure falls across an expansion wave
- Pressure throughout jet is always same as ambient (i.e. the back pressure) unless jet is supersonic and there are shocks or expansion waves in jet to produce pressure differences

Environmental issues are likely to impose fundamental limitation on air transportation growth in the 21st century. The two major contributors: Noise and Emissions

Noise :Primarily exhaust jet and fan (whirl) noise Noise impact of subsonic aircraft is constrains air transportation system through curfews, noise budgets and slot restrictions. Some solutions are Exhaust mixers, Liners that absorb sound and Shaping of stators and fan blade components for low noise.

IV. Types of Nozzle

Convergent nozzles are normally used in subsonic aircraft.These nozzles operate under choked condition, leading to incomplete expansion.This may lead to a pressure thrust. A C-D nozzle can expand fully to the ambient pressure and develop greater momentum thrust.However due to increased weight,geometric complexity and diameter, it is not used in subsonic transport aircraft

Variable Geometry Nozzles

Variable area nozzles or adjustable nozzles are required for matched operation under all operating conditions.Three types of variable area nozzles are:

– Central plug at nozzle outlet
– Ejector type
– Iris nozzle

• The Central plug is very similar to the spike of an intake. Unlike intake, the central plug causes external expansion fans.

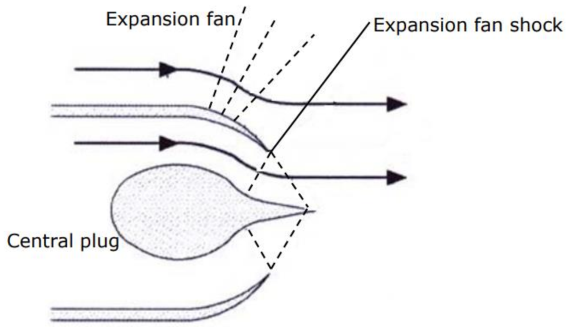

Fig 5.Centre plug Nozzle

Ejector nozzle

It creates an effective nozzle through a secondary airflow. At subsonic speeds, the airflow constricts the exhaust to a convergent shape. As the speed increases, the two nozzles dilate and the two nozzles form a CD shape

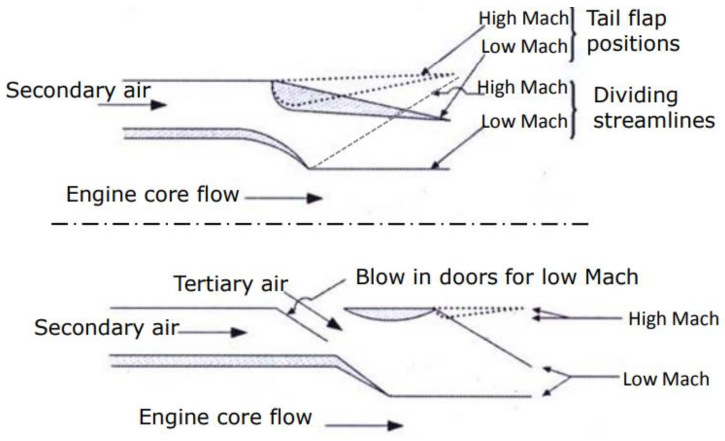

Fig 6 Ejector Nozzle

Iris nozzle

It uses overlapping, adjustable petals. • More complicated than the ejector type nozzle. • Offers significantly higher performance. • Used in advanced military aircraft. •

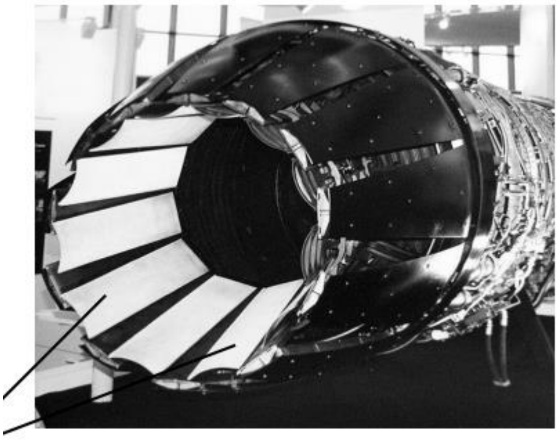

Fig 7 Iris Nozzle

Some of the modern aircraft also have iris nozzles that can be deflected to achieve vectored thrust.

V. Thrust Vectoring

Thrust vectoring was originally developed as a means for V/STOL (Vertical or Short Take Off and Landing).Thrust vectored aircraft have better climb rates, besides extreme manoeuvres. Most of the modern day combat aircraft have thrust vectoring. Some of the latest aircraft also have axisymmetric nozzle thrust vectoring.

There are two types of thrust vector controls:

– Mechanical control

– Fluidic control

Mechanical control involves deflecting the engine nozzle and thus physically alter the direction of thrust. Fluidic vectoring involves either injecting fluid or removing it from the boundary layer of the primary jet. Mechanical vectoring system is heavier and complex.

There are two types of mechanical thrust vectoring

Internal thrust vectoring- Internal thrust vectoring permits only pitch control

External thrust vectoring- External thrust vectoring can be used for pitch and yaw controls

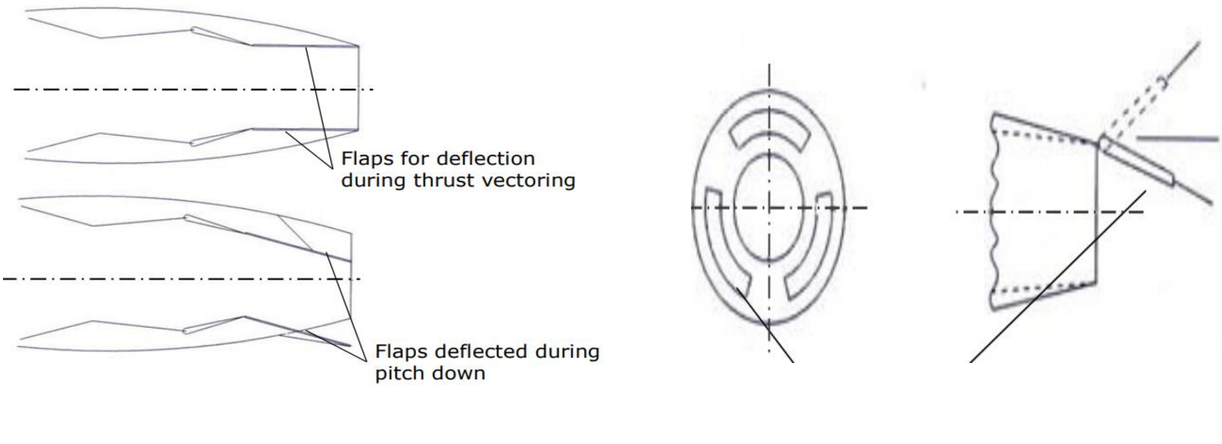

Fig 8 Internal and External thrust vectoring

Fluidic thrust vectoring

Fluidic thrust vectoring has been demonstrated successfully at a laboratory scale..This method has several advantages over the mechanical control. Main challenge lies in ensuring an effective control with a linear response.Other concepts like Shock thrust vector control, coflow and counter flow thrust vectoring concepts are also being pursued.

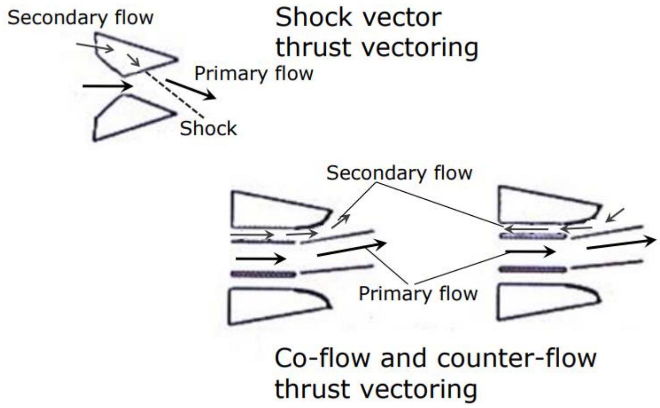

Fig 9 Fluidic thrust vectoring

VI. Thrust Reversal

With increasing size and loads of modern day aircraft, wheel brakes alone cannot brake and aircraft. Deflecting the exhaust stream to produce a component of reverse thrust will provide an additional braking mechanism.Most of the designs of thrust reversers have a discharge angle of about 45 degree. Therefore a component of the thrust will now have a forward direction and therefore contributes to braking.

There are three types of thrust reversal mechanisms that are used

– Clamshell type
– External bucket type
– Blocker doors

Clamshell type: is normally pneumatically operated system. When deployed, doors rotate and deflect the primary jet through vanes. These are normally used in non-afterburning engines.

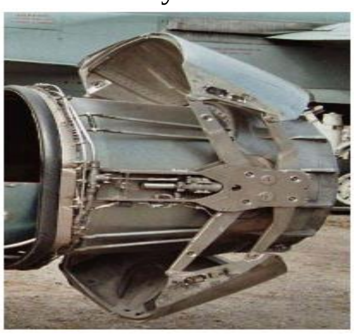

Fig 10 Clamshell type thrust reverser

Bucket type system uses bucket type doors to deflect the gas stream. In normal operation, the reverser door form part of the convergent divergent nozzle. Blocker doors are normally used in high bypass turbofans. The cold bypass flow is deflected through cascade vanes to achieve the required flow deflection.

Fig 11 Bucket type thrust reverser

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SCHOOL OF MECHANICAL ENGINEERING

DEPARTMENT OF AERONAUTICAL ENGINEERING

UNIT – III- COMBUSTION CHAMBERS – SAE1302

I. COMBUSTION CHAMBERS

The combustion chamber conversion of fuel chemical energy to heat energy. The combustion chamber 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

Requirements for Combustion Chamber

1. Easy and safety mixture ignition in every working conditions
2. Stable mixture burning in every engine mode
3. Uniform pressure and velocity field in outlet Low losses
4. Short length of flame

$$PO3 = PO4(IDEALCASE)$$

$$REALCASE, P03 <= PO4$$

$$\left(\frac{PO3 - P04}{P03}\right) * 100 = \forall P\%$$

$$\left(1 - \frac{P04}{P03}\right) = \forall P$$

$$PO4 = (1 - \forall P) * PO3$$

$$IDEAL, Q = m_f * C.V$$

$$REALCASE, Q = m_f * C.V * \Im_{CC}$$

$$Q = m * C. * dT$$

$$m_f * C.V * \Im_{CC} = m_g * Cp_g * T04 - m_a * Cp_a * T03$$

$$m_f * C.V * \Im_{CC} = (m_a + m_f) * Cp_g * T04 - m_a * Cp_a * T03$$

$$divide, ma, onbothsides$$

$$f * C.V * \Im_{CC} = (1 + f) * Cp_g * T04 - Cp_a * T03$$

$$f = \frac{Cp_g * T04 - Cp_a * T03}{C.V * \Im_{CC} - Cp_g * T04}$$

Main Problems For Combustion

- High heat dilatation
- Problematic ignition Stability of burning
- Materials

Combustion Process

Air from compressor enters the combustion chamber at a velocity approximately 100-150 $m.s^{-1}$.At first air is apportioned in Combustion Chamber snout to: Primary flow (20-40% of air) and Secondary flow (60-80% of air). In primary flow provides a fuel burning and dilution, this velocity of inlet air from compressor is far too high for combustion and it is necessary to decelerate it. Deceleration is provided by swirl vanes and flare. Air is decelerated to speed of burning which is around 15 to 20 $m.s^{-1}$. In primary zone is air mixed with fuel from spray nozzle and it is ignited by electric spark. Temperature in core of burning achieve 1800-2000 °C. Air from secondary flow enter through holes and cool the flame tube. In dilution zone air from secondary stabilize and make uniform the outlet flow. Combustion in the normal, open cycle, gas turbine is a continuous process in which fuel is burned in the air supplied by the compressor; an electric spark is required only for initiating the combustion process, and thereafter the flame must be self-sustaining. Combustion of a liquid fuel involves the mixing of a fine spray of droplets, vaporization of the droplets, the breaking down of heavy hydrocarbons into lighter fractions, the intimate mixing of molecules of these hydrocarbons with oxygen molecules, and finally the chemical reactions themselves. A high temperature, such as is provided by the combustion of an approximately stoichiometric mixture, is necessary if all these processes are to occur sufficiently rapidly for combustion in a moving air stream to be completed in a small space. But in actual practice A/F ratio is in the range of 100:1, while ratio is around 15:1. This is to reduce the turbine inlet temperatures due to practical limits.

II. Zones of Combustion Chamber

Primary zone (15-20% air) Air is introduced around the jet of fuel burns at approximately the Stoichiometric Ratio. Therefore, high temperature and thus, Rapid Combustion occurs.

Secondary Zone (30% air): Introduced through holes in the flame-tube in the secondary zone to complete the combustion. For high combustion efficiency, air must be injected carefully at the right points in the process, to avoid chilling the flame locally and drastically reducing the reaction rate in that neighborhood

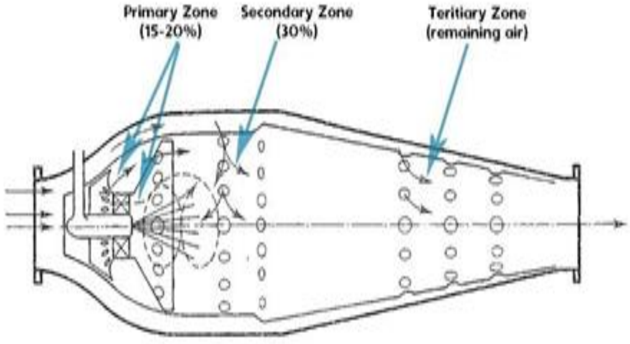


Fig 1: Zones of Combustion Chamber

Tertiary Zone (remaining air): These zones have Dilution Zone and Cooling. The Sufficient turbulence must be promoted so that the hot and cold streams are thoroughly mixed to give the desired outlet temperature distribution, with no hot streaks which would damage the turbine blades.

Factors influencing design

Low turbine inlet temperature. Uniform temperature distribution at turbine inlet (i.e., to avoid local heating of turbine blades). The Stable operation even when factors like air velocity, A/F ratio & pressure varies greatly, especially for aircraft engines. The limit is the '*flame- out'* of the combustion chamber & at the event of a flame-out the combustor should be able to relight quickly. The formation of carbon deposits ('coking') must be avoided. Can damage the turbine if breaks free. Aircraft engines should avoid visible smoke as it hinders visibility in airports. Finally, Pollutants like NO_x, CO, Unburned Hydrocarbons (UHCs) etc. should be limited.

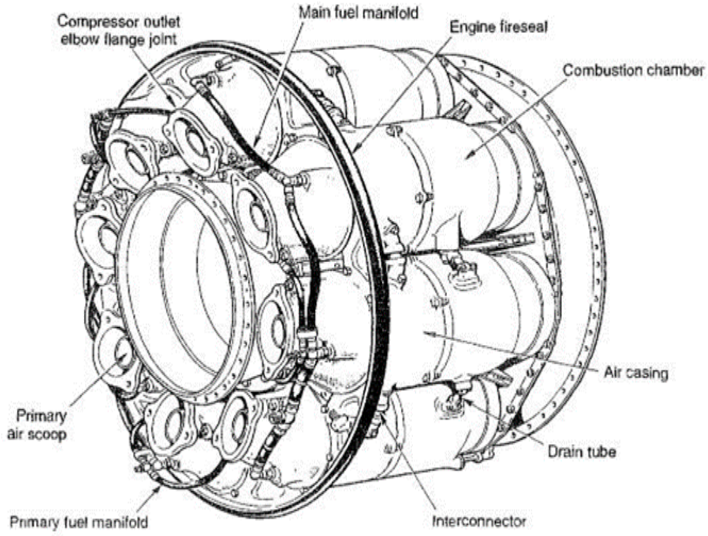
III. Types of Combustion Chamber

The combustion chamber works on heat addition with constant pressure. The adiabatic process involves in the aircraft combustion chamber. The combustion chamber is classified into three types.

1. Multiple (Can or Tubular)
2. Annular
3. Turbo-annular (Can-Annular)

Can Combustion Chamber

The earliest aircraft engines made use of can (or tubular) combustors. Air leaving the compressor is split into a number of separate streams, each supplying a separate chamber. These chambers are spaced around the shaft connecting the compressor and turbine, each chamber having its own fuel jet fed from a common supply line. The well suited to engines with centrifugal compressors, where the flow is divided into separate streams in the diffuser.



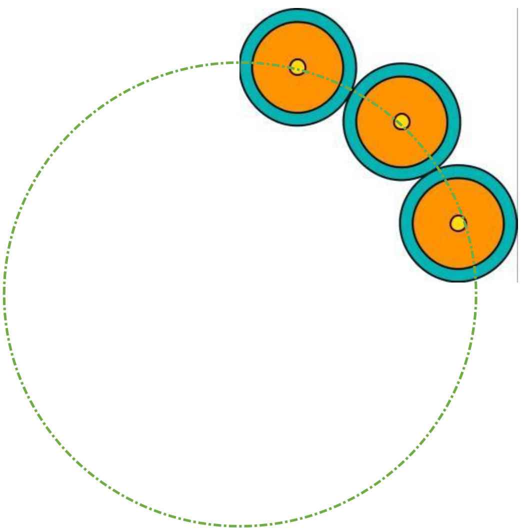


Fig 2 : Can Combustion Chamber

Merits

1. Easier development could be carried out
2. On a single can using only a fraction of the overall airflow and fuel flow

De-Merits

1. Increased Volume, weight & frontal area
2. Increased Pressure drop (more surface area in contact with air/gas)

Tubo-annular Combustion Chamber (Can - annular)

Individual flame tubes are uniformly spaced around an annular casing. Uses a reverse flow arrangement which allows a significant reduction in the overall length of the compressor-turbine shaft and also permits easy access to the fuel nozzles and combustion cans for maintenance.

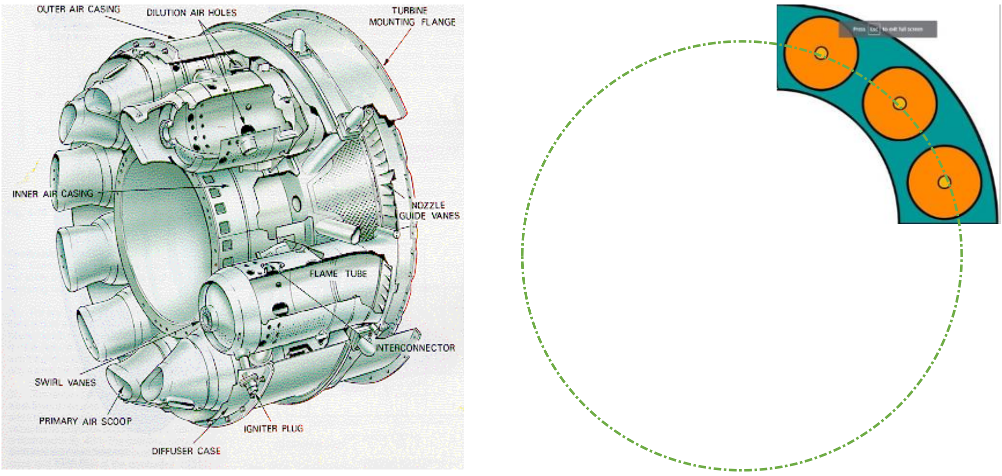


Fig 3 : Can-Annular Combustion Chamber

Merits

1. Reduced shaft length
2. Easy maintenance
3. Easier development

Demerits

1. Increased Volume, weight & frontal area
2. Increased Pressure drop

Annular Combustion Chamber

The ideal configuration in terms of compact dimensions is the annular combustor, in which maximum use is made of the space available within a specified diameter; this should reduce the pressure loss and results in an engine of minimum diameter. The combustion does not take place in individual flame tubes, but instead in an annular region around the engine.

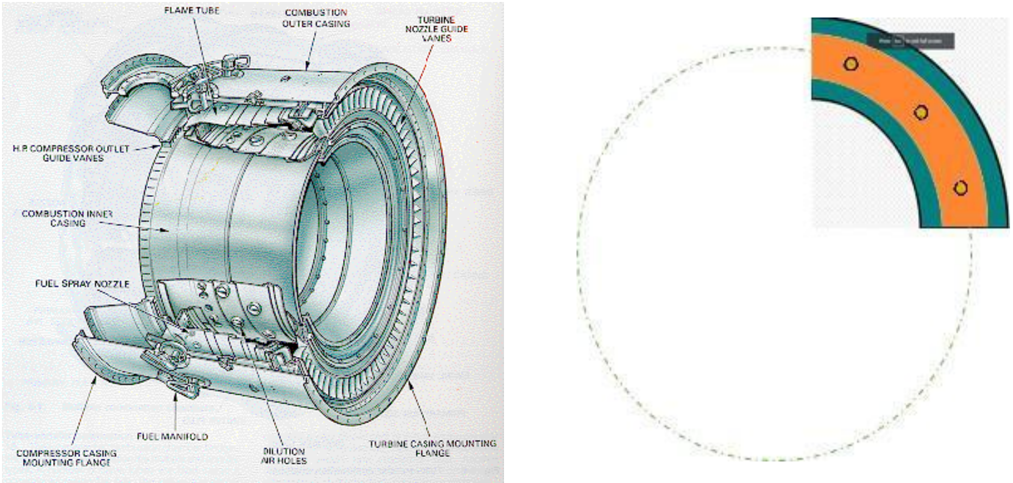


Fig 4 : Annular Combustion Chamber

Merits

1. Reduces the pressure loss (less surface exposed to air/gas flow)
2. Compact size

Demerits

1. Less structural integrity
2. Difficult to obtain even temperature distribution

Flame stability

The zonal method of introducing the air cannot by itself give a self-piloting flame in an air stream which is moving an order of magnitude faster than the flame speed in a burning mixture. The second essential feature is therefore a recirculating flow pattern which directs some of the burning mixture in the primary zone back on to the incoming fuel and air.

Recirculating flow pattern is achieved by,

1. Swirl vanes
2. Holes downstream of a hemispherical Baffle
3. Upstream injection
4. Vaporizer system (walking stick/T-shaped tubes)

Performance Parameters

Fundamental Loss

The rise in temperature during combustion. An increase in temperature implies a decrease in density and consequently an increase in velocity and momentum of the stream. A pressure drop must be present to impart the increase in momentum.

Cold Loss

Pressure Loss due to,

- Skin friction
- Turbulence

The pressure loss due to friction is found to be very much higher than that due to combustion, mainly due to turbulence, which is required for proper mixing and Temperature uniformity.

Stability Loop

For any particular combustion chamber there is both a rich and a weak limit to the air/fuel ratio beyond which the flame is unstable. Usually the limit is taken as the air/fuel ratio at which the flame blows out, although instability often occurs before this limit is reached.

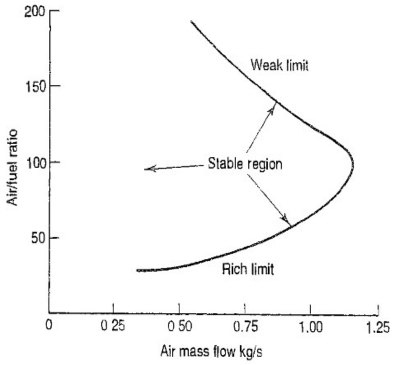


Fig 5 : Stability Loop

The range of air/fuel ratio between the rich and weak limits is reduced with increase of air velocity, and if the air mass flow is increased beyond a certain value it is impossible to initiate combustion at all.

Combustion Intensity

Lower the combustion intensity easier to design a system with desired requirements. It cannot compare to systems based on efficiency and pressure loss if they vary widely in the combustion intensity.

Flame Holder

A flame holder is a component of a jet engine designed to help maintain continual combustion. All continuous-combustion jet engines require a flame holder. A flame holder creates a low-speed eddy in the engine to prevent the flame from being blown out.

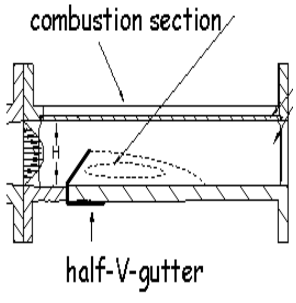


Fig 6 : Flame Holder

The design of the flame holder is an issue of balance between a stable eddy and drag. The H-gutter flame holder, which is shaped like a letter H with a curve facing and opposing the flow of air. The V-gutter flame holder, which is shaped like a V with the point in the direction facing the flow of air.

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SCHOOL OF MECHANICAL ENGINEERING

DEPARTMENT OF AERONAUTICAL ENGINEERING

UNIT – IV- COMPRESSORS– SAE1302

I. Compressor

Turbomachinery is at the heart of gas turbine engines. The role of mechanical compression of air in an engine is given to the compressor. The shaft power to drive the compressor typically is produced by expanding gases in the turbine. The machines that exchange energy with a fluid, called the working fluid, through shaft rotation are known as turbomachinery. The machines where the fluid path is predominantly along the axis of the shaft rotation are called axial-low turbomachinery. In contrast to these machines, in radial-low turbomachinery the fluid path undergoes a 90◦ turn from the axial direction. These machines are sometimes referred to as centrifugal machines. A mixed-low turbomachinery is a hybrid between the axial and the radial-low machines. In aircraft gas turbine engines, the axial-low compressors and turbines enjoy the widest application and development. The centrifugal compressors and radial-low turbines are used in small gas turbine engines and automotive turbocharger applications.

In turbomachinery, energy transfer between the blades and the fluid takes place in an inherently unsteady manner. This is achieved by a set of rotating blades, called the rotor. The rotor blades are three-dimensional aerodynamic surfaces, which experience aerodynamic forces. The rotor blades are cantilevered at the hub and thus feel a root bending moment and a torque. The reaction to the blade forces and moments is exerted on the fluid, via the action–reaction principle of Newton. Stationary blades called the stator follow the rotor blades in what is known as a turbomachinery stage. The stator blades are three-dimensional aerodynamic surfaces as well. They are cantilevered from the casing and experience forces and moments, like the rotor. The exception is that the stator forces and moments are stationary (in the laboratory frame of reference) and thus perform no work on the fluid. The energy of the fluid is, thus, expected to remain constant in passing through the stator blades. The position of the observer is, however, important in viewing the flow field and the energy exchange in turbomachinery. An observer fixed at the casing (or laboratory) is called an absolute observer. If the observer is attached to the rotor blade and spins with it, then it is called a relative observer. The frames of reference are then called the absolute and relative frames of reference, respectively. Consider an isolated rotor in a cylindrical duct. An absolute observer sees the blades' aerodynamic forces are in motion, at an angular rate, that is, the angular velocity with respect of the shaft. Hence, as measured by this observer, the total enthalpy of the fluid goes up in crossing the rotor row. On the contrary, let us put ourselves in the frame of reference of a relative observer who is spinning with the rotor. According to a relative observer, the blades are not moving! An observer fixed at the rotor measures aerodynamic forces and moments of the blades, however, as the forces are stationary, there is no work done on the fluid according to this observer. Thus, the relative observer measures the same total enthalpy across the blade row. The flow field as seen by a relative observer attached to an isolated rotor in a cylinder is thus steady. The absolute observer on the casing, however, sees the passing of the blades and thus experiences an unsteady flow field. As the rotor blades pass by, a periodic pressure pulse (due to blade tip) is registered at the casing, which signifies an unsteady event with a periodicity of blade passing frequency.

To be able to analyse a flow field in a steady frame of reference offers tremendous advantages in the nature and the solution of governing equations. Consequently, in analysing the low within a rotor blade row, employ the relative observer stance, while the stator lows are viewed from the standpoint of an absolute observer. We need to be mindful, however, that in practice there are no isolated rotors and thus the flow field in rotating machinery is inherently unsteady The velocity components as seen by observers in the two frames of reference are related. First, we note that the radial and axial velocity components are identical in the two frames, as the relative observer moves only in the angular, direction. Therefore, the swirl or tangential velocity is the only component of the velocity vector field that is affected by the observer rotation. At a radial position on the rotor, the relative observer rotates with a speed and thus registers a tangential velocity, which is less than the absolute swirl velocity.

II. Axial flow Compressor

An axial compressor is a machine that can continuously pressurize air. It is a rotating, air foil-based compressor in which the gas or working fluid principally flows parallel to the axis of rotation. Axial flow compressors produce a continuous flow of compressed air, have the benefits of high efficiency and large mass flow rate, particularly in relation to their size and cross-section. They do, however, require several rows of air foils to achieve a large pressure rise, making them complex and expensive relative to other designs. Axial compressors are integral to the design of large gas turbines such as jet engines, high speed ship engines, and small scale power stations. Due to high performance, high reliability and flexible operation during the flight envelope, they are also used in aerospace engines.

Construction

Axial flow compressor consists of casing fitted with several rows of fixed blades & several rows of moving blades which are attached on rotor as shown in fig. The fixed blades are placed on alternative rows. The fixed blades & moving blades are as possible for efficient flow. The one set of rotor blades & one set of stator blades called stage. The number of stages in axial flow compressor depends upon pressure ratio required. Usually 5 to 14 stages are used. The length of blades is reduced in direction of flow to compensate for the reduction in volume resulting from the increased pressure. The blades are so arranged that the spaces between blades form diffuser passage& hence velocity of air is reduced as it passes through them & pressure increases. Axial flow compressor is also high speed machine & speed may even vary from 10000 to 30000 RPM. Generally, maximum pressure ratio achieved in a stage of axial compressor is about 1.12 to 1.2, hence to obtain pressure ratio of 12, attainable by axial flow compressor 15 to 20 stages are required.

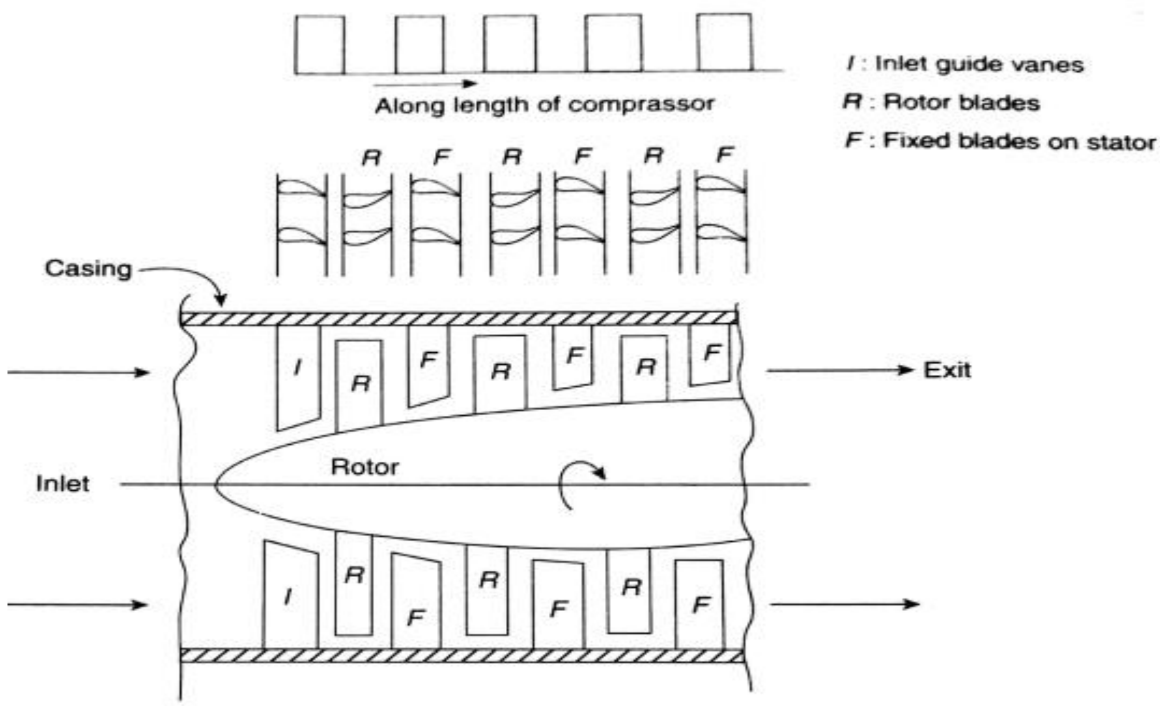


Fig 1: Axial Flow Compressor

Working

As the fluid enters and leaves in the axial direction, the centrifugal component in the energy equation does not come into play. Here the compression is fully based on diffusing action of the passages. The diffusing action in stator converts absolute kinetic head of the fluid into rise in pressure. The relative kinetic head in the energy equation is a term that exists only because of the rotation of the rotor. The rotor reduces the relative kinetic head of the fluid and adds it to the absolute kinetic head of the fluid i.e., the impact of the rotor on the fluid particles increases its velocity (absolute) and thereby reduces the relative velocity between the fluid and the rotor. In short, the rotor increases the absolute velocity of the fluid and the stator converts this into pressure rise. The increase in pressure produced by a single stage is limited by the relative velocity between the rotor and the fluid, and the turning and diffusion capabilities of the air foils. A typical stage in a commercial compressor will produce a pressure increase of between 15% and 60% at design conditions with a polytrophic efficiency in the region of 90–95%.To achieve different pressure ratios, axial compressors are designed with different numbers of stages and rotational speeds. As a general rule-of-thumb we can assume that each stage in a given compressor has the same temperature rise (δT).Therefore, at the entry, temperature (Tstage) to each stage must increase progressively through the compressor and the ratio (δT)/(Tstage) entry must decrease, thus implying a progressive reduction in stage pressure ratio through the unit. Hence the rear stage develops a significantly lower pressure ratio than the first stage.Higher stage pressure ratios are also possible if the relative velocity between fluid and rotors is supersonic, but this is achieved at the expense of efficiency and operability.Such compressors, with stage pressure ratios of over 2, are only used where minimizing the compressor size, weight or complexity is critical, such as in military jets.

A axial Compressor consists of stator and rotor blades placed alternatively from the Inlet end to the exit end It has a moving inner core called rotor and static outer portion called stator or casing. The rotor has a set of blades mounted on it which rotate with the rotor. The casing or stator has static blades mounted on it. The fluid is admitted in the compressor from the inlet end through the inlet guide vanes smoothly impinge upon rotor blades. flows over the rotor blades to the Stator blades and then again on rotor blade, and Stator blade

The working of axial flow compressors is based upon the addition of kinetic energy to the flowing fluid by the rotor blades and its subsequent conversion into the pressure rise. Here the fluid enters axially. through the inlet guide Vanes, to the rotor blades at a suitable angle to ensure smooth flow. Then the fluid is rotated by the rotor blades and its kinetic energy gets increased. During this process, there occurs a very small rise in the pressure too. The rotor blades then discharge the fluid to the stator blades where the maximum rise in the pressure occurs due to the diffusion in the stator section. The fluid subsequently enters into the rotor blades which are followed by the stator blades and the process continues so on till the exit end.. The change in the total values of the pressure and temperature, and enthalpy occurs only in the rotor section. In an axial flow compressor, the fluid successively passes through the compressor stages causing slight rise in the pressure and temperature. This low pressure ratio of the order of 1.1 to 1.4 offers high efficiency and the high pressure ratio up to 40 may be obtained.

Velocity Triangles

The flow geometry at, the entry and exit Of a turbomachine stage is described by the velocity triangles at these stations.

As already mentioned earlier, the velocity triangles for a turbomachine contain the following three components

)i(the peripheral velocity,)u(, of the rotor blades,

)ii(the absolute velocity,)c(, of the fluid, and

)iii(the relative Velocity,)w(, of the fluid.

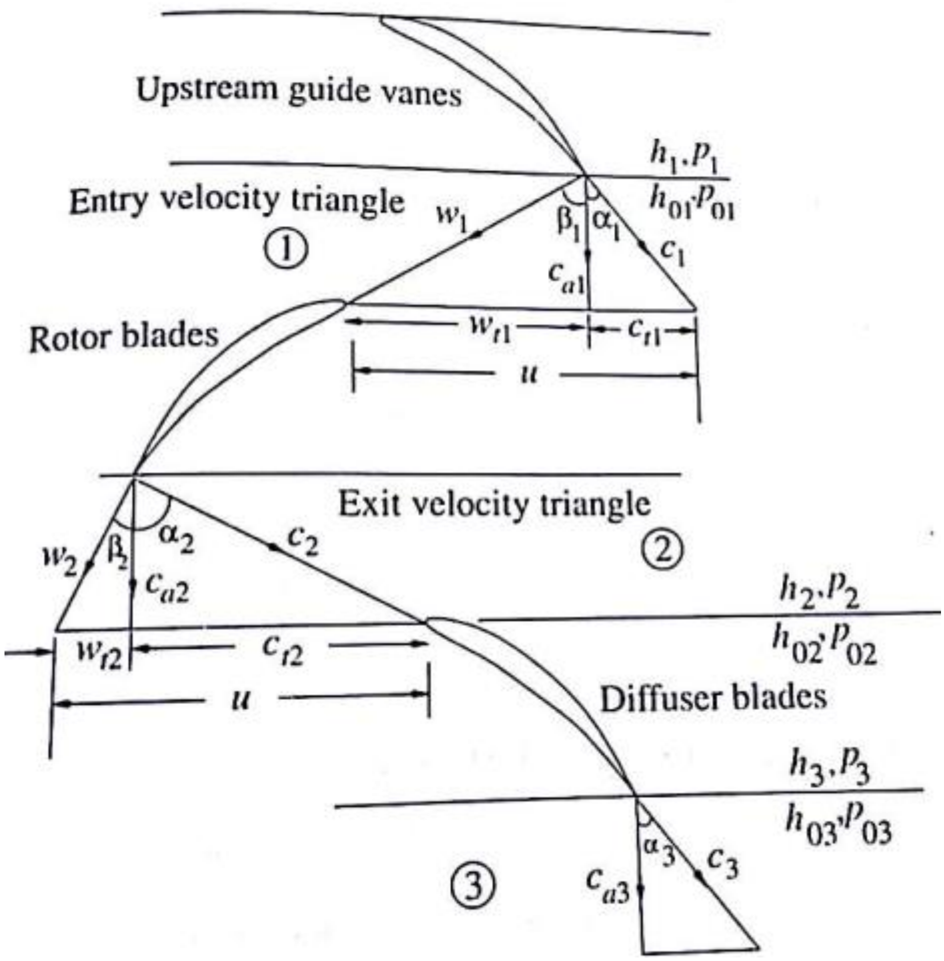


Fig 2: Velocity triangle for Axial Flow Compressor

These velocities are related by the following well-known vector equation:

$$\vec{c} = \vec{u} + \vec{w}$$

This simple relation is frequently used and is very useful in drawing the velocity triangles for turbomachines. The notation used heré to draw velocity triangles correspond to the x-y coordinates; the suffix)a(identifies components in the axial direction and the suffix)t(refers to the tangential direction. Air angles in the absolute system are denoted by alpha)α(, where as those in the relative system are represented by beta)β(

From velocity triangles.

$$C_{a1} = C_1 \cos\alpha_1 = W_1 \cos\beta_1 \quad \text{— (1)}$$

$$C_{t1} = C_1 \sin\alpha_1 = C_{a1} \tan\alpha_1 \quad \text{— (2)}$$

$$W_{t1} = W_1 \sin\beta_1 = C_{a1} \tan\beta_1 \quad \text{— (3)}$$

$$u = C_{t1} + W_{t1} \quad \text{— (4)}$$

similarly

$$C_{a2} = C_2 \cos\alpha_2 = W_2 \cos\beta_2 \quad \text{— (5)}$$

$$C_{t2} = C_2 \sin\alpha_2 = C_{a2} \tan\alpha_2 \quad \text{— (6)}$$

$$W_{t2} = W_2 \sin\beta_2 = C_{a2} \tan\beta_2 \quad \text{— (7)}$$

$$u = C_{t2} + W_{t2} \quad \text{— (8)}$$

From eqn 4 & 8 $\rightarrow$

$$C_{t1} + W_{t1} = C_{t2} + W_{t2}$$

$$C_{a1} \tan\alpha_1 + C_{a1} \tan\beta_1 = C_{a2} \tan\alpha_2 + C_{a2} \tan\beta_2$$

But $C_a = C_{a1} = C_{a2}$.

$$C_a [\tan\alpha_1 + \tan\beta_1] = \tan\alpha_2 + \tan\beta_2] C_a$$

$$C_a [\tan\beta_1 - \tan\beta_2] = [\tan\alpha_2 - \tan\alpha_1] C_a$$

$$C_{t2} - C_{t1} = W_{t1} - W_{t2}$$

work done by the compressor

$$W = u(C_{t2} - C_{t1}) = C_a u [\tan\alpha_2 - \tan\alpha_1]$$

Degree of Reaction

A degree of Reaction for axial compressors can also be defined a ratio of actual change of the enthalpy in the rotor to actual change of enthalpy in the stage

$$R = \frac{h_2 - h_1}{h_{03} - h_{01}} \qquad \text{— (1)}$$

$$= \frac{1}{2u}\frac{w_1^2 - w_2^2}{(C_{t2} - C_{t1})} \qquad \text{— (2)}$$

useing Phythagoras theorom

$$w_1^2 = w_{t1}^2 + Ca_1^2$$

$$w_2^2 = w_{t2}^2 + Ca_2^2$$

$$w_1^2 = w_2^2 = w_{t1}^2 - w_{t2}^2$$

$$R = \frac{w_{t1}^2 - w_{t2}^2}{2u\,(C_{t2} - C_{t1})} \qquad \text{— (3)}$$

$$R = \frac{(w_{t1} + w_{t2})\,(w_{t1} - w_{t2})}{2u\,(C_{t2} - C_{t1})}$$

From velocity triangles

$$w_{t1} - w_{t2} = C_{t2} - C_{t1}$$

$$R = \frac{w_{t1} + w_{t2}}{2u} \qquad \text{— (4)}$$

$$R = \frac{Ca_1 \tan\beta_1 + Ca \tan\beta_2}{2u}$$

$$R = \frac{Ca}{2u}\left[\tan\beta_1 + \tan\beta_2\right] \qquad \text{— (5)}$$

Compressor Characteristics

The characteristic curves indicate that the pressure rise is quite steep at higher speeds of the rotation.Surging starts occurring even before the curves achieve the maximum value and due to this the design point is close to the peak of characteristic curve near the surge line.Even a small reduction in the mass flow will show the increase in the pressure ratio and density and this will cause reduction in the axial velocity causing stalling

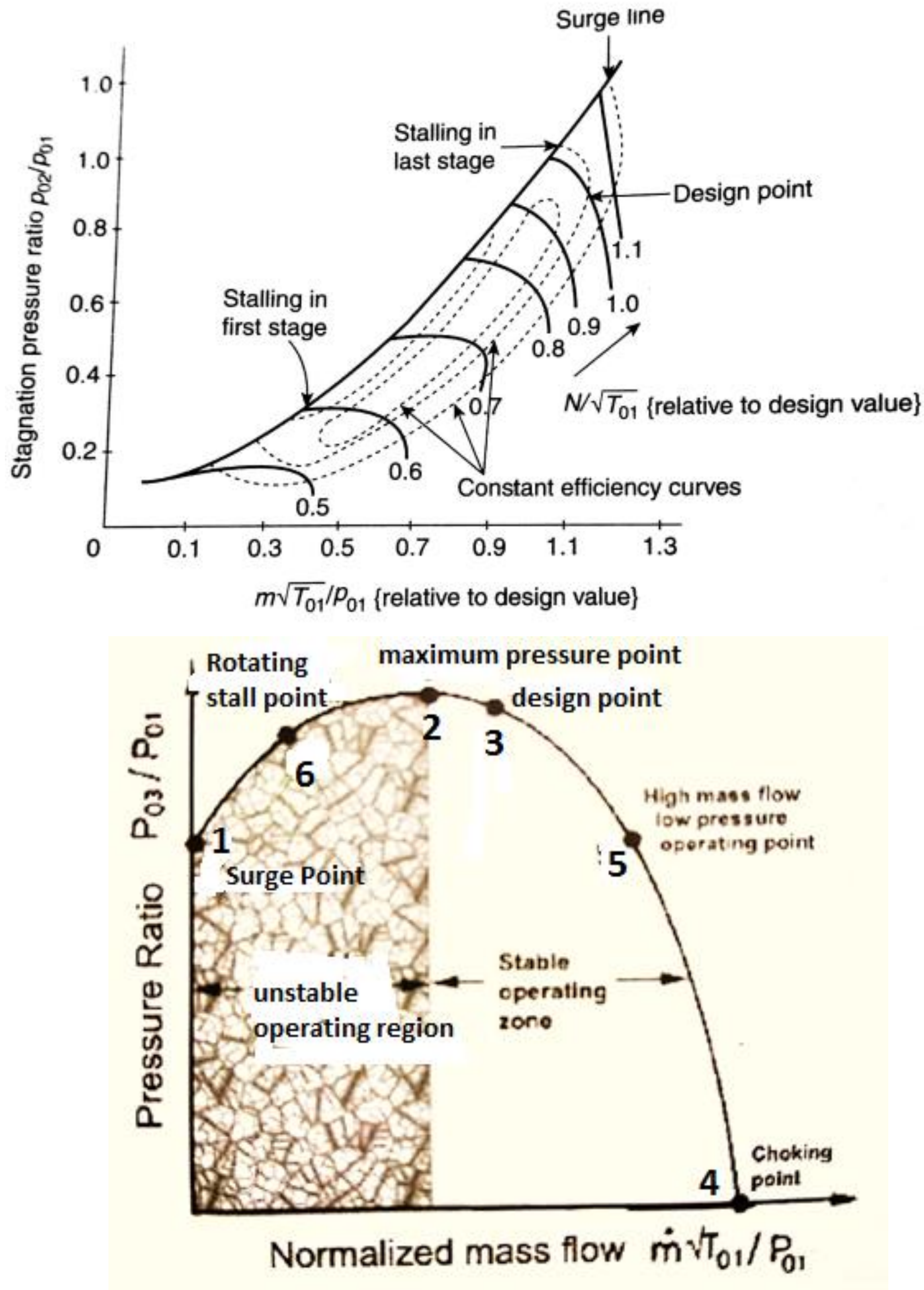


Fig 3: Characteristic lines for Compressor

Even small increase in the mass flow will cause drop in the pressure ratio and so reduction in the density causing increase in the axial velocity causing stalling at later stages.The reduction in the speed of rotation will cause reduction in the mass flow, axial velocity and blade, thus stalling in the initial stageThe stable operation range is quite narrow and so the use of axial flow compressor needs high care while operating at off design conditions. Quite large speed of rotation leads to nearly vertical characteristic curve, thus increase in the density pressure ratio, speed U and so increased mass flow leading to choking.

Consequences of Surge

- There is no steady operating point once surge occurs. Thus, it is impossible to achieve design pressure rise and mass flow and consequently the thrust of the engine.
- Transient consequences such as inlet overpressure can be severe.
- Surge may damage blades due to vibrational stresses)aero-elastic flutter and divergence(.
- Internal component damage due to hot air passing upstream through the cold sections of
- the engine. Other consequences of surge include power loss and engine flame out

Surging

If the flow rate drops to a point 6, to left of the peak point 2, on the positive (upward) Slope the delivery pressure P03 will continue to decrease causing further drop in mass flow rate and a further drop in Po3 and so on until point 1 is reached where the minimum mass flow is reached. The mass flow may even become negative (i.e. flow reversal) through the compressor. When the back pressure po has reduced sufficiently, due to reduced flow rate the positive flow becomes established once again and compressor picks up until a restricted mass flow is reached, when the pressure fails once again. The pressure therefore surges back and forth in an unstable fashion which, if severe enough. could lead to the mechanical or material failure of parts of compressor. Surging tends to originate in diffuser passages where frictional effects of the vane surfaces on the fluid retard the flow. The likelihood of surging may be reduced by making the number of diffuser vanes an odd number multiple of the impeller vanes.

Chocking

If the mass flow increases in the compressor to the right of point 3 on the negative slope of the characteristic a point 5 is reached where no further increase in mass flow is possible no matter how wide open the throttle control is. This indicates that at some point within the compressor sonic conditions have been reached, causing the limiting maximum mass flow rate and there will be possibility of shock waves that may form within certain passages. Chocking may take place at the inlet, within the impeller, or later in the vaned diffuser section. In a low pressure rise centrifugal compressor or fan choking corresponds to maximum flow rate that is possible to push through the compressor system for the pressure rise.

Rotating Stall

The important cause of instability and poor performance, which may contribute to surge but can exist in the nominally stable operating range: this is the rotating stall. When there is any non-uniformity in the flow or geometry of the channels between vanes or blades, breakdown in the flow in one channel, say B as shown in figure causes the air to be deflected in such a way that channel C receives fluid at a reduced angle of incidence and channel A at an increased incidence. Channel A then stalls, resulting in a reduction of incidence to channel B enabling the flow in that channel to recover. Thus the stall passes from channel to channel:at the impeller eye it would rotate In a direction opposite to the direction of rotation of the impeller. Rotating stall may lead to aerodynamically induced vibrations resulting in fatigue failures in other parts of the gas turbine.

Consequences of rotating stall are:

- Reduced pressure rise in the stage affected.
- Vibrations: The stall cells rotate at a fraction of the rotor speed. As the blade passes in and out of these stalled regions, vibrations are induced. If the natural frequency of blades coincides with which stall cell crosses a blade, resonance occurs leading to possible fatigue failure.
- Hysteresis: Throttling is used to correct rotating stall. However, to recover from stall, the corrected mass flow)or flow coefficient(has to be much larger than that at which the compressor would stall. This phenomenon is known as hysteresis

Numerical

1. Air at 1.0 bar and 288 K enters an axial flow compressor with an axial velocity of 150 m/s. There are no inlet guide vanes. The rotor stage has a tip diameter of 60 cm and a hub diameter of 50 cm and rotates at 100 rps. The air enters the rotor and leaves the stator in the axial direction with no change in velocity or radius. The air is turned through 30.2 DEG as it passes through the rotor. Assume an overall pressure ratio of 6 and a stage pressure ratio of 1.2.

Find

a) the mass flow rate of air,

b) the power required to drive the compressor,

c) the degree of reaction at the mean diameter,

d) the number of compressor stages required if the isentropic efficiency is 0.85

$$U = \pi \times \left(\frac{d_t + d_h}{2}\right) \times N = \pi \times \left(\frac{0.6 + 0.5}{2}\right) \times 100 = 172.76 m/s$$

$$\beta_1 = \tan^{-1}\left(\frac{U}{C_a}\right) = 49.2^\circ$$

$$\beta_2 = 49.2 - 30.2 = 19^\circ$$

$$\tan\alpha_2 = \left(\frac{U - C_a \tan\beta_2}{C_a}\right) = 80.75$$

$$\alpha_2 = 38.92^\circ$$

$$\dot{m} = \frac{\pi}{4} \times \left(d_t^{\,2} - d_h^{\,2}\right) \times C_a \times \rho_2 \qquad \& \qquad T_1 = T_{01} - \frac{C_a^2}{2C_p} = 276.8K$$

$$T_{02} = T_{01} \times \left(\frac{P_{02}}{P_{01}}\right)^{\frac{\gamma-1}{\gamma}} \therefore \quad T_{02} = 303.41K$$

$$T_2 = 303.41 - \frac{C_2^{\,2}}{2C_p} \& \quad \cos\alpha_2 = \frac{C_a}{C_2}$$

$$\therefore C_2 = \frac{C_a}{\cos\alpha_2} = \frac{150}{\cos 38.92} = 192.79m/s$$

$$T_2 = 303.41 - \frac{192.79^2}{2010} = 284.91K$$

$$P_2 = 0.963bar$$

$$\rho_2 = \frac{0.963 \times 101325}{287 \times 284.9} = 1.193Kg/m^3$$

$$\dot{m} = 15.46Kg/s$$

$$P = U \times C_a \times \dot{m} \times (\tan\beta_1 - \tan\beta_2)$$
$$= 172.76 \times 150 \times 15.46 \times (\tan 49.2 - \tan 19) = 326KW$$

$$R_X = \frac{C_a}{2U} \times (\tan\beta_1 + \tan\beta_2)$$
$$= \frac{150}{2 \times 172.76} \times (\tan 49.2 + \tan 19) = 0.65$$

$$\Delta T_{0s} = \frac{U \times C_a}{C_p} \times (\tan\beta_1 - \tan\beta_2)$$
$$= \frac{172.76 \times 150}{1005} \times (\tan 49.2 - \tan 19) = 20.99K$$

2. An axial flow compressor is to be designed to generate a total pressure ratio of 4.0 with an overall isentropic efficiency of 0.85. The inlet and outlet blade angles of the rotor blades are 45 degree & 10 degree, respectively and the compressor stage has a degree of reaction of 50 percent. If the blade speed is 220 m/s and the work done factor is 0.86, find the number of stages required. Is it likely that the compressor will suffer from shock losses? The ambient air static temperature is 290 K and the air enters the compressor through guide vanes.

Axial velocity, $C_a = \dfrac{U}{\tan\beta_1 + \tan\beta_2} = 187 m/s$

Absolute velocity at inlet, $C_1 = \dfrac{C_a}{\cos\alpha_1} = 190 m/s$

The per stage temperature rise,

$$\Delta T_{0s} = \frac{\lambda \times U \times C_a \times (\tan\beta_1 - \tan\beta_2)}{C_p} = 29K$$

Total temperature at compressor inlet,

$$T_{01} = T_1 + \frac{C_1^2}{2Cp} = 331.8K$$

Isentropic total temperature at compressor exit,

$$T_{02s} = T_{01} \times \pi_c^{\frac{\gamma-1}{\gamma}} = 493.9\,K$$

Actual total temperature at compressor exit, $T_{02} = T_{01} + \dfrac{(T_{02s} - T_{01})}{\eta_c} = 522.5\,K$

Therefore total temperature rise across the compressor = $T_{02} - T_{01} = 190.74K$

The number of stages required=

$$\frac{\text{Overall temperature rise across the compressor}}{\text{Per stage temperature rise}}$$

$$= \frac{190.74}{29} = 6.6 \approx 7$$

To determine whether the compressor will suffer from shock losses, we need to find the relative Mach number

$$M_{rel} = \frac{V_1}{\sqrt{\gamma R T_1}}$$

$$V_1 = \frac{C_a}{\cos \beta_1} = 264.5 m/s$$

$$\therefore M_{rel} = 0.77$$

Since relative Mach number is less than unity, the compressor is not likely to suffer from shock losses.

3.The conditions of air at the entry of an axial compressor stage are P1=1 bar and T1=314 K. The air angles are β1=51 deg, β2=9 deg, α1= α3=7 deg. The mean diameter and peripheral speed are 50 cm and 100 m/s respectively. Given that the work done factor is 0.95, stage efficiency is 0.88, mechanical efficiency is 0.92 and the mass flow rate is 25 kg/s,

Determine

a) air angle at stator entry,

b) blade height at entry and hub–tip diameter ratio,

c) Stage loading coefficient,

d) Power required to drive the stage

$$\frac{U}{C_a} = \tan \alpha_1 + \tan \beta_1$$

$$\frac{100}{C_a} = \tan 7 + \tan 51 \qquad \therefore C_a = 73.65 m/s$$

$$\tan \alpha_2 + \tan \beta_2 = \frac{U}{C_a}$$

$$\tan \alpha_2 + \tan 9 = \frac{100}{73.65} \qquad \therefore \alpha_2 = 50.18^\circ$$

$$\dot{m} = \rho_1 \times C_a \times (\pi \times d_m \times h), \quad and \quad \rho_1 = P_1 / RT_1$$

Substituting known values in the above,

$$h = 0.19\,m$$

$$d_t = 50 + 19 = 69cm,$$

$$d_h = 50 - 19 = 31cm$$

$$\text{The hub - tip ratio is } \frac{d_h}{d_t} = 0.449$$

$$\Psi = \frac{w}{U^2} \;\&\; \quad w = \lambda \times C_a \times U \times (\tan\beta_1 - \tan\beta_2)$$

$$w = 0.95 \times 100 \times 73.65 \times (\tan 51 - \tan 9) = 7534.8J / Kg$$

$$\Psi = \frac{7534.8}{100^2} = 0.7535, \text{ is the loading coefficient.}$$

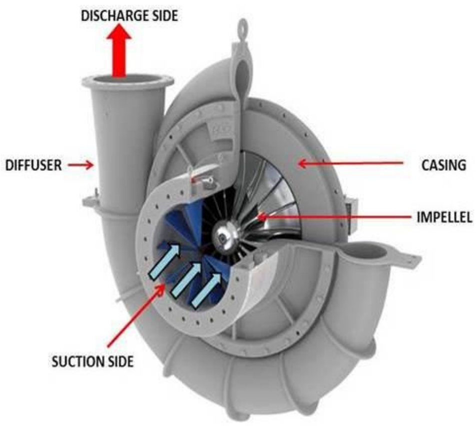
d)

$$P = \frac{\dot{m} \times w}{\eta_m} = 204.75KW \text{ is the power required.}$$

III. Centrifugal flow Compressor

Centrifugal compressor is called so because the flow through the compressor is turned perpendicular to the axis of rotation. This type of compressors is composed of three main elements, namely:

1. Rotating part or impeller
2. Stationary part or stator
3. Manifold or a collector



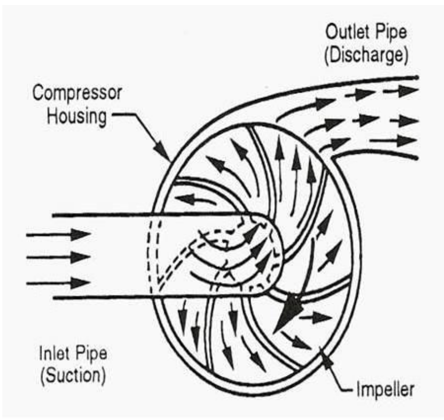


Fig 3: Centrifugal flow Compressor

Air is drawn at the centre or eye of the impeller, then accelerated through the fast spinning speed of the impeller and finally thrown out at tip. At the eye (inlet), the vanes are curved to induce the flow: this axial portion is called the inducer or rotating guide vanes and may be integral with or separated from the main impeller. The number of vanes is usually a prime number, typically from 19 to 37The outer curve of the vanes is sealed by the shroud, which may be part of the stationary structure or may rotate with the rotor. Typical impeller proportions are that the eye root diameter is about half the eye tip diameter, and the tip (outlet) diameter is nearly twice the eye tip diameter. The impeller material is often aluminium, with titanium or steel for smaller, high-duty machines.

Work input Factor: The power input factor represents an increase in the work input, the whole of which is absorbed in overcoming frictional loss and therefore degraded into thermal energy. The fact that the outlet temperature is raised by this loss, and incidentally enables the maximum cycle temperature to be reached without burning so much fuel, so that as far as the efficiency of the whole gas turbine unit is concerned these losses are not entirely wasteful. This effect is outweighed by the fact that more turbine work is used in driving the compressor and isentropic. It follows that the power input factor should be as low as possible, a low value of implying simultaneously a high value of η_c.

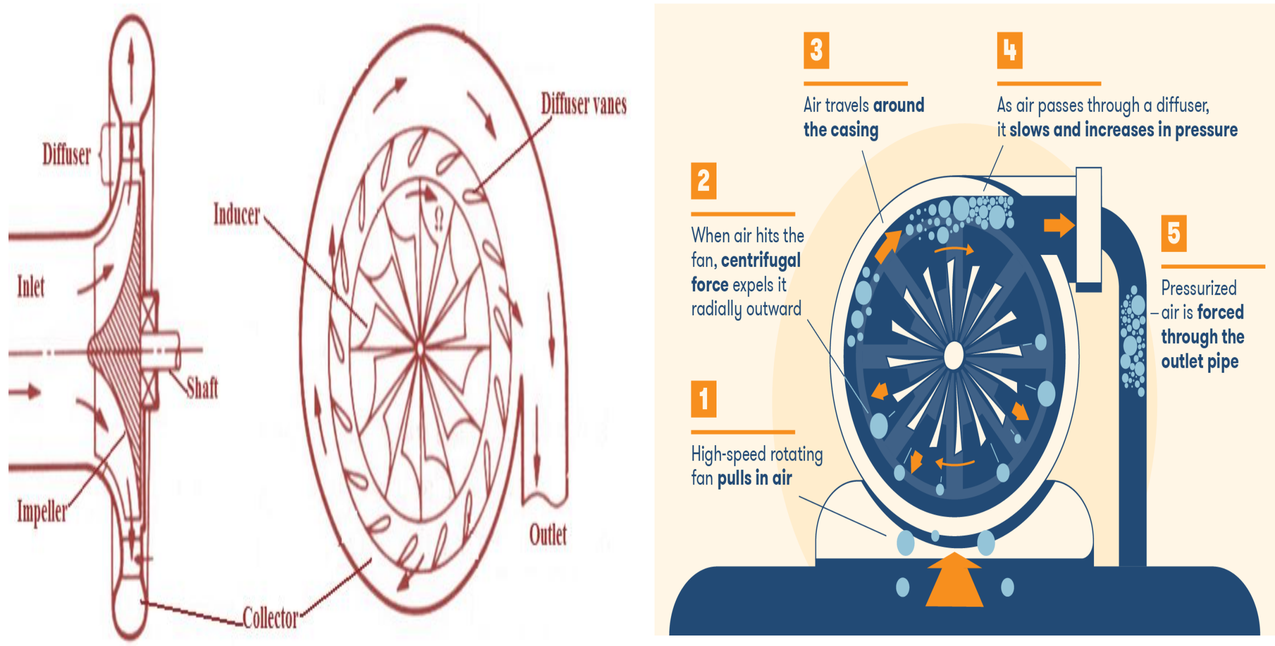


Fig 4: Centrifugal flow Compressor

Diffuser:

The problem of designing an efficient combustion system is eased if the velocity of the air entering the combustion chamber is as low as possible. The velocity of the air at the compressor outlet is in the region of 90 m/s. If the divergence is too rapid, this may result in the formation of eddies. A small angle of divergence, however, implies a long diffuser and a high value of skin friction loss.

Experiments have shown that the optimum included angle of divergence is about 7 degrees.In order to control the flow of air effectively and carry out the diffusion process in as short a length as possible, the air leaving the impeller is divided into a number of separate streams by fixed diffuser vanes. Usually the passages formed by the vanes are of constant depth.

The angle of the diffuser vanes at the leading edge must be designed to suit the direction of the absolute velocity of the air at the radius of the leading edges, so that the air will flow smoothly over the vanes. As there is always a radial gap between the impeller tip and the leading edges of the vanes, this direction will not be that with which the air leaves the impeller tip. For a given pressure and temperature at the leading edge of the vanes, the mass flow passed will depend upon the total throat area of the diffuser passages. Once the number of vanes and the depth of passage have been decided upon, the throat width can be calculated to suit the mass flow required under given conditions of temperature and pressure.

- The number of diffuser vanes is appreciably less than the number of impeller vanes.
- The length of the diffuser passages will of course be determined by the maximum angle of divergence permissible, and the amount of diffusion required,
- Although up to the throat the vanes must be curved to suit the changing direction of flow, after the throat the air flow is fully controlled and the walls of the passage may be straight.

After leaving the diffuser vanes, the air may be passed into a volute (or scroll) and thence to a single combustion chamber this would be done only in an industrial gas turbine unit: in some small industrial units the diffuser vanes are omitted and the volute alone is used. For aircraft gas turbines, where volume and frontal area are important, the individual streams of air may be retained, each diffuser passage being connected to a separate combustion chamber.

Effect of Impeller Blade Shape on Performance

The various blade shapes utilized in the impellers of centrifugal compressors can be classified as

(j) forward curved blades (β> 900),

(n) radial-curved blades (β= 90°),

(iii) backward-curved blades β<90°).

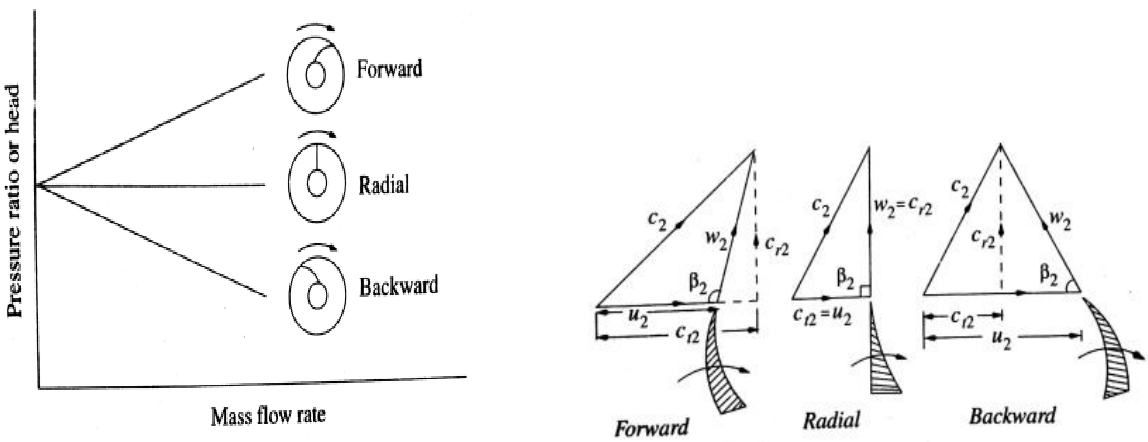


Fig 5: Effect of Impeller Blade Shape

The following represents the variation of pressure-ratio that can be obtained with respect to mass flow rate for the above mentioned various for operation at a given rpm.Centrifugal effects of the curved blades create a bending moment and produce increased stresses which reduce the maximum speed at which the impeller can be run. Good performance can be obtained with radial impeller blades. Backward-curved blades are slightly better in efficiency and are stable over a wider range of flows than either radial or forward curved blades. The forward-curved impeller can produce the highest pressure ratio for a given blade tip speed; hut is inherently less stable and has a narrow operating range. Its efficiencies are lower than that are possible with the backward-curved or radial-curved blades

Prewhirl

The tangential component of velocity at the inlet to the impeller is usually zero as the flow enters the impeller axially. If prewhirl)or inlet guide(vanes are installed to the inlet duct before the impeller, then the incoming air has a tangential component of velocity. In aero engines, positive prewhirl is frequently used to reduce the inlet relative speed.

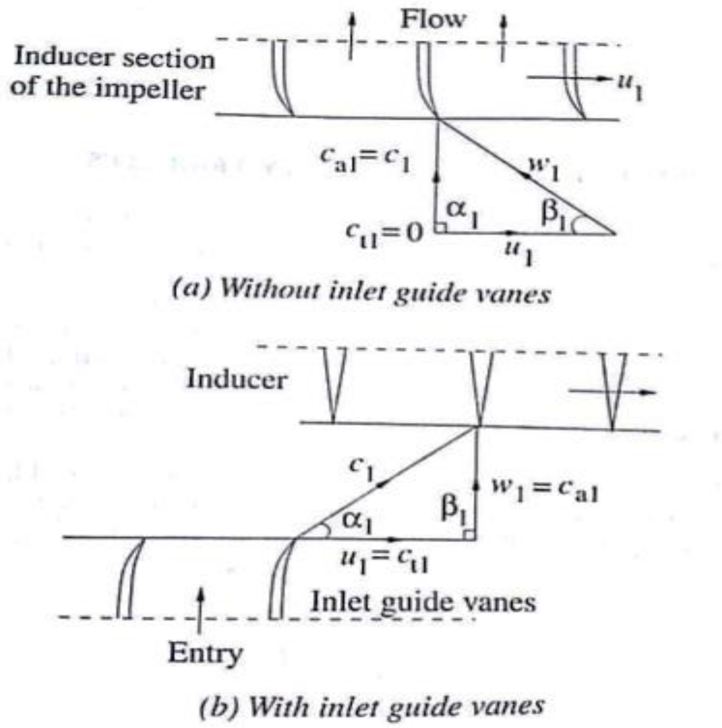


Fig 6: Inlet Guide Vanes in Centrifugal compressor

The objective here is to avoid the formation of shock waves on the blade suction side. The designer of compressor seeks for a small inlet area of the engine to reduce the drag force. At the same time, the air mass flow rate is chosen as maximum as possible to maximize the thrust force. Both factors led to an increase in the axial absolute velocity at inlet, and this in turn increases the relative velocity. Since the relative velocity is maximum at the tip radius of the inlet than when accelerated, there is always a tendency for the air to break away from the convex face of the curved part of the impeller vane. Here a shock wave might occur, which upon interaction with the boundary layer.

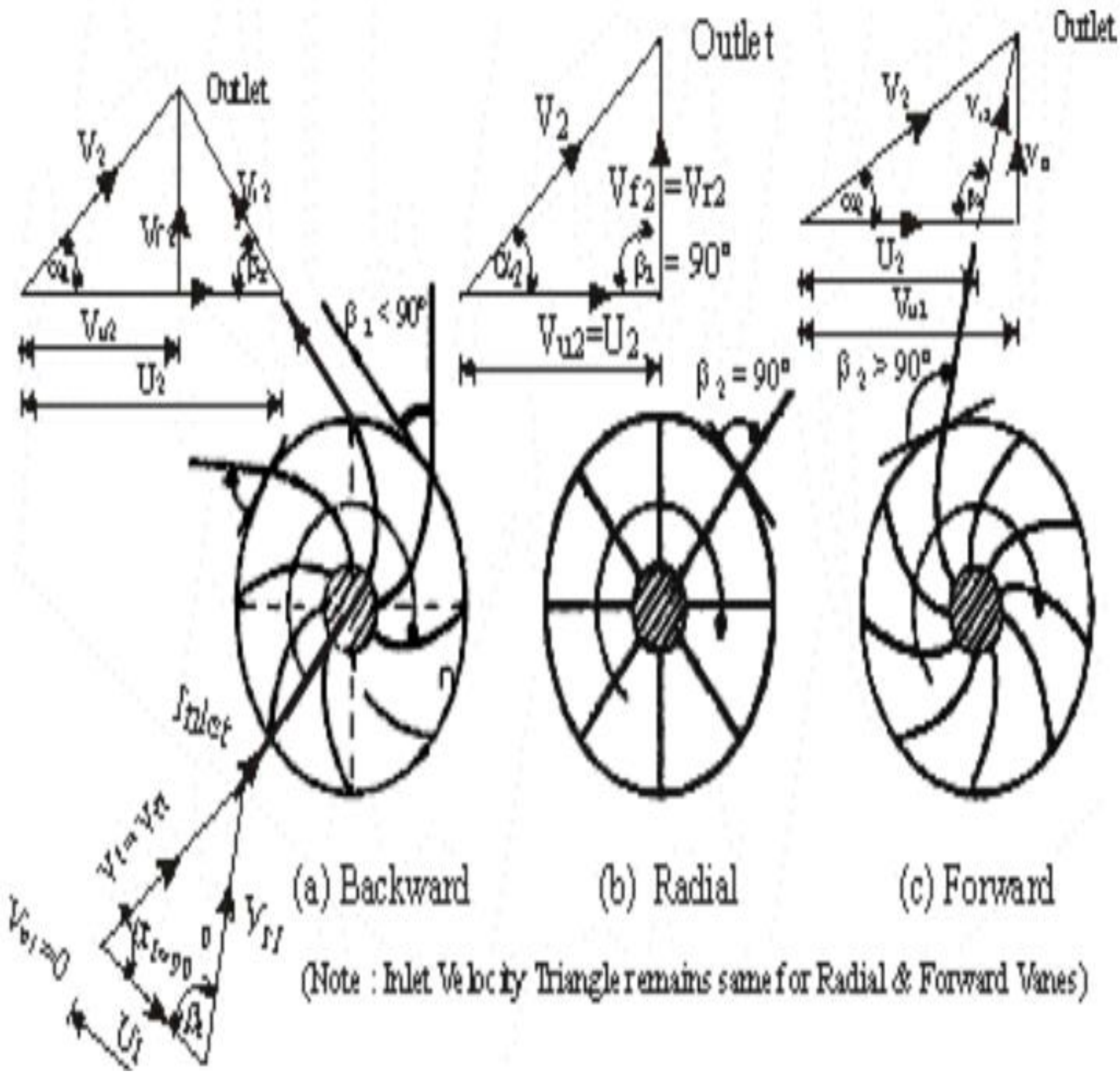


Fig 7: Velocity triangle for Centrifugal compressor

The convex surface of blades, causes a large increase in boundary layer thickness. Thus, excessive pressure loss occurs. The value of the inlet relative Mach number must be in the range from 0.8 to 0.85 to avoid the shock wave losses described previously, or where T1 is the static inlet temperature. Though this Mach number may be satisfactory on ground operation, it may be too high at altitude as the ambient temperature decreases with altitude. For this reason, IGVs are added to decrease the Mach number. These IGVs are attached to the compressor casing to provide a positive prewhirl that decreases the magnitude of the maximum relative velocity)W1(at the eye tip.

The slip factor : The factor limiting the work capacity of the compressor even under isentropic conditions, and this quantity should be as great as possible. An increase in the number of vanes, which would increase absolute velocity decrease in the effective flow area. Additional friction losses arise because, for the same mass flow the inlet velocity must be increased. The thermal energy produced by friction resulting in an increase slip factor in and reduction in efficiency.

Advantages of Centrifugal-Flow Compressor

1. Higher stage pressure ratio (5:1 or even 10:1), while the maximum value for research axial stage is only 2.4:1
2. Simplicity and ruggedness of construction
3. Shorter length for the same overall pressure ratio
4. Generally, less severe stall characteristics
5. Less drop in performance with the adherence of dust to blades
6. Cheaper to manufacture for equal pressure ratio

SCHOOL OF MECHANICAL ENGINEERING

DEPARTMENT OF AERONAUTICAL ENGINEERING

UNIT – V- TURBINES – SAE1302

I. Turbine

A turbine converts the potential and kinetic energy of a moving fluid (liquid or gas) to mechanical energy. In a turbine generator, a moving fluid—such as water, steam, combustion gases, or air which pushes a series of blades mounted on a shaft, which rotates the shaft connected to a generator. A turbine from the Greek meaning "vortex", related to the Latin turbo, meaning a vortex, is a rotary mechanical device that extracts energy from a fluid flow and converts it into useful work.

Turbines can be classified on the basis of following factors:

1. **Based on type of working fluid** — On the basis of the type of working fluid used in a turbine, it could be classified as follows:

- **Steam turbine**: which runs on steam.
- **Gas turbine:** which runs on gas or combustion products of the fuel.
- **Air turbine:** which runs on air.
- **Hydraulic turbine:** which runs on water.

Steam turbines can be further classified as depends upon the exhaust of the steam

- Condensing type: If the exhaust steam is sent to a condenser, then it is called condensing type steam
- Non-condensing type: if the discharge is made to the atmosphere, then it is called non-condensing type steam turbine

2. **Direction of fluid flow** — The direction of fluid flow while expanding in turbine can also be the basis for broadly classifying turbines as follows:

- **Axial now turbines:** have the fluid flowing in the axial direction during the expansion.
- **Radial flow turbines:** have the fluid entering the turbine in radial direction and leaving axially.

3. **Fluid action**: Turbines can also be classified on the basis of the fluid action as

- Impulse turbine
- Reaction turbine
- Impulse-reaction turbine.

The impulse turbine: An impulse stage is characterized by the expansion of the gas which occurs only in the stator nozzles. The rotor blades act as directional vanes to deflect the direction of the flow. Further, they convert the kinetic energy of the gas into work by changing the momentum of the gas more or less at constant-pressure

Impulse turbines can be further sub classified on the basis of compounding or Staging as

- Pressure compounded turbine.
- Velocity compounded turbine.
- Pressure and velocity compounded turbine.

Reaction turbine: A reaction stage is one in which expansion of the gas takes place both in the stator and in the rotor. The function of the stator is the same as that in the impulse stage, but the function in the rotor is two fold.

(i) the rotor converts the kinetic energy of the gas into work, and

(ii) contributes a reaction force on the rotor blades.

The reaction force is due to the increase in the velocity of the gas relative to the blades. This results from the expansion of the gas during its pass through the rotor.

.Reaction turbines can be further sub-classified on the basis of the degree of reaction.

The degree of reaction quantifies the fraction of enthalpy change in the rotor with respect to the total enthalpy change in the stage. Reaction turbines have non-symmetrical blading while impulse turbines have symmetrical blades.

Velocity triangles of a single stage machine

The flow geometry at the entry and exit of a turbo machine stage is described by the velocity triangles at these stations. The velocity triangles for a turbo machine contain the following three components

(i) the peripheral velocity, (u), of the rotor blades,

(ii) the absolute velocity, (c), of the fluid, and

(iii) the relative velocity, (w), of the fluid.

These velocities are related by the following well-known vector equation:

$$\vec{c} = \vec{u} + \vec{w}$$

The notation used here to draw velocity triangles correspond to the x-y coordinates; the suffix (a) identifies components in the axial direction and the suffix (t) refers to the tangential direction. Air angles in the absolute system are denoted by alpha (a), where as those in the relative system are represented by beta (β).

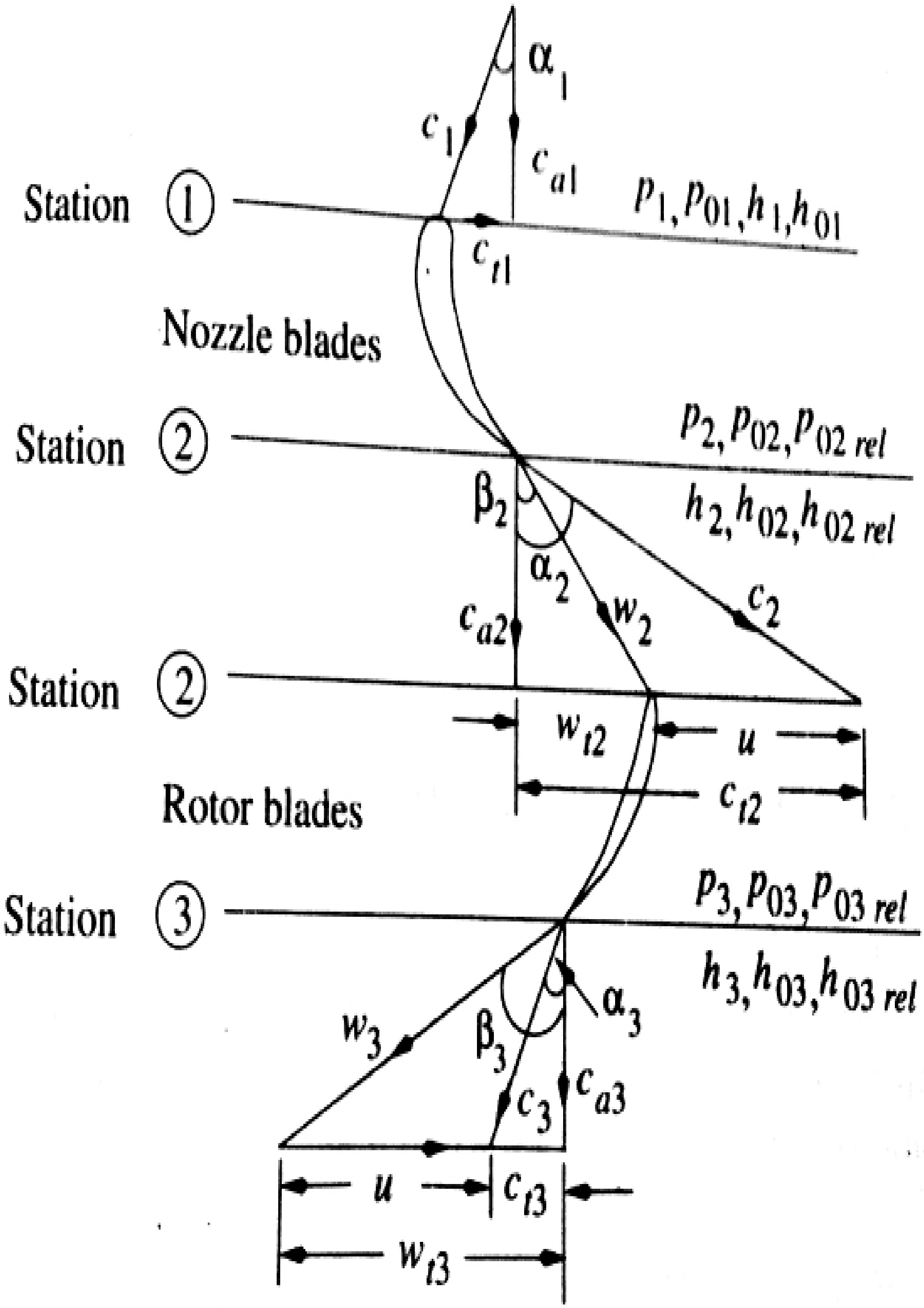


Fig 1: Velocity triangle for Axial flow turbine

From the velocity triangles

$$C_{a2} = C_2 \cos\alpha_2 = W_2 \cos\beta_2 \quad — (1)$$

$$C_{t2} = C_2 \sin\alpha_2 = C_{a2} \tan\alpha_2 \quad - (2)$$

$$W_{t2} = W_2 \sin\beta_2 = C_{a2} \tan\beta_2 \quad — (3)$$

$$u = C_{t2} - W_{t2} \quad — \quad (4)$$

Similarly

$$C_{a3} = C_3 \cos\alpha_3 = W_3 \cos\beta_3 \quad — (5)$$

$$C_{t3} = C_3 \sin\alpha_3 = C_{a3} \tan\alpha_3 \quad — (6)$$

$$W_{t3} = W_3 \sin\beta_3 = C_{a3} \tan\beta_3 \quad - (7)$$

$$u = W_{t3} - C_{t3} \quad — (8)$$

From (4) & (8) $\Longrightarrow$

$$C_{t2} - W_{t2} = W_{t3} - C_{t3}$$

$$C_{t2} + C_{t3} = W_{t2} + W_{t3}$$

useing eq^n 2, 3, 6, 7

$$C_{a2} \tan\alpha_2 + C_{a3} \tan\alpha_3 = C_{a2} \tan\beta_2 + C_{a3} \tan\beta_3$$

But = $C_a = C_{a1} = C_{a2} = C_{a3}$

$$C_a[\tan\alpha_2 + \tan\alpha_3] = [\tan\beta_2 + \tan\beta_3]\, C_a$$

work done by the turbine

$$W = u(C_{t2} + C_{t3}) = C_a u(\tan\alpha_2 + \tan\alpha_3)$$

Degree of Reaction:

A degree of Reaction for axial compressors can also be defined a ratio of actual change of the enthalpy in the rotor to actual change of enthalpy in the stage

$$R = \frac{h_2 - h_3}{h_{01} - h_{03}}$$

$$R = \frac{h_2 - h_3}{h_1 - h_3}$$

$$R = \frac{w_3^2 - w_2^2}{2u(C_{t2} + C_{t3})}$$

From velocity triangles useing Pythagoras theorom.

$$w_3^2 = w_t^2 + C_a^2 \qquad w_2^2 = w_t^2 + C_a^2$$

$$R = \frac{w_{t3}^2 - w_{t2}^2}{2u(C_{t2} + C_{t3})}$$

$$R = \frac{(w_{t3} + w_{t2})(w_{t3} - w_{t2})}{2u(C_{t2} + C_{t3})}$$

$$\because C_{t2} + C_{t3} = w_{t2} + w_{t3}$$

$$R = \frac{(w_{t3} + w_{t2})}{2u} \quad \text{---} \quad (1)$$

From velocity triangle

$$\tan\beta_3 = \frac{w_{t3}}{C_{a3}} \quad \text{---} \quad (2) \qquad \tan\beta_2 \ \frac{w_{t2}}{C_{a2}} \quad \text{---} \quad (3)$$

$$\tan\alpha_3 = \frac{C_{t2}}{C_{a3}} \quad \text{---} \quad (3) \qquad \tan\alpha_2 = \frac{C_{t2}}{C_{a2}} \quad \text{---} \quad (5)$$

Equation 2 & 3 in 1 $\Rightarrow$

$$R = \frac{C_{a3} \tan\beta_3 - C_{a2} \tan\beta_2}{2u}$$

$\because$ $C_a = C_{a1} = C_{a2} = C_{a3}$

$$\boxed{R = \frac{C_a}{2u}\left[\tan\beta_3 - \tan\beta_2\right]} \quad \text{— (6)}$$

Equation 2 & 5 in 1 $\Rightarrow$.

$$R = \frac{C_{a3}\tan\beta_3 - \left[C_{a2}\tan\alpha_2 - u\right]}{2u}$$

$$= \frac{C_a\left[\tan\beta_3 - \tan\alpha_2\right] + u}{2u}$$

$$\boxed{R = \frac{1}{2} + \frac{C_a}{2u}\left[\tan\beta_3 - \tan\alpha_2\right]}$$

Equation 4 & 5 in 1 $\Longrightarrow$

$$R = \frac{C_{a3}\tan\alpha_3 + u - \left[C_{a2}\tan\alpha_2 - u\right]}{2u}$$

$$R = \frac{C_a\left[\tan\alpha_3 - \tan\alpha_2\right] + 2u}{2u}$$

$$\boxed{R = 1 + \frac{C_a}{2u}\left[\tan\alpha_3 - \tan\alpha_2\right]}.$$

Impulse stage: The stage of turbine, having impulse blading, has no expansion in the rotor, that is, no pressure drop. There is no relevance of the degree of reaction for impulse stage.

Zero reaction stage: Mathematically, zero reaction stage refers to R = O condition and it is the condition similar to the case having no pressure drop in the rotor. The combined velocity diagram shows that it is skewed towards left. For R = O.

$$0 = \phi \frac{(\tan\beta_3 - \tan\beta_2)}{2} \Rightarrow \tan\beta_3 = \tan\beta_2 \Rightarrow \beta_3 = \beta_2$$

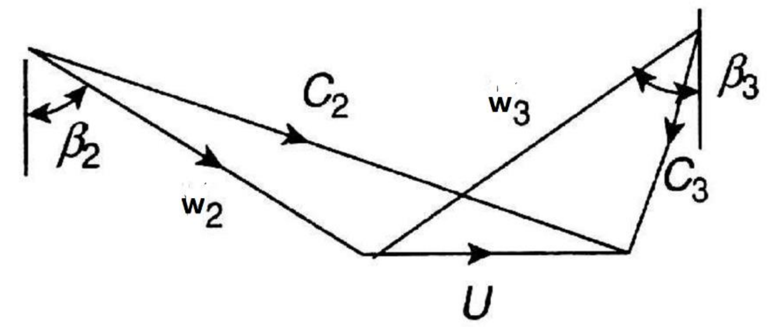


Fig 2: Zero reaction stage

50% reaction stage: Mathematically 50% reaction stage refers to the case whcn R = 0.5 which means this same enthalpy drops occur in the rotor and stator. This offers symmetrical velocity diagram

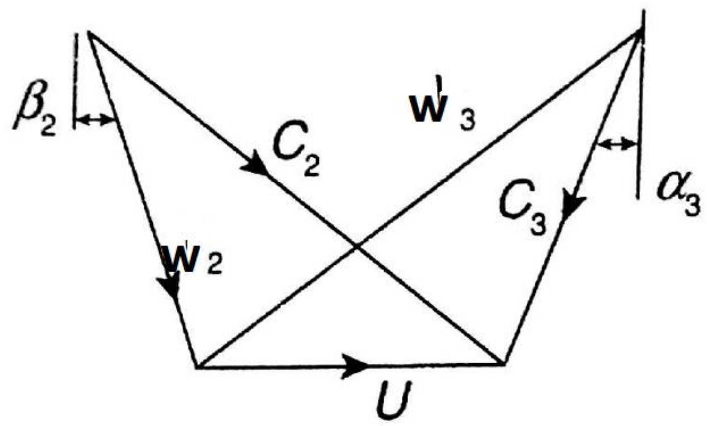


Fig 3: 50% reaction stage

100% reaction stage: Mathematically, 100% reaction stage refers to the case when R =1 which means that there is no enthalpy drop in the stator and total enthalpy drop occurs in the rotor only. The combined velocity diagram is seen to be skewed to the right as shown ¡n Fig. 8. In this case, α2 = α 3 and C-, = C3 as evident from the velocity diagram.

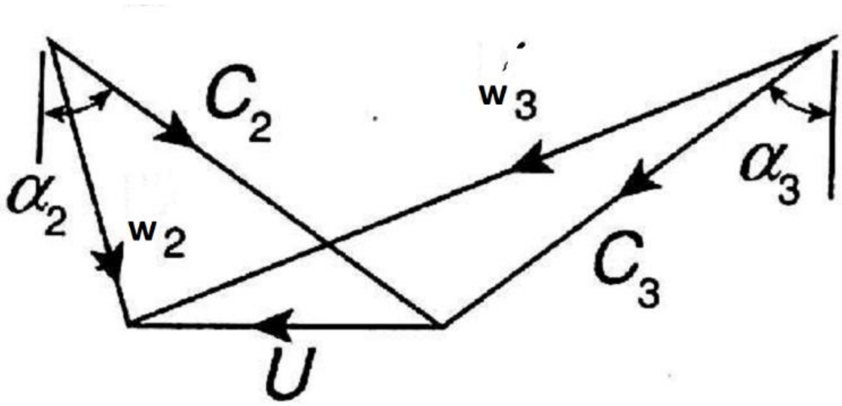


Fig 4: 100% reaction stage

Negative reaction stage: This may happen when the flow is irreversible and there occurs an increase in enthalpy in the rotor. It refers to the rise in the pressure in the rotor due to diffusion observed on the relative velocity This will be accompanied by the adverse pressure gradient leading to flow separation. In this case $w3 < w2$.

Turbine Cooling Techniques

Convection: Convection cooling with a single internal passage was the only available cooling technique in the 1960s. Cooling air was injected through the airfoil attachment and to the inside of airfoil. The cold air was discharged at the blade tip. Development has led to multipass internal cooling. The blades are either cast, using cores to from the cooling passages, or forged with holes of any size and shape that are produced by electrochemical drilling.

The effectiveness of convection cooling is limited by

- Size of the internal passages
- Quantity of cooling air available

The main disadvantage of early convection technique

- Failure to cool the thin trailing edges of blades
- Failure to cool the leading edge which is subjected to the highest temperature.

Impingement cooling :In this high-intensity form of convection cooling, the cooling air is blasted on the inner surface of the aerofoil by high-velocity air jets, permitting an increased amount of heat to be transferred to the cooling air from the metal surface .This cooling method can be restricted to desired sections of the aerofoil to maintain even temperatures over the entire surface .For instance, the leading edge of a blade needs to be cooled more than the mid-chord section or trailing edge.

Film cooling: It involves the injection of a secondary fluid (cold air) into the boundary layer of the primary fluid (hot gases). This is an effective method to protect the surface from the hot gases as the cooling air acts as an insulating layer to maintain a lower blade material temperature.

Disadvantages:

- Causes turbine losses due to injection into the boundary layer.
- If too much air is used or if ejected at high speeds it could penetrate the boundary layer. If close holes are used, they cause stress concentration.

However, it is more effective than normal convection or impingement methods. The cooling air absorbs energy as it passes inside the blade and through the holes, then it further reduces the blade temperature by reducing the amount of energy transferred from the hot gases to the blade

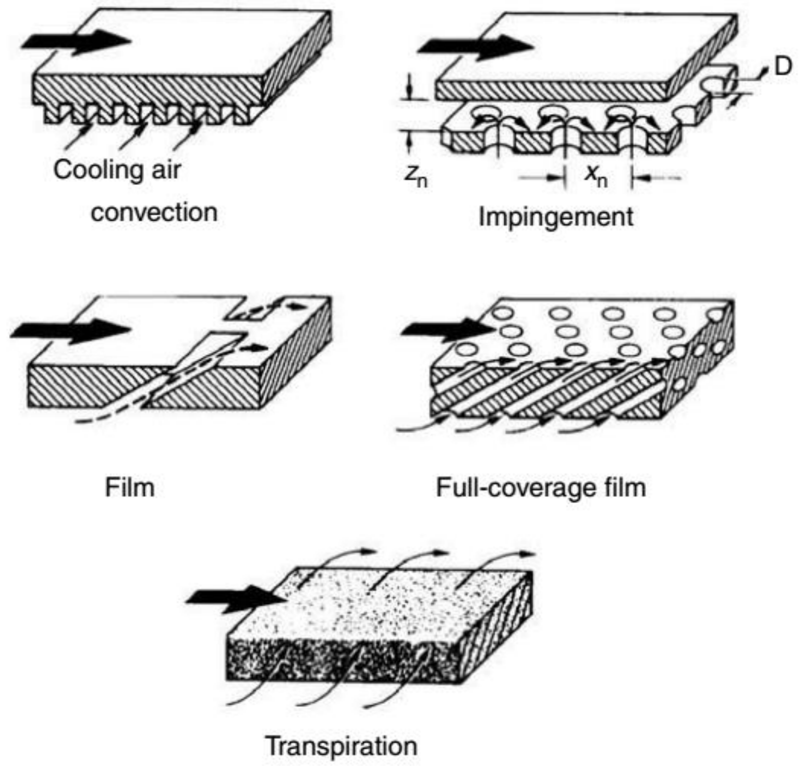


Fig 5 : Turbine Film Cooling

Full-coverage film cooling: It involves the injection of cold air from an array of discrete holes. Thus it represents an attempt to draw on some advantages of transpiration cooling without paying its penalties.

Transpiration cooling: Transpiration cooling of a porous blade wall is the most efficient cooling technique and requires the least cooling air in comparison with the other techniques. It involves the use of a porous material through which the cooling air is forced into boundary layer to form a relatively cooling, insulating film or layer.

Disadvantages

For efficient transpiration cooling, the pores should be small, which leads to problems of blockage due to oxidation and foreign contaminate.

Disadvantages of Cooling Turbine Blades

- Added cost of producing turbine blades
- Turbine blade reliability
- Loss of turbine work due to the cooling air bypassing one or two of the turbine stages
- Loss due to the cooling air being mixed with hot gas steam

Heat Transfer in Turbine Cooling:

The hot as from the combustion chamber strikes the leading edge of the turbine blade which is the stagnation point. The flow comes to a halt. When the flow comes to a halt, thermally that place the total temperature is equal to the static temperature. The blade feels the entire total temperature of the gas flow; that means a static temperature plus the kinetic head that the gas is carrying.

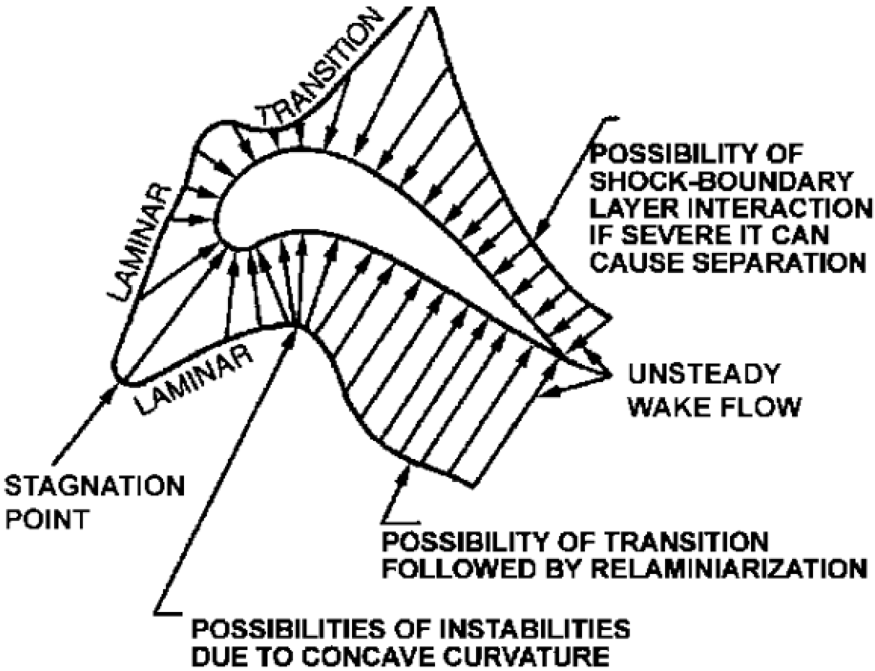


Fig 6: Heat transfer in Turbine Film Cooling

The entire thing is felt over here at the stagnation point. So, the stagnation point is indeed the Hot Spot, the hottest spot on the blade surface. As the blade picks up speed, it actually drops of the temperature, and then, through the transition from laminar to a turbulent flow as the, it flows over the gas blade surface, the gas actually accelerates. As it accelerates, the local temperature, the static temperature reduces

Losses in Turbines:

Tip Clearance Loss)*Y* **k**(
Tip clearance loss occurs in the rotors. Some fluid leaks in the gap between the blade tip and the shroud, and therefore contributes little or no expansion work.

Secondary Flow Loss:
Secondary flows are contrarotating vortices that occur due to curvature of the passage and boundary layers. Secondary flows tend to scrub both the end wall and blade boundary layers and redistribute low momentum fluid through the passage Secondary flow losses represent the major source of losses.Both annulus loss and secondary flow loss cannot be separated and they are accounted for by a secondary loss coefficient

Profile Loss)Y **p(**
The profile loss is the loss due to skin friction on the area of the blade surface**.** It depends on several factors including the area of blade in contact with fluid, the surface finish, and the Reynolds and Mach numbers of the flow through the passage**.**

Annulus Loss
Annulus losses are similar to profile losses as both are caused by friction**.** However, a fresh boundary layer grows from the leading edge of blade whereas the annulus boundary layer may have its origin some way upstream of the leading edge depending on the details of the annulus itself**.**

Radial Turbine:

A radial flow turbine refers to a turbine in which the fluid enters the turbine with high tangential velocity inwards and comes out of the turbine rotor with small whirl velocity at a smaller diameter close to the axis of rotor. In appearance, a radial flow inward turbine looks like a centrifugal compressor with a ring of nozzle vanes instead of diffuser vanes

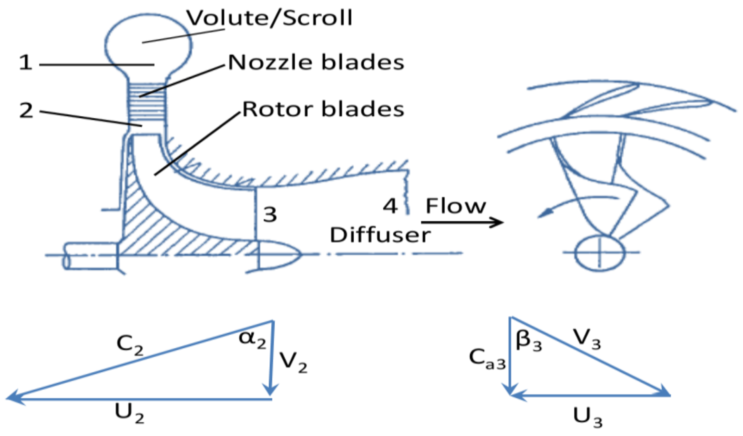


Fig 7: Radial Turbine

Generally, radial flow turbines have blades perpendicular to the tangent at the outer circumference of the rotor inlet. These turbines run at high speed ranging from 40000 to 180000 rpm . The flow accelerated through the stator to velocity C2 is made to enter the rotor radially with velocity V2. It leaves the rotor with velocity V3 and absolute velocity C3 axially Thus at the tip of the rotor the vanes are usually radial and straight. Immediately thereafter the vanes are given 3-D curvature to guide the flow, in an accelerated manner, to a lower radial station and finally let it out at an angle ß with velocity V3. The rotational speed of the gas has gone down from U2 to U3. The flow now assumes an absolute velocity C3 (which by design is made axial i.e. Ca3) and in many turbines is then axially diffused to a lower exit velocity C4. This is done to recover the kinetic energy which would otherwise be wasted with the exhaust