Applied Thermodynamics
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Module - 04
Gas Turbine Engines
Lecture - 36
Aircraft Propulsion Cycle - II

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Dear learners, greetings from IIT Guwahati. We are back again to this course Applied Thermodynamics, module 4, Gas Turbine Engines. Previously, we have covered 5 lectures on this topic, that is module 4, gas turbine engines and we are discussing in the last class about the aircraft propulsion cycle part I.

Now, we are in the last lecture of this module, that is aircraft lecture part II. Of course, in the aircraft propulsion category, we have covered most of the things, but here we will put same concept, but in a different context.

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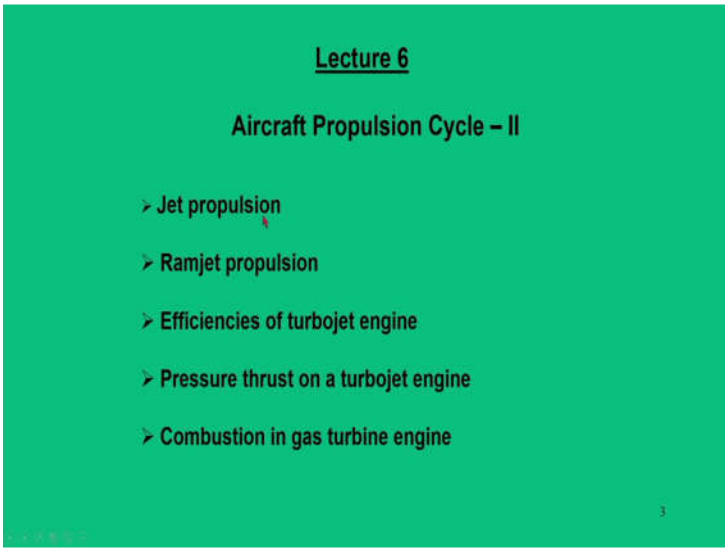

So, the main intention of this particular lecture is based on some concepts like jet propulsion, ramjet propulsion, and we will also discuss about efficiency of turbojet engines. And in fact, we are going to calculate the pressure thrust on a turbojet engines, and to some extent we will cover about the combustion process in a gas turbine engines.

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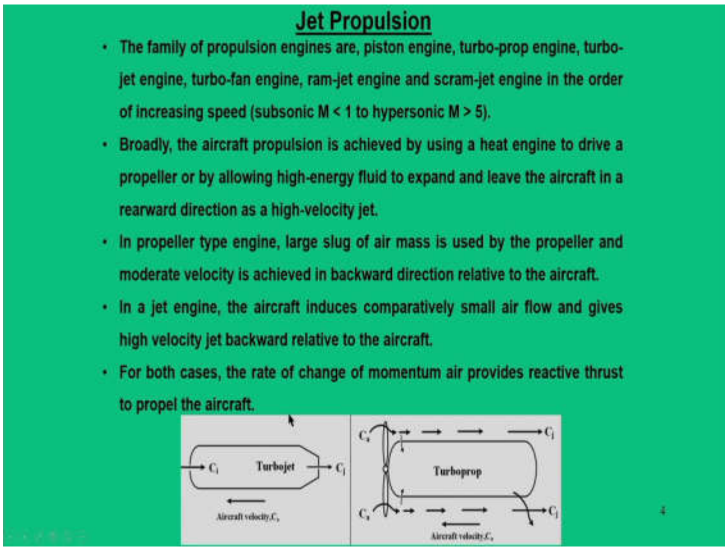

So, previously in the our last lecture, we discussed about the a jet engine and it is divided into two broad categories, one is turbojet engine, other is turboprop engines. So, in a turbojet engine the thrust is developed through a propelling nozzle, we call this as a jet.

And in a turboprop engine majority of the thrust is developed through propeller. And air passes through the propeller and gives the forward thrust, as and when it goes out from the engine.

Now, over the period of time, there are many developments in the propulsion engines, such as we have piston engines, we have turboprop engines, we have turbojet engines, turbofan engines, ramjet engine, scramjet engines. And if you want to travel faster, then we have to change the propulsion engine in this order.

For example, for subsonic aircraft, we can achieve it through a turboprop engines or turbojet engines. But if you want to go for supersonic engines then you have to use ramjet engine, its operated in a different concept. And further, if you want to travel at hypersonic speed, then we have to use this scramjet engines

But in a thermodynamic sense, we call this aircraft propulsion as a device and which is nothing but a heat engine that drives the propeller or by allowing high energy fluid to expand and leave the aircraft in the rearward direction as a high-velocity jet. But this is what happens in a turbojet engines. And in a turboprop engines, there are large slug of masses that enters from the front end that is through the propeller and this gives the necessary thrust and for which the moderate velocity is achieved in the backward directions relative to the aircrafts.

So, in a turbojet engine the aircraft induces comparatively small airflow and gives high velocity jet backward relative to the aircraft. But whatever you call this as a turboprop or turbojet engine, the very fundamental principle that hold good that is the Newton’s law, which means that rate of change of momentum of air provides reactive thrust to propel the aircraft.

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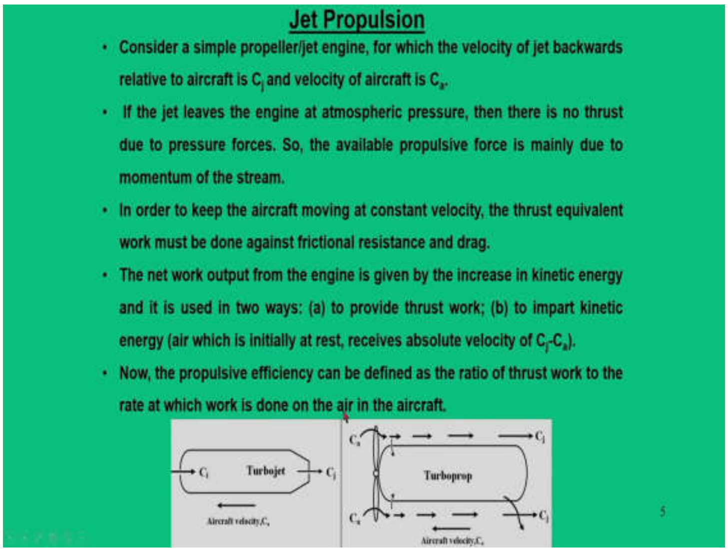

Now, let us do a very simple analysis whether it is a propeller or jet engine for which there is a velocity of the jet which is defined as C_j and the velocity of the aircraft which is C_a. So, you can see here, the C_j is your jet velocity whereas, aircraft velocity is C_a. And similarly, same thing happens here also for turboprop engines.

And when the jet expands it expands to atmospheric pressure, and then there is no thrust due to pressure force, since it has already expanded to atmospheric pressure. But if you do not expand it to atmospheric pressure, then we will have a pressure thrust. So, in order to keep the aircraft moving at a constant velocity, and typically this is what we call as a cruise velocity at high altitude, the thrust equivalent work must be provided to overcome the frictional resistance and drag.

So, basically the net work output from the engine is given by increasing the kinetic energy that is through the jet. And it is used in two ways, first one it provides the thrust work because this kinetic energy provides the thrust work, and also it imparts the kinetic energy to the air which was initially at rest, so that it receives say absolute velocity C_j - C_a. So, initially C_a was 0, but as and when the kinetic energy of the jet is imparted, the relative component of C_a and C_j keeps changing.

So, in a sense, that we can define the propulsive efficiency which we have already defined in our last class by analyzing the entire thing as a propulsive duct. Now, we are going to apply the same concept for a turbojet and turboprop engines. But the concept remains same. So, the propulsive efficiency is defined as the ratio of thrust work to the rate at which work is done on the air in the aircraft.

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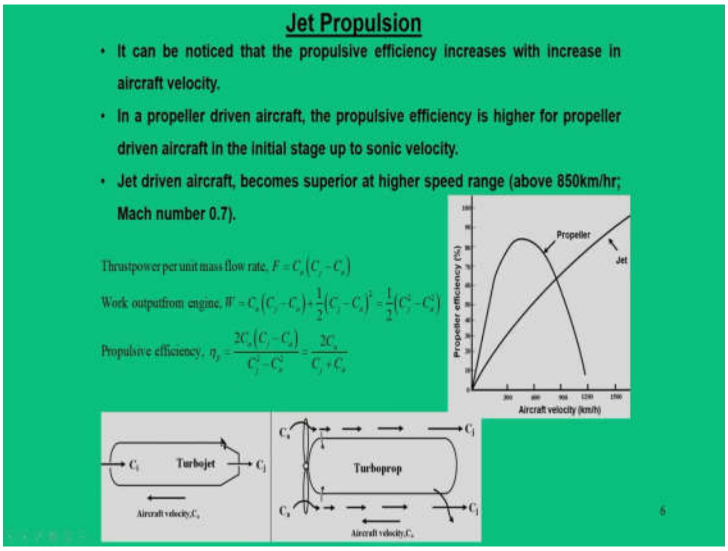

From this, a simple relations can be made that thrust power per unit mass flow rate that is $F = C_a(C_j - C_a)$. So, that work output from the engine becomes W that is nothing but $\frac{1}{2}(C_j^2 - C_a^2)$. So, you can say propulsive efficiency this can be defined by $\frac{2C_a}{C_j + C_a}$.

Now, one interesting thing that we can see here that the propulsive efficiency increases with aircraft velocity. So, with increasing the aircraft velocity, the propulsive efficiency increases. But one interesting thing is that if you analyze it for a propeller driven aircraft, which is turboprop or a jet engine aircraft which is turbojet, some interesting features can be seen.

What you can see is that in the beginning of period, that means, when you keep on increasing the aircraft velocity you find that propeller gives a better approach means better propeller efficiency. The propulsion efficiency is high as compared to jet. But what happens? At one particular time maybe about 600 km/h, this propulsive efficiency drops.

So, when the propulsive efficiency drops, in contrast if you say jet engine its propulsive efficient keeps on increasing with the aircraft speed. So, at one particular instant or time, the jet engine takes the advantage of a propeller engines.

So, all the commercial aircraft, when they are supposed to run at lower speed, they prefer to have a propeller driven machine or engine. So, we call as a propeller engine. When they go for a very high velocity at the at high altitude, then turbojet or turbofan engines are preferred.

And this remarkable change we say that jet engines becomes more efficient, when the Mach number is about 0.7 or when you expect the flight velocity to be more than 850 km/h.

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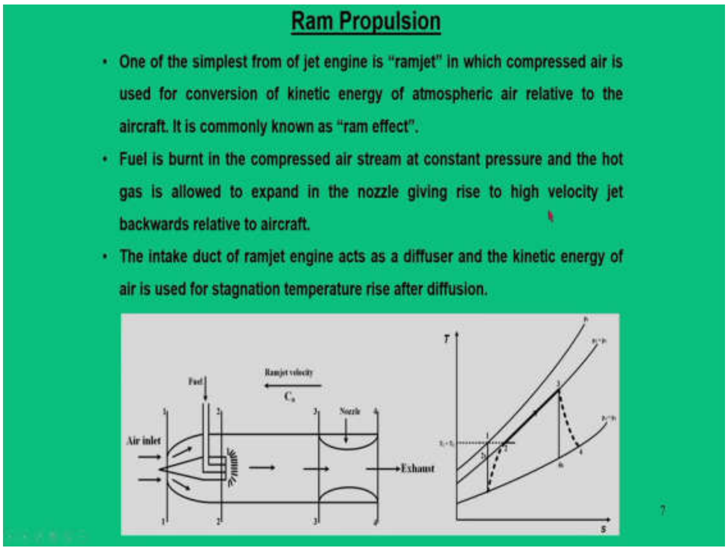

Another concept that drops in from this jet is what we call as a ramjet engine. So, it is simply a ramjet engine, in which the compressed air is used for conversion of kinetic energy of atmospheric air relative to the aircraft. And we call this as a ram effect.

So, what happens you can say in ramjet engines, the air inlet that comes from the station 1, and finally, it goes at station 2. So, through this process, the inlet act acts as a intake and it acts as a diffuser, that means, the flow velocity decreases. So, the intake of ramjet engine acts as a diffuser, and the kinetic energy of the air is used for the stagnation temperature rise after this diffusion process.

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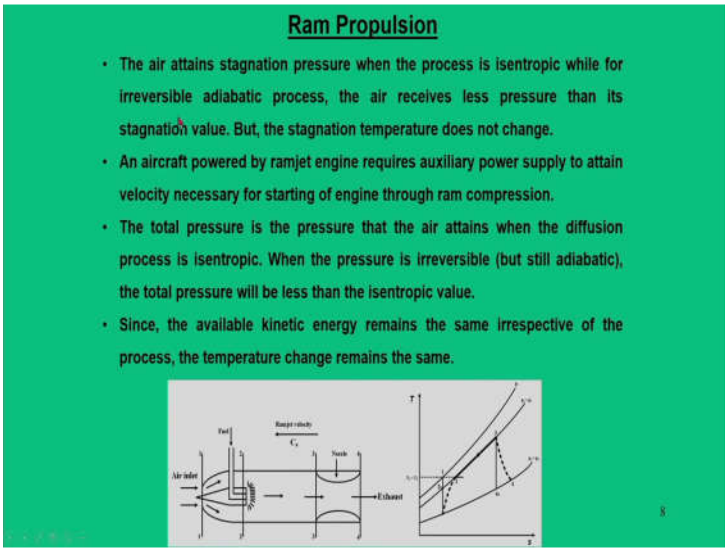

Other concept that are used here that air attains the stagnation pressure when the process is isentropic and in fact, this duct is typically an adiabatic duct, where the air receives less pressure than the stagnation value. But whatever the case may be, the stagnation temperature does not change, that is 1 and 2, the stagnation temperature does not change.

The other concept of ram propulsion is that when the aircraft is powered by a ramjet engine it requires auxiliary power supply to attain this velocity which is necessary for starting of the engine through ram compression. The idea of ram propulsion drops in when you are supposed to attain supersonic speed. But in order to reach this supersonic speed we cannot go in a single instance.

So, what we have to do? First thing we have to go to certain altitude where you have to cruise at the supersonic velocity. But to go to that altitude, we require some other means of propulsion system which you call as auxiliary power supply. So, it could be conventional turbojet engine or turboprop engines, and finally, we have to fire this ramjet engines for which the propulsion is achieved through this high velocity duct.

So, means for the starting of the engine we require additional power unit to attain the air velocity that is at the inlet. The other thermodynamic aspects are the total pressure is achieved by the air when the diffusion process is isentropic. And if it is irreversible the total pressure will be less than the isentropic value. And of course, the available kinetic energy remains the same irrespective of the process because the temperature remains the same.

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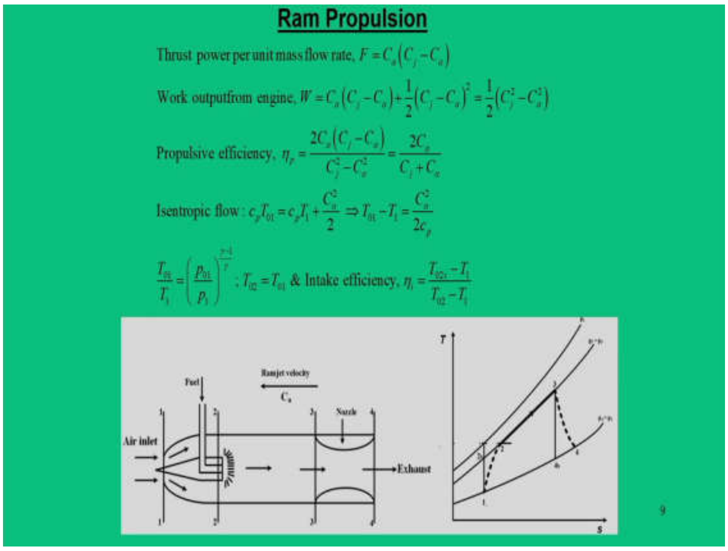

So, a brief calculations in this aspect we can have here is that the thrust power per unit mass flow rate which you can calculate $F = C_a\left(C_j - C_a\right)$. So, from this, we can get also work output from the engine that is $\frac{1}{2}\left(C_j^2 - C_a^2\right)$. So, based on this, we define this propulsive efficiency.

Now, considering this flow to be isentropic flow, and looking at the T-S diagram, we can write that is $c_pT_{01} = c_pT_1 + \frac{C_a^2}{2}$. So, from this we will get a temperature difference, total temperature difference at station 1 and that is nothing but $T_{01} - T_1 = \frac{C_a^2}{2c_p}$.

So, based on this, the process is isentropic, other relations between stagnation pressure and static pressure will hold good. And of course, we said that the total temperature $T_{02} = T_{01}$ because there is no extra heat added into this system.

And based on this, since it acts as intake, we can write the intake efficiency through the temperature drop that is $\eta_i = \dfrac{T_{02s} - T_1}{T_{02} - T_1}$. So, this is all about the ram propulsion when you want to travel faster.

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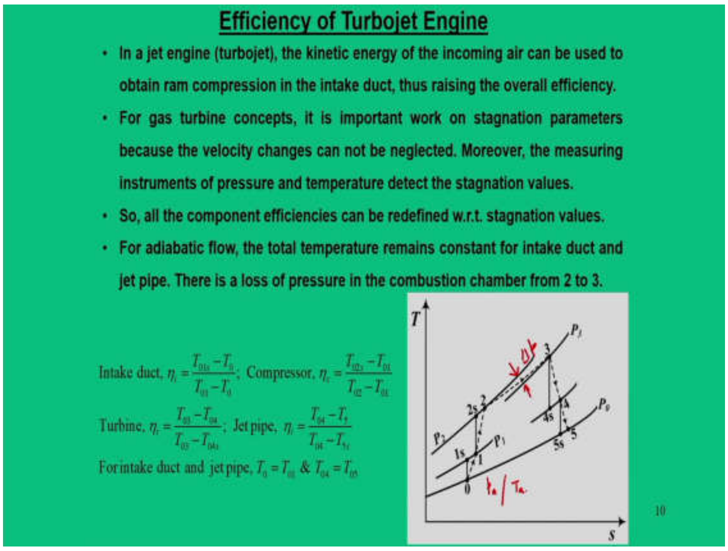

Apart from this, let me discuss some other thermodynamic aspects that we typically define for isentropic efficiency and with respect to a turbojet engine. So, in earlier situations, when we define this as efficiency, we normally do not encounter or do not take into account about the kinetic energy or potential energy change in our temperature or pressure calculations.

But when we are going for aircraft engines and that too turbojet engines, it is not advisable to use those static value, rather we prefer to use the stagnation values because implicitly the kinetic energy changes are already taken into account while calculating the stagnation temperatures or pressures.

Now, let us see this typical T-s diagram in which we have plotted different pressure changes that happens. And what has been plotted is that when you start the process maybe at 0, here 0 stands for p_0 that is nothing but your ambient p_a or T_a.

So, as if we are travelling at altitude maybe 110 kilometer above the earth surface and at that point whatever ambient pressure and temperature conditions, the static value and stagnation values remain same as air is stagnant. And from this 0 to 1 process is your intake process, and through this intake process, had the process been isentropic, it would have gone from 0 to 1s, but now it goes to 1.

Now, from 1 to 2, it is a compression process, and if it is isentropic compression then it would have gone to from 1 to 2s. So, these are the static P_1 and P_2 value, and corresponding stagnation values can be also calculated. Now, the process 2 to 3 is in a heat exchanger process or in our case we call it as a combustion chamber.

So, in a combustion chamber, some kind of pressure drop is always ensured. So, this pressure drop is taken into account as delta p, that is in the combustion chamber, because of which whatever pressure that enters into the turbine at state 3, it is not same as the compressor outlet pressure.

So, there is a typical drop in this pressure. So, we say that at state 3, air enters to the turbine. And the expansion process takes in two steps. One in the turbine if it is in the turbine and the process is isentropic, then it will go as 3 to 4 s. And if it is a real process, so it will go as a 3 to 4 and from 4 the process that happens is in the nozzle because the expansion takes place in the nozzle to generate the thrust. So, had the process been isentropic, then it would have been 4 to 5s, but actually it should be 4 to 5.

Now, here the typical conditions that when this expansion takes place that point of time, whatever type of nozzle people use, or in the engines they use, typically nozzle becomes choked; that means, that is the optimum flow rate or optimum thrust that can be achieved from the nozzles. So, normally condition 5 and 5s, they are almost above the ambient value, and the flow condition is typically to be choked. And in fact, 5 and atmospheric are not equal. So, all entire calculations are governed through the choking condition of the nozzle.

And when you talk about efficiencies we have 4 efficiencies that comes into picture, one for intake duct, other for compressor, third for turbine, and four for jet pipe or we can simply say propelling nozzle. So, in the similar sense, the efficiencies are defined based on their ratios of actual value and isentropic values.

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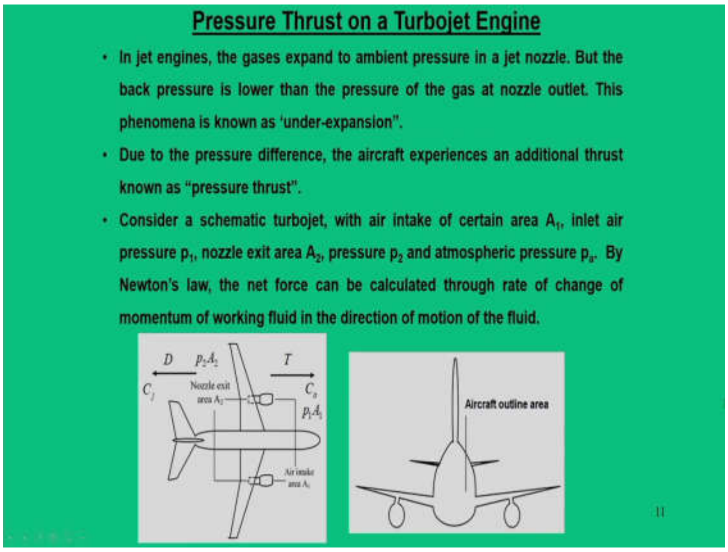

Now, moving further, we are going to discuss on the pressure thrust on a turbojet engines. In fact, in previous lecture, we discussed this pressure thrust and that concept was typically different in a sense that there we defined 3 pressure ratios, one is nozzle pressure ratio, one is ram pressure ratio, and we found out a non-dimensional thrust coefficients as a function of nozzle pressure ratio.

So, as a layman, if you want to go for a change in the thrust coefficient, we can simply change the nozzle pressure ratio. But that was a different context in which pressure thrust was calculated. But here we will use some similar concept, but our approach will be something different, and let us see that how we are going to analyze here the pressure thrust calculations. And again, we will stick to our simple engine that is turbojet engine.

So, very basic philosophy that gas expands to ambient pressure in a jet nozzle. But the back pressure is always lower than the pressure at the nozzle outlet. So, this phenomena is called as under-expansions. That means, it could have expanded, but your pressure difference does not allow.

And due to this pressure difference, in addition to momentum thrust, the aircraft also experiences additional thrust which is known as pressure thrust. So, we are going to consider this pressure thrust, and for this we take care about a turbojet engines in which the air intake has a certain area A_1, that is at intake location, and the exit nozzle area that is flow comes out is from A_2 and it is at pressure p_2.

And based on this, we can see jet velocity C_j which is in this direction and your aircraft velocity C_a in the forward directions and when the aircraft is cruising, total thrust that acts is equal to the drag, because they have to balance so that, it will keep the aircraft always in the moving conditions.

And in another view, that is frontal view, actually this particular area you call this as a aircraft outline area. And although the aircraft are of big area enclosure, but for our thrust calculations, it is taken with some outline area and that is decided at which the engine is located.

And from this, we can use the Newton's law which says that net force is nothing but rate of change of momentum of working fluid in the direction of motion of the fluid.

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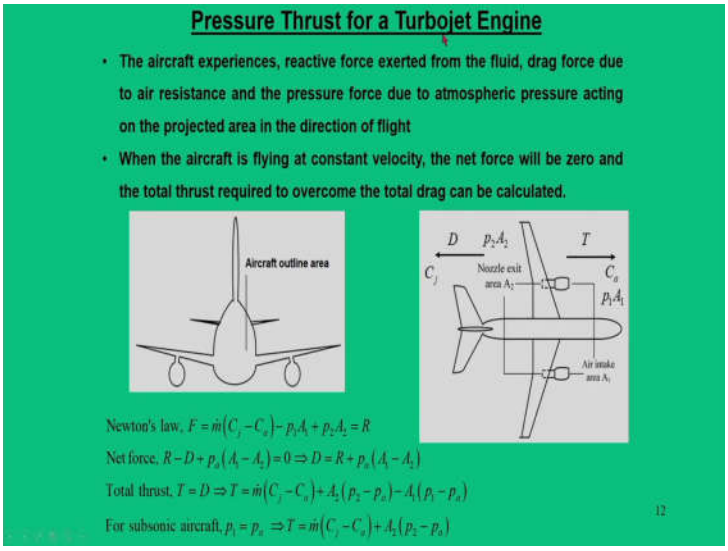

So, let us do this calculations that from the Newton's law, we can say how much force that acts on the aircraft that is through the jet and from the pressure force. And that is nothing but your reaction force. So, the net force becomes reaction force minus drag and plus this pressure with respect to this aircraft outline area; $R - D + p_a(A_1 - A_2)$. So, from this we get an expression for drag which is which is nothing but $D = R + p_a(A_1 - A_2)$.

Now, when thrust equal to drag, so that we can write $T = D \Rightarrow T = \dot{m}(C_j - C_a) + A_2(p_2 - p_a) - A_1(p_1 - p_a)$. So, that by putting D in this equation, we can calculate the total thrust. And if you look at this particular equations, the total thrust composed of 3 parts, one is through this jet. Other is with respect to area A_2, and the pressure difference that is p_2 minus p_a. And another with respect to area A_1, and the pressure difference that is p_1 minus p_a, where p_a is your ambient pressure.

And for a subsonic aircraft, the second term does not appear because there is some different concept drops in that there is no shock waves. So, that this particular turn does not appear for the subsonic aircraft. But for a supersonic aircraft, the entire flow which is outside this medium also gets disturbed. So, because of this region, this particular term remains.

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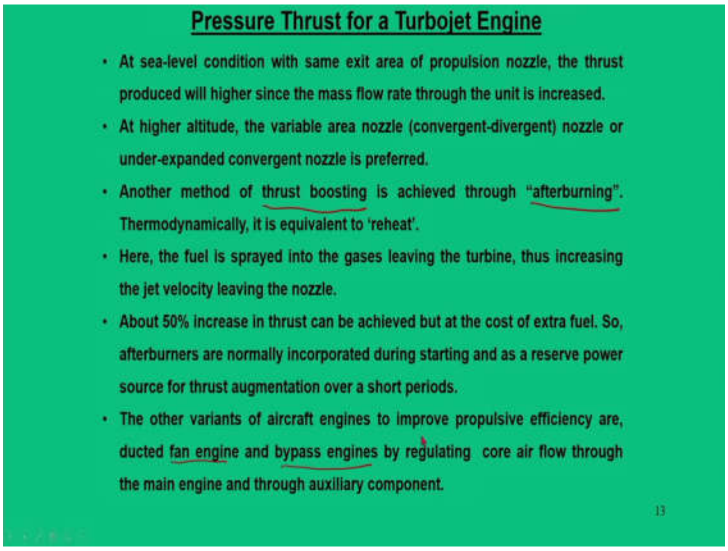

And the other concept that the pressure thrust calculation in the turbojet engines has some salient features. What are they? So, at least sea level conditions with same exit area of the propulsion nozzle, the thrust produced is higher than the mass flow rate through the unit because since the mass flow rate of the unit is increased the thrust produce will be higher.

That means, at sea level conditions your thrust will be higher because the mass flow rate into the unit is increased, because at the sea level conditions your density is high, but at higher altitude your density is less. So, mass flow rate that is entering is also less.

So, at higher altitude, it is always preferred to use a variable type nozzle, and typically this variable area nozzle is nothing but a convergent divergent nozzle or simply a under expanded convergent nozzle. And based on this, when we say that nozzle is choked and all our calculations are normally driven through the choking flow conditions of the nozzle. And in fact, the choking flow calculations is very simple relations.

In many aircrafts, people think about of thrust boosting. So, whatever you calculated that we say momentum thrust or pressure thrust that is based on the flow jet velocity, and the ambient pressure, and what pressure we are operating. So, there is a fixed amount of thrust that is involved whether it is at cruising stage or takeoff stage or landing stage. So, in all 3 cases we can calculate the thrust.

But what happens in some situations we expect to travel faster while cruising. So, in some cases, people prefer the method what we call as a thrust boosting. And this thrust boosting is normally achieved through after burning. After burning is nothing but simply we call this as a reheat. So, when you deal with the reheat cycle of gas turbine, we say that the heat is added separately and to reach the maximum temperature and the gas after the reheating further expanded in another turbine.

But here we do add heat, and instead of calling this as a reheat we call this as after burning. And fuel is burnt separately in another method and we call this as a after burning or an after burner is used or you can say it is a secondary type of combustion chamber. And in the secondary combustion chamber, the fuel is spread into the gases which leaves from the turbine. Thus, it increases the jet velocity.

And through this after burning concept there is a possibility that we can increase the thrust by about 50 percent higher, but at the cost of extra fuel. And in aircraft terminology the cost of extra fuel means it is a very costly affair. So, it may not be economically viable for going for after burning unless and until it is really required.

So, the after burnings are normally incorporated during the starting where we require higher initial thrust or as a reserve power source for thrust augmentations. And in fact, whenever this after burner is used, it is used for a very short period.

And apart from this thrust boosting, people used the other variant of aircraft engines and we call this as a fan engine or bypass engines. So, in fact, when the air enters to the engine, whether turboprop or turbojet engines, air that comes we call this as a primary air. But the other variants; that means, in order to improve this efficiency of this airflow people have used the concept as a ducted fan or bypass engines; that means, out of this total air that enters, some of the air, majority of the air goes as a primary air, other quantity of air we call as a secondary air.

So, through that managing this air flow will be better and through this we can also get a better approach in running this after burner. So, in some cases, the bypass air is directly goes as a after burner. Either it can be used as after burning of the fuel or it can give an additional thrust which can go without burning as well. So, that comes in the separate aspect, and all these things are covered in the aircraft propulsions. And in fact, in our course, we only deal with the thermodynamic aspects of this propulsion system.

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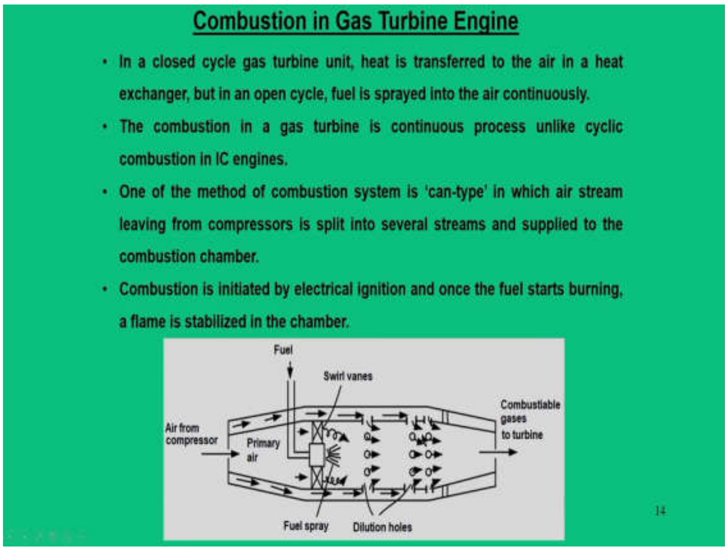

And the last segment of this aircraft propulsion or in fact, in gas turbine engines is combustions. And we have covered all the components, turbine, compressor, intake, nozzle, all these components are covered, but we have not discussed anything on how the combustion process happens in the gas turbine engines.

So, irrespective of the fact that whether you use combustion process in a conventional gas turbine engine or in a jet engine, the more or less the mechanism of combustions remain same. Another point I need to emphasize that when you deal with the gas turbine combustion and if you try to compare IC engine combustions, both are little bit different because IC engines it can operate as a constant pressure mode or constant volume mode whereas, in a gas turbine engine it is always operated at a constant pressure based combustion.

Another difference that we can have here that when you deal with the combustion in IC engines these are cyclic in nature. So, that means, in a continuous cycle combustion takes place only for certain duration of crank rotation in IC engines. But here in the gas turbine engines, it is always a continuous combustion.

Now, when a continuous combustion is used; side by side we also expect that we have sufficient level of turbulence, there should be proper mixing of air fuel. And another important aspect is that here the air velocity is very fast because as and when your speed increases air velocity also increases. And proper mixing is another criteria of importance for gas turbine engine combustions.

So, for that different types of mechanism that has been used like instead of sending the air from the combustion chamber, it has been proposed that air does not go as a single slog, it goes as a multiple slogs at different outlet points from the compressors. So, that is what we call this as a combustion process in the gas turbine engine is different.

So, if you look at the particular figure here, whatever I have discussed if I just explain with respect to this figure, we can say that in the inlet the air comes from the compressors and after the combustion process is over the combustible gases goes to the turbine for expansion.

But in this process, what happens? Now, when air enters to the combustion chamber there are possible ways of entering that air can come through different bypass region, and we call this as a primary air. So, majority of the air comes as a primary air. Other type of air that comes through the circumference holes we call this as a dilution holes. And in this dilution holes also secondary air also comes in.

Now, when the primary air goes in, it encounters the fuel injection process and this fuel injection process is nothing but typically whatever you do in the constant pressure based combustion in CI engine. And here additional devices; that means, fuel goes as a spray, through fuel nozzle, and also, we have swirling wing vanes, so it gives a proper mixing of this primary air with this fuel.

So, the flame front gets generated and it tries to propagate. This is a fuel rich zone because the entire fuel sees this air and as and when the flame goes, it gets additional air through this dilution hole. So, in a process, what happens? By this means the flame gets stabilized, so that we get a sustained flame throughout this combustion chamber.

And when we have a sustained flame; that means, we can ensure that combustion efficiency is very high and means all the fuels gets combusted and finally, towards the end of this combustion chamber the combustible gases goes to the turbine.

So, this particular arrangement we call this as a can-type where the air leaving from the compressor is split into the several streams and it is supplied to the combustion chamber. Of course, initially the combustion is initiated electrically, and when the fuel starts burning the flame is stabilized in the chamber.

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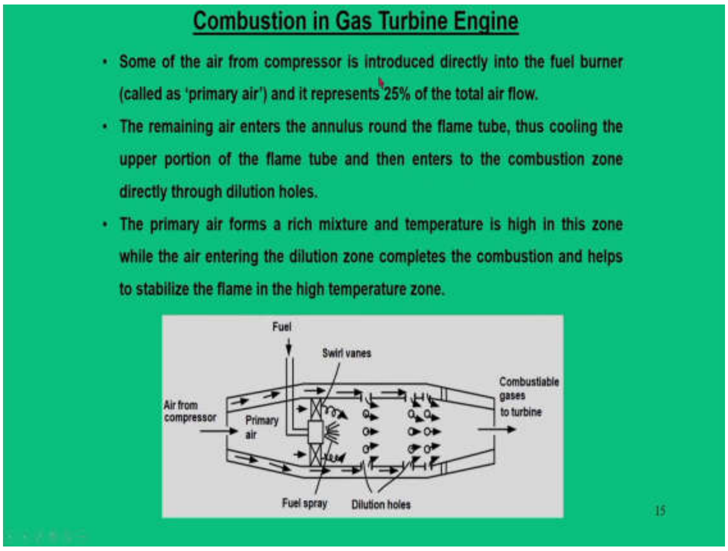

And some of the air from the compressor is introduced directly. So, we call this as a primary air. And it is only about 25 percent of the total air. The remaining air enters to the annulus or through these dilution holes, and in the flame tube. The primary air forms the rich mixtures and temperature is high in this zone. So, while air entering the dilution zone completes the combustion and it helps to stabilize the flame in the high temperature zone.

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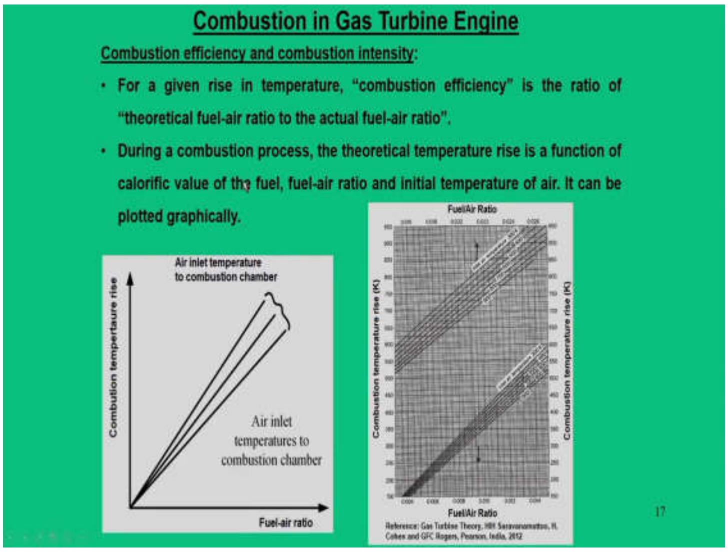

And if you give some typical numbers for this, the air fuel ratio in a gas turbine engines is 60 to 120 whereas, in our IC engines it is 14.5 that is stoichiometric air fuel ratio. Air velocity at the entry of the combustion chamber is limited with 75 m/s because if you keep on increasing the airspeed, we do not have sufficient mixing time. So, to the best available technology till today we say that the air velocity at the entry to the combustion chamber is limited by 75 m/s.

And of course, in a similar context when you deal with rich and weak limits in the IC engines, we call either rich mixture or lean mixture, and it leads to a condition where there are possibilities of knocking. Similar case also happens here in the gas turbine engines, and this particular phenomena we call as a screech.

So, screech is nothing but a situation what we call as a combustion instability and similar terminology that you use in the IC engine is nothing but knock. We always ensure that knock should be avoided in IC engine combustions, similarly we should avoid this screech in the gas turbine combustions.

And in some situations, the gas turbine engines use after burners. So, this is similar to the concept called as reheating. And this reheating is performed between turbine stages or after burning of the fuel for satisfactory usage of additional fuel requirement. That means, all of a sudden, we do not use this fuel directly in the primary combustion chamber, rather fuel path is outlined in such a way that as and when is required, fuel can be supplied to the after burner to satisfy the requirement of additional need.

And of course, there are some pressure losses we have been keep on telling in our analysis, and this pressure loss is mainly due to non-adiabatic flow, turbulence, and frictions. And there is a continuous pressure loss, but at a given cross sections typically the pressure is constant. At a given cross section of the combustion chamber pressure may be treated as almost constant.

And in combustion term the pressure loss which is happens due to friction is called as a cold loss, and the pressure loss that happens due to heating process we call as a fundamental loss. And more or less what you can say pressure loss is inevitable.

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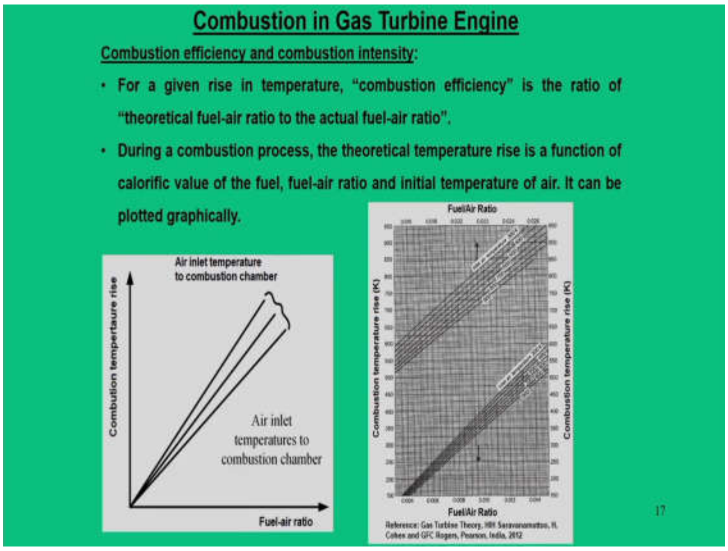

And there are some terms like combustion efficiency and combustion intensity. So, combustion efficiency we have been utilizing many aspects in our problems. It is nothing but for a given temperature rise, the combustion efficiency can be defined as a theoretical air pressure ratio to the actual air pressure ratio. And to calculate this theoretical pressure air ratio, we can plot this combustion pressure rise and fuel-air ratio for given air inlet temperatures. This is how we use this graphical approach.

A concrete picture for gas turbine theory from the book has been shown that a graphical approach is very widely used. In x axis the fuel-air ratio is plotted, and the combustion temperature is plotted in this case on the y axis. And for a given air inlet temperatures these are the lines which are drawn here. And as a user you have to use this particular graph to calculate the fuel-air ratio.

Now, to calculate this fuel-air ratio, you need two parameters, one is the combustion temperature rise, other is the air inlet temperatures. And the combustion temperature rise can be calculated by knowing the temperature or stagnation temperature differences across the combustion chamber or it is nothing but the difference in total temperatures at the turbine inlet and the compressor outlet. So, that is $T_{03} – T_{02}$.

And the air inlet temperature is nothing but your compressor delivery pressure. So, based on this, we can fix this point and the accordingly you can drop a vertical, so it will cut the fuel-air ratio axis. So, when you do this, then you can calculate graphically the fuel-air ratio. This is a common approach which is followed in the gas turbine engines.

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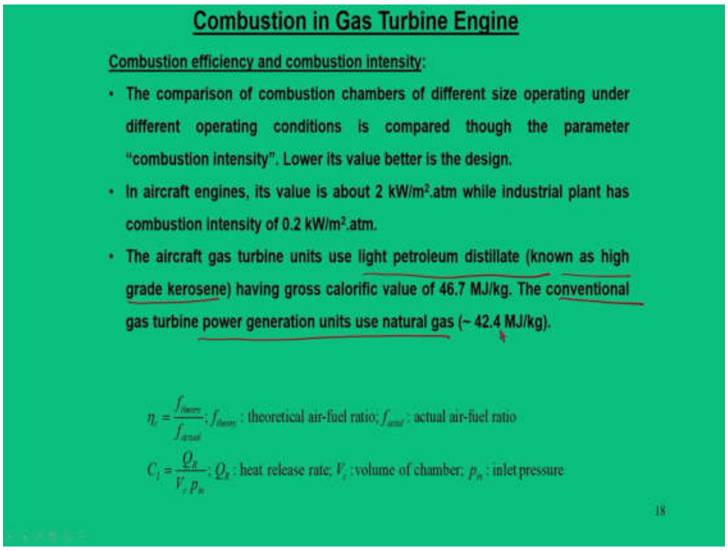

The other term that is also important is nothing but the combustion intensity. Now, if you want to compare two combustion chambers, another way of quantification is that combustion intensity that is $C_I = \frac{Q_R}{V_c\, p_{in}}$; where, Q_R : heat release rate; V_c :volume of chamber; p_{in} : inlet pressure

So, $V_c\, p_{in}$ becomes an additional parameter to compare two combustion chambers. And some realistic numbers of this combustion intensity is about 2 kW/m^2 atmosphere for in for aircraft engines and 0.2 kW/m^2 atmosphere for aircraft industrial plant.

And another last point that I want to mention here, that when you talk about gas turbine engines what are the fuel that are used. So, for aircraft gas turbines, we use light petroleum distillates, and typically they are nothing but the high-grade kerosene. For conventional gas turbine power generation, people use natural gas. And if you look at their gross calorific value, high-grade kerosene has about 46.7 MJ/kg whereas, for natural gas it is about 42.4 MJ/kg.

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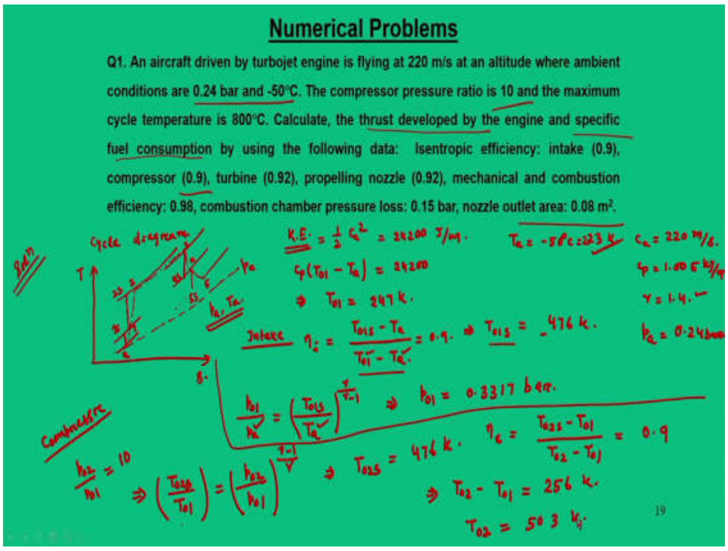

So, with this I conclude this gas turbine aircraft propulsion topic. And lastly, I will touch upon another problem that is again on the turbojet engines. And similar problem was solved in our last lecture, but still I stick to this fact that solving these problems will help a complete understanding of aircraft engines. So, let me start this particular numerical problems.

So, it is a turbojet engines which is flying at 220 m/s, and it is an altitude conditions where it is 0.24 bar and -50°C. In fact, we know the compression ratio, the maximum cycle temperature that is turbine inlet temperatures, and for this type of problems typical analysis or calculation requirement would be calculation of thrust and specific fuel consumptions.

Other additional data you require about isentropic efficiency, that is for intake, compressor, turbine, and propelling nozzle, also we require mechanical and combustion efficiency, combustion pressure loss, and nozzle outlet area. And this nozzle outlet area is required for thrust calculations.

So, let us solve this problem. Now, to solve this problem first thing that you do is that draw the cycle diagram. The cycle diagram can be drawn like this and it is nothing but your T-s diagram. So, in the T-s diagram, we have to draw first intake process, then we have to draw this compression process, then we have to draw the turbine process, and in the turbine, there is a drop in pressure. And finally, this straight line will be isentropic process, dotted line will be actual process.

So, if I know this location, I can put this number as a, 1s, 1. So, 1 to a is intake, 1 to 2 is compression, 2s is isentropic compression, 2 to 3 is the heat addition in the combustion chamber, 3 to 4s is your isentropic expansion, 3 to 4 is actual expansion, 4 to 5s is your isentropic expansion in the nozzle, 4 to 5 is the actual expansion. Of course, you can draw this as constant pressure line which is ambient p_a and T_a. These values are already known as 0.24 bar and -50°C.

$$\text{K.E.} = \frac{1}{2} C_a^{\,2} = 24200 \text{J/kg}$$

$$c_p\left(T_{01} - T_a\right) = 24200 \Rightarrow T_{01} = 247\text{K}$$

$$\text{Intake: } \eta_i = \frac{T_{01s} - T_a}{T_{01} - T_a} = 0.9 \Rightarrow T_{01s} = 476\text{K}$$

$$\frac{p_{01}}{p_a} = \left(\frac{T_{01s}}{T_a}\right)^{\frac{\gamma}{\gamma-1}} \Rightarrow p_{01} = 0.3317\text{bar}$$

$$\text{Compressor:} \frac{p_{02}}{p_{01}} = 10 \Rightarrow \frac{T_{02s}}{T_{01}} = \left(\frac{p_{02}}{p_{01}}\right)^{\frac{\gamma-1}{\gamma}} \Rightarrow T_{02s} = 476\text{K}$$

$$\eta_i = \frac{T_{02s} - T_{01}}{T_{02} - T_{01}} = 0.9 \Rightarrow T_{02} = 503\text{K}$$

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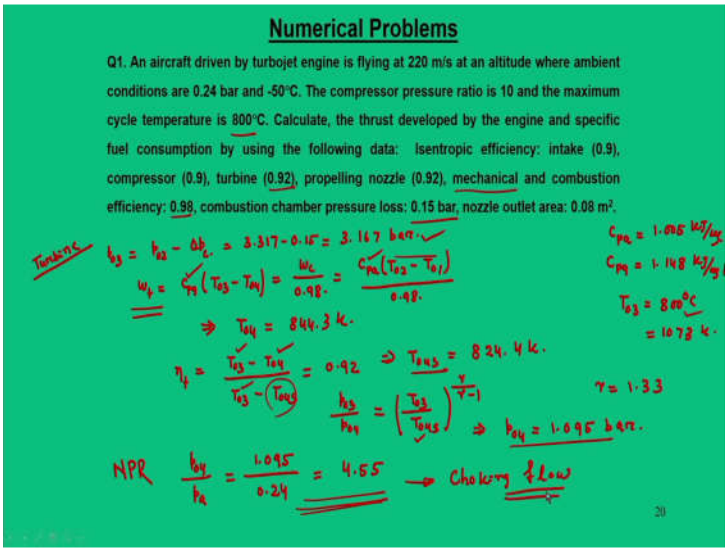

$$p_{03} = p_{02} - \Delta p_c = 3.317 - 0.15 = 3.167\text{bar}$$

$$w_t = \frac{w_c}{\eta_m} \Rightarrow C_{pg}\left(T_{03} - T_{04}\right) = \frac{C_{pa}\left(T_{02} - T_{01}\right)}{\eta_m} \Rightarrow T_{04} = 844.3K$$

$$\eta_t = \frac{T_{03} - T_{04}}{T_{03} - T_{04s}} = 0.92 \Rightarrow T_{04s} = 824.4K; \frac{p_{04}}{p_{03}} = \left(\frac{T'_{04}}{T_{03}}\right)^{\frac{\gamma}{\gamma-1}}, \gamma = 1.33 \Rightarrow p_{04} = 1.095\text{bar}$$

$$\text{NPR} = \frac{p_{04}}{p_a} = 4.55$$

And if you look at this nozzle pressure ratio, this will tell you that we get a choking flow because this pressure ratio is very high.

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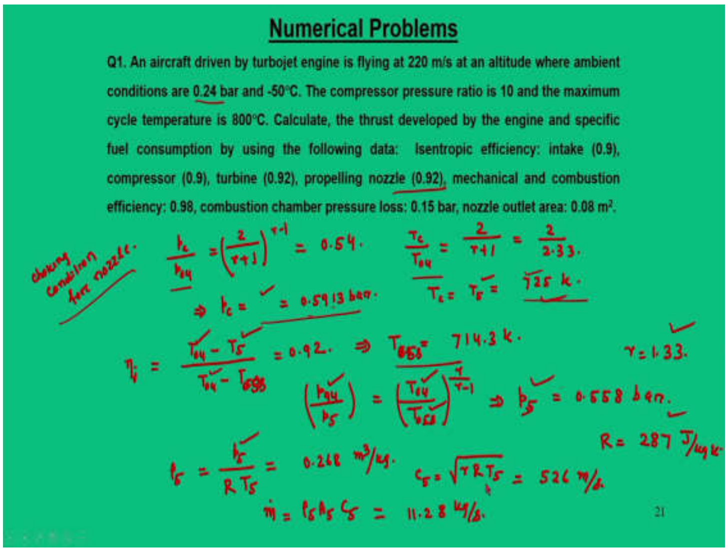

And for this reason again, we have to calculate choking condition for nozzle.

$$\frac{p_c}{p_{04}} = \left(\frac{2}{\gamma+1}\right)^{\gamma-1} = 0.54; \frac{T_c}{T_{04}} = \frac{2}{\gamma+1}$$
$$\Rightarrow p_c = 0.5913\text{bar}; T_c = T_5 = 725\text{K}$$

So, basically condition 5 is now known, and this pressure and temperature is much high. So, whatever assumption you have said that choking condition is nozzle, that is taken care by looking at ambient pressure values and this critical pressure value from the nozzle.

$$\eta_j = \frac{T_{04} - T_{05}}{T_{04} - T_{05s}} = 0.92 \Rightarrow T_{05s} = 714.3K; \frac{p_{04}}{p_5} = \left(\frac{T_{04}}{T_{05s}}\right)^{\frac{\gamma}{\gamma-1}} \Rightarrow p_5 = 0.558\text{bar}$$
$$\rho_5 = \frac{p_5}{RT_5} = 0.268 kg/m^3; \therefore \text{As choked}, M = 1 \Rightarrow C_5 = \sqrt{\gamma RT_5} = 526\text{m/s}$$
$$m_5 = \rho_5 A_5 C_5 = 11.28\text{kg/s}$$

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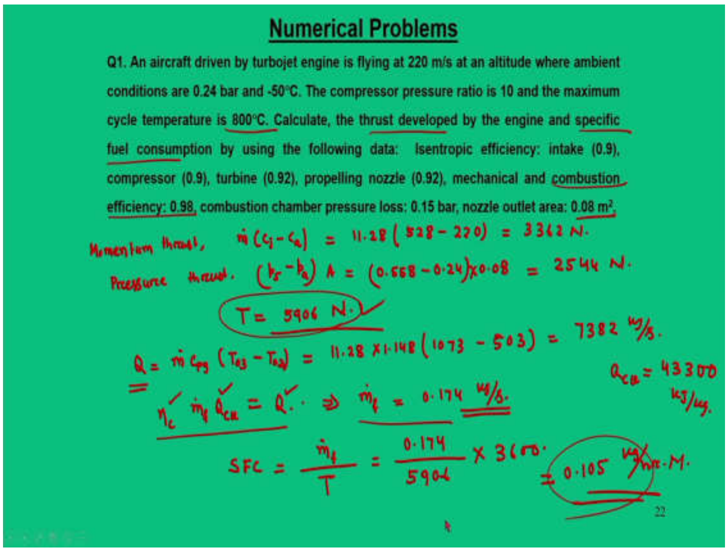

And last segment that we are going to calculate is about specific thrust. So, here there are two parts, one is momentum thrust and pressure thrust.

$$T = \dot{m}\left(C_j - C_a\right) + A\left(p_5 - p_a\right) = 11.28\left(528-220\right) + 0.08\left(0.568-0.24\right) = 2544\text{N}$$

$$Q = \dot{m}c_{pg}\left(T_{03} - T_{02}\right) = 11.28 \times 1.148\left(1073-503\right) = 7382\text{kJ/s}$$

$$\Rightarrow \eta_c \dot{m}_f Q_{CV} = 7382\left(\because Q_{CV} = 43300\text{kJ/kg}\right) \Rightarrow \dot{m}_f = 0.174\text{kg/s}$$

Now, we have got this mass flow rate, then we can calculate Specific Fuel Consumption, $SFC = \frac{\dot{m}_f}{T} = \frac{0.174}{5906} \times 3600 = 0.105\text{kg/hr.N}$. So, after a long process, at the end of the day, we only require this thrust and specific fuel consumptions. And this simple turbojet engines, gives you the entire concept how entire process has to be followed.

With this I conclude the lecture for the today, and module 4 is completed, that is gas turbine is completed. Thank you for your attention. We will see you in the next module that is on Refrigeration and Air Conditioning.

Thank you for your attention once again.