Jet and Rocket Propulsion

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Lecture - 24

Good morning. So, let us continue our course on rocket and space craft propulsion. For the last couple of lectures, we have been discussing the performance of ideal rockets, we have listed the assumptions that are required, and then we have defined certain parameters.

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One of those parameters was the thrust coefficient defined as F T upon A star P c naught, where F T is the thrust produced by the rocket, A star is the throat area for the converging diverging nozzle; here, we are considering the rocket to be like this, a combustion chamber and a converging diverging nozzle. So, this area is A star; the minimum area, the throat area, the combustion chamber conditions are P c naught and T c naught; that is the stagnation pressure and stagnation temperature in the combustion chamber. So, we had defined the thrust coefficient like this. Then, we have proved that the critical mass flow rate or the choke mass flow rate; since the nozzle is going to be choked so this is the mass flow rate, is equal to A star P c naught multiplied by square root of gamma upon square root of R T c naught whole multiplied by 2 upon gamma plus 1 to the power gamma plus 1 upon 2 multiplied by gamma minus 1. So, this is the mass flow rate, as we can see here the mass flow rate is a function of the throat area and the chamber conditions.

Now, in order to increase the mass flow rate since the throat area is constant, only way to increase the mass flow rate is by changing the chamber conditions. If we reduce the combustion chamber temperature, the mass flow rate is going to increase or, if we increase the combustion chamber pressure the mass flow is going to increase. So, the possibility for increasing the mass flow rate, and we know that the mass flow rate will dictate the thrust, total thrust, because thrust is m dot U v plus the pressure term or m dot U equivalent. So, in order to increase the thrust, one way is to increase the mass flow rate.

Now, we have also proved in the last class that the thrust coefficient can be made independent of every other parameter by the pressure ratio and gamma. So, we have proved this in the last class, 1 minus P e upon P c naught to the power gamma minus 1 by gamma to the power half, this is the momentum thrust, and the pressure thrust given as; here P e is the exit pressure at the exit of the nozzle, P a is the ambient pressure, A e is the exit area that is the area here. Therefore, we see that the thrust coefficient is function of gamma, the pressure ratio and the area ratio. However, we have shown before that area ratio is a function of mark number, therefore it is a function of pressure for the isentropic; a special ratio, therefore the area ratio is function of pressure ratio.

So, we can eliminate this area ratio from here, and we can get it completely as a function of pressure ratio, and we had discussed that. So, what we have shown that C F is a function of the pressure ratio and gamma. Now, once the design is fixed, P e is fixed, then P e by P c naught is a function of area ratio, so that becomes a fixed ratio. Then, in order to change this we have to change the pressure, but we have also shown that as we increase the pressure the thrust is going to increase, but we have also shown that it cannot increase to infinity; there is a maximum limit. When this ratio tends to 0, that is the exit pressure is very small compared to the stagnation pressure, we get the ultimate thrust coefficient.

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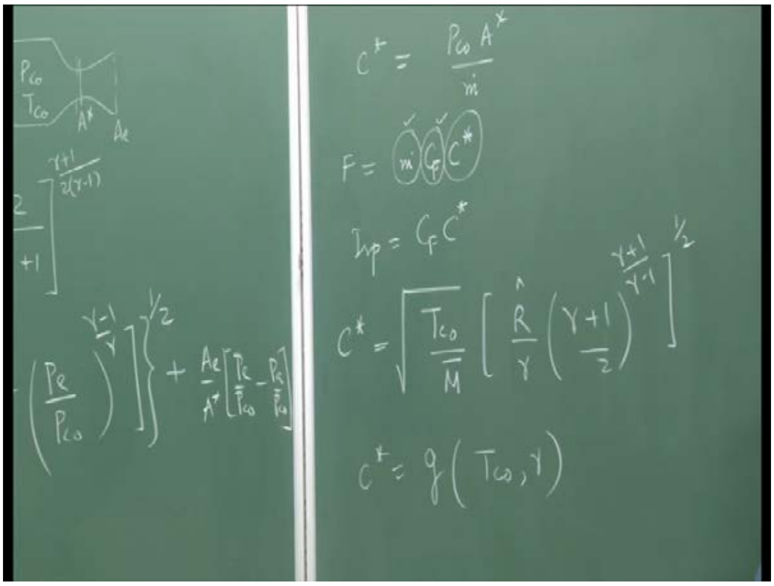

After this, we had defined the characteristic velocity as P c naught A star upon m dot. I have said in the previous class that the thrust coefficient essentially decreases the performance of the nozzle and the characteristic velocity decrease the performance of the combustion chamber. So, with this we have proved that total thrust is the product of mass flow rate, thrust coefficient and characteristic velocity. And based on this we have also proved that the specific impulse will be product of C F and C star, we have proved that in the last class. After that we had derived an expression for C star and we have shown that C star is equal to this. So, we have shown that in the last lecture, and we had stopped here at the end of last lecture. Now, we have the two expressions for specific thrust sorry, the thrust coefficient and characteristic velocity, we see that I s p is the product of this two. From our machine requirement, from the very beginning we have been saying that we want to maximize the Is p so that our equivalent velocity will be higher.

Now, the two parameters that dictate, if I look at this expression C star that is primarily a function of P c naught and gamma, so gamma is present in both and of course the molecular width, so for a given fuel it is basically fixed . So now, again gamma for a given fuel or a given composition will be fixed because the final composition of product will dictate what will be value of gamma. So, the operating conditions that we have is essentially P c naught and P e by P c naught. Once again I have said that for the given area ratio P e is fixed for the given P c naught, so therefore this can be replaced by P c naught. So, the two parameters that we have are the stagnation chamber pressure and stagnation chamber temperature. Now, coming to this expression or rather this expression, first of all we can increase thrust in three ways looking from this equation, either we increase the mass flow rate or we increase the C F or we increase the c star.

Let us say we want to increase the mass flow rate, and the parameters that we have are P c naught and T c naught, so here is our expression for mass flow rate. We see that in order to increase mass flow rate we have to either increase the stagnation pressure or decrease the stagnation temperature. If we increase the stagnation pressure, C F is increasing, so in this equation then both, this will increase, this will increase, and C star is not dependent on stagnation pressure to that extent. So, therefore, the thrust is going to increase if we are going to increase the stagnation pressure. Other way of increasing the mass flow rate is by reducing this T c naught, as we reduce T c naught, we can see from this expression C F is not that much affected. But if we reduce T c naught our C star is going to reduce, so characteristic velocity is going to reduce, and the net effect can be that the thrust may reduce, so that also we do not want. Therefore, the better way of achieving a higher thrust is by increasing both P c naught and T c naught.

However, there is a limit to P c naught, first of all the structure puts a limit because that will be the force exerted on the wall of the rocket by the gases, so structures puts a limit to that. Secondly, the temperature if it is very high there is a possibility of thermal defects on the wall; you need lot of cooling etc, and secondly very high temperature will lead to disassociation of the gases. And when the gases start to disassociate, gamma is going to be change which is something we do not want; disassociation actually is an endothermic reaction so it takes away some energy.

So, therefore, the overall energy is going to reduce because of that again the thrust produce will be less; I will discuss this disassociation effect again. So, bottom line is although we can increase both P c naught and T c naught to get higher thrust, but there is a limit to it, that is why we cannot keep on increasing the thrust as much as we want. Now, this discussion where we have defined C F and C star, we will allow us to attempt design of rockets, so that is what we are going to talk next. Simple rocket chamber, thrust chamber design, a nozzle design, and then I will solve a problem today on that.

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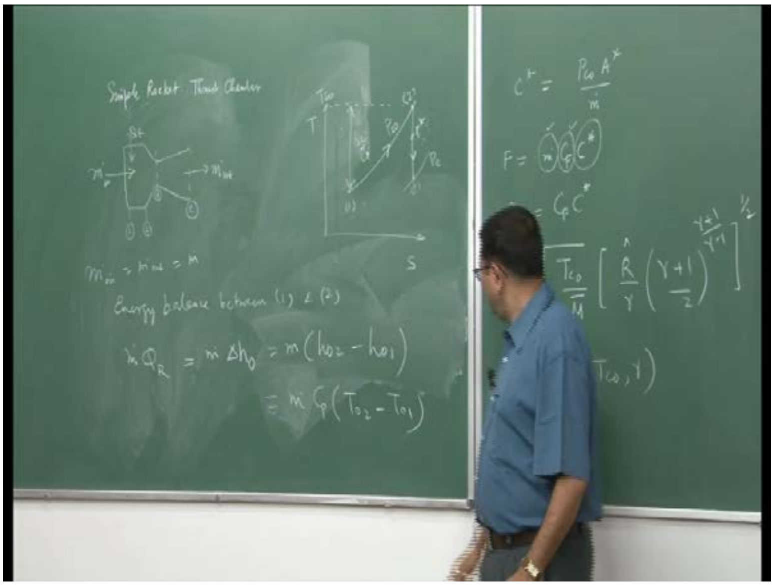

Let us now look at simple rocket thrust chamber, the rocket thrust chamber is a rocket motor and the nozzle because that produce the thrust. Let us consider that the inlet of the combustion chamber is one, the exit of the combustion chamber or inlet of the nozzle is two, the throat is given by star, and the exit of the nozzle is e; these are the different stations. Let us also consider that the amount of flow going in is m dot in; this is the combination of both fuel and oxidizer all the propellants. Let us also consider that because of chemical reaction the heat liberated is QR, so that is the Q that is added to this flow rate, and then this is the same flow that goes out, right.

So, first of all m dot in is equal to m dot out because the mass flow must be conserved. If I draw the TS diagram for this process; TS temperature entropy diagram of this process, at the beginning of the thrust chamber at one we are here. Then what is happening here in the combustion chamber, in the combustion chamber we consider the process are to be isobaric because it was considered constant pressure process that is what was proved for isentropic process, so we consider the process of heat edition to be a constant pressure process or isobaric process. So, therefore, in the combustion chamber between one and two, there is a constant pressure heat edition.

So, if I plot it, it will be going to two, and the pressure remains constant at P c naught. Now, because of the heat edition there is an increasing temperature, right, so the temperature here at two is now T c naught, so this is the chamber temperature. Now, this raise in temperature depends on the amount of heat that is added, right, so this will be equal to QR upon C P that can be proved from energy values. Now, after the combustion chamber in the nozzle there is an expansion; we are considering the expansion to be ideal, so it is isentropic, so it goes round vertically. So, this is our point e exit, this is the pressure at the exit given by P e; we are not commenting about P e, but it is isentropic, so it is a ideal expansion nozzle. In between, somewhere it crosses the critical condition, also P star, so from two to P star is isentropic, P star to e is isentropic. So, this is the cycle that the flow goes through from one to two to star to e, so once again one to two is a constant pressure heat edition, and two to e is isentropic expansion; these are the two processes.

Now, let us see what is happening, first of all the energy is added in the combustion chamber, so let us first do a energy balance for the combustion chamber; so energy balance between one and two. The total heat added is m dot which is the mass flow rate, times Q R this is the total heat that is added, and then this is the flowing process, so from thermodynamics; you know for the flowing process of first law of thermodynamics, since there is no work the only work is the P d P work, therefore the heat added is equal to the change in enthalpy; stagnation enthalpy, so the stagnation enthalpy change. Now, the stagnation enthalpy change is nothing but the stagnation enthalpy at two minus stagnation enthalpy at one that is the exit of the combustor minus the inlet of the combustor, so this is equal to m dot h 0 2 minus h 01.

Now, we have assumed the working fuel to a perfect gas for ideal rocket, which means it, is thermally perfect, so therefore, h is equal to C PT, and we consider C P T to be constant. So, therefore, this can be written as m dot C P T 0 2 minus T 0 1, that is what we see here, that is what I have written here QR by C P is equal to T 0 2 minus T 0 1. So, from this equation now, I can derive the value, estimate the value of T 0 2.

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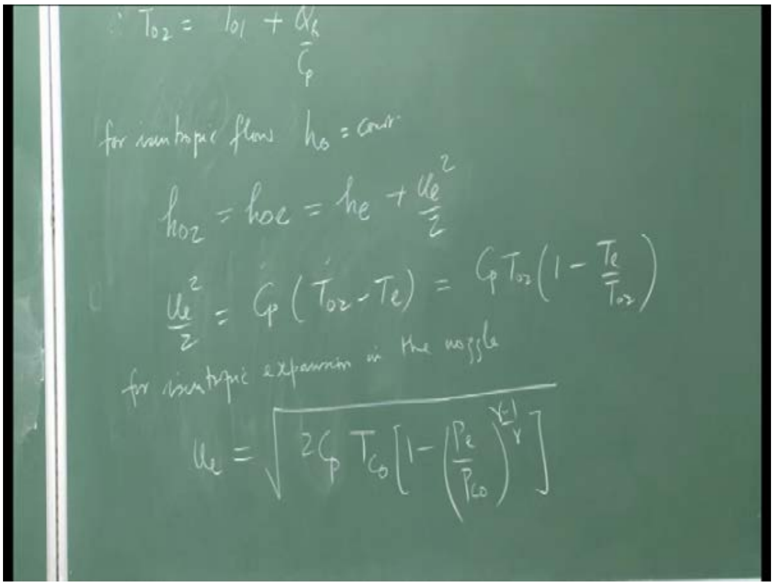

So, if I do that, from this equation I will get T 0 2 which is the stagnation temperature at the exit of the combustor which will be our T c 0 is equal to T 0 1 plus Q R upon C p. So, if the fuel oxidizers are given, the propellants are specified, then for a particular propellant combination there is a particular heating value Q R, so that is the known quantity. Then from the product, C P is also a known quantity. So, if this two are given, initial temperature is given then we can estimate the final temperature.

Now, in the nozzle the flow is isentropic, and we have proved that for an isentropic flow h naught is constant. Therefore, h 0 2 is equal to h naught e; stagnation enthalpy remains constant. So, stagnation enthalpy at the inlet of the nozzle is equal to that at the exit of the nozzle, and h 0 e is equal to the static enthalpy at the exit plus the kinetic energy term U e square by 2. Therefore, we get U e square by 2 is equal to C P into T 0 2 minus T e. So, now, in this equation our C P is known, T 0 2 is known, if we can estimate T e we can estimate the exit velocity U e; we are considering the nozzle to be isentropic.

So, therefore, the process for the nozzle flow is to be considered to be isentropic, so we can use the isentropic relationship, and then for isentropic expansion in the nozzle we get the exit velocity is equal to square root of 2 C P T c naught into 1 minus P e upon P c naught to the power gamma minus 1 by gamma. Here, P c naught is equal to T 0 2 as I have said in the combustion chamber stagnation temperature. Here, what we have done is we have written this as C P T 0 2 into 1 minus T e upon T 0 2, and then T 0 2 we have put equal to T c naught, P by T naught 2 is replaced by P by P c naught to the power gamma minus 1 by gamma, this is the isentropic relationship. So, since we are considering the expansion to be isentropic, we can write it like this. So, for the isentropic condition nozzle, this is the expression for the exit velocity. So, this is first time now we have an expression for the exit velocity U e, right; this we have also seen in caster mines, same expression. So, this will now give us the exit velocity at the exit of the nozzle. What we will do is just simplify this little more, we will express it in terms of the known quantities which are given to us. So, in order to do that, what I will do is T c naught I will replace by this equation.

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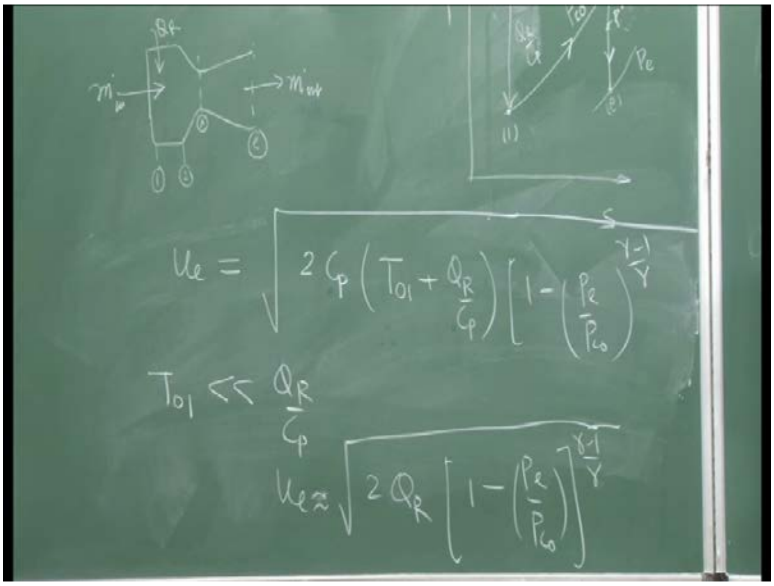

So, if I do that I will get the exit velocity equal to square root of two C P T naught 2 equal to T naught 1 plus Q R upon C P into 1 minus P e upon P c naught to the power gamma minus 1 by gamma. Now, we are doing an engineering estimate, as engineers we always make some approximations, now let us make some approximation. Let us look at this term T 0 1 plus Q R upon C P. T 0 1 is the incoming flow temperature; typically for caster mines when the flow enters the combustor it has gone through the compressor so is fairly hot about 400 500 Kelvin temperature will be there, but for the rocket it is coming directly from the supply, usually it is not hot, so temperature is low, so T 0 1 is typically low. And then combustion occurs there is a raise in temperature, temperature can increase by about 2000 say up to 2000 3000 Kelvin; quite a bit of temperature raise. So, this delta C P which is a temperature raise is much larger than the incoming temperature, so essentially we can write T 0 1 is much smaller than Q R upon C P. Now, if we make this assumption then from this term we can neglect T 0 1, then U e will be simplify to as a function of which is approximation 2 Q R up, now once we make that assumption we have a C P here, and a C P here, so this two will cancel of, so it will be approximately two Q R and only this term 1 minus P e upon P c naught to the power gamma minus 1 by gamma.

Now, once again we are noticing something, we have eliminated the temperature again. What the exit velocity is a function of; this is a heating value, this is the propellant property, right, it is the heating value of the propellant and P e upon P c naught, but there is a gas to it. Here, we are considering that the entire available chemical energy has been converted to the thermal energy or heat, right. So, we are assuming that the entire Q R is released, that is why we are eliminating T naught, then T c naught becomes a function of the total heating value; in reality it may not happened, there is something called combustion efficiency, all the energy may not be released. In that case, the temperature comes into picture, or what we do is we say that ninety percent energy is released, so we multiplied with the combustion efficiency and then accordingly we can adjust to the requirement.

Now, this is our expression for the exit velocity. Here, we can also express the heating value as heating value per unit mole. If we do that then molecular weight will also come into picture like we had done earlier when we got in the molecular weight in characteristic velocity, so same thing we can do here also.

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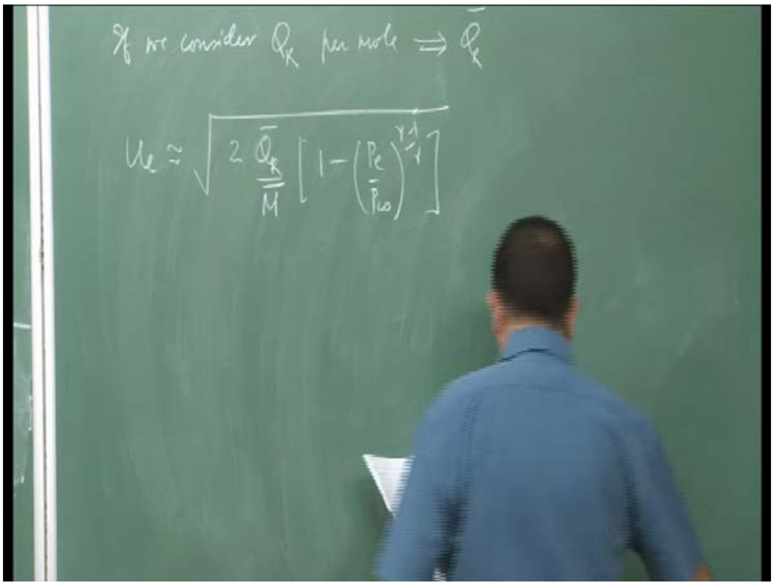

So, if we consider Q R per mole that is Q R bar, then my exit velocity will be equal to square root of 2 Q R bar by molecular weight into 1 minus P e upon P c naught to the power gamma minus 1 by gamma. Here, this Q R needs to be calculated using thermo chemical analysis which is the part of the combustion analysis as can be done which we will do later in the later part, when we will go to the combustion we will there study how to estimate the value of Q R. So, at present this is our exit velocity.

So, now, we have the exit velocity estimated, so for a given operation now we can decide whether a rocket is going to fulfill the mission or not. Now, to continue from here what I will do now is solve the problem to make this concept more clear. Essentially what I am going to do is, I am going to design a rocket; when we are talking about design your boss will tell you that you have to produce certain amount of thrust, right, that is what the rocket designer will make. Then how do you design it that is the question, so let us solve a problem. So, the question is that we are supposed to design a rocket which means the thrust chamber as well as the rocket thrust chamber; so rocket thrust chamber means the combustor and nozzle. So, we have to design a rocket thrust chamber, I will give the conditions or the parameters under which we have to design, so design of a simple rocket thrust chamber.

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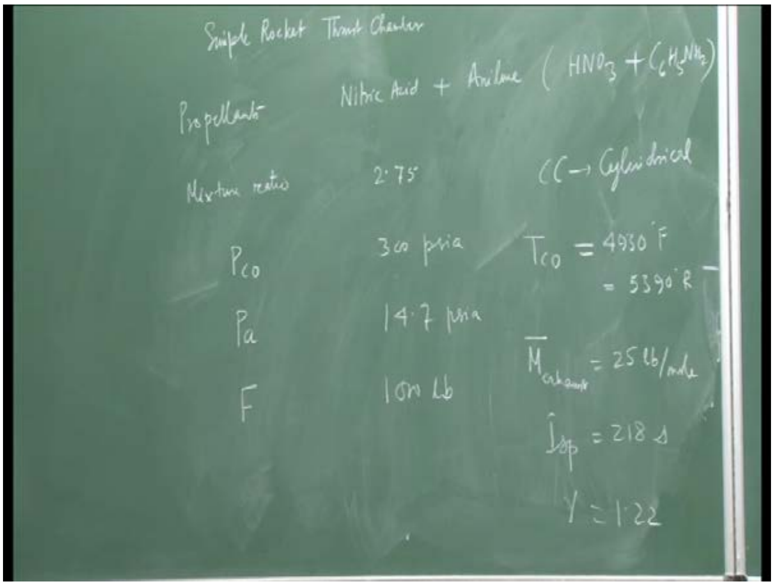

Let us say that the propellants first of all is a very important parameter because this M bar comes here, Q bar comes here, Q R bar, these are all propellant properties, gamma is also propellant property. So, first let me say that the propellants are given, it is nitric acid plus aniline; both of them are very commonly used liquid propellants in rockets. So, the chemical composition is, so these are the propellants. Let us see now, it is also said that the mixture ratio that is the ratio of nitric acid and aniline is equal to 2.75; at present this information we will not need because that will go into the chemical composition and chemical analysis. Here, now we are not doing the thermo chemical analysis, so we need this information, but for completeness of the problem we are providing that.

What we are going to need is the combustion chamber pressure P c naught, so stagnation pressure in the combustion chamber is specified to be 300 psia. I will give this problem in British units, fps units because of the fact that this is essentially an American problem; typically these aniline nitric acid propellants are used by Americans, so American vote to this system of units, I will give this system of unit. We can convert this also to SI units, so P c naught is equal to 300 psia. Atmospheric pressure, let us say this is the initial stage, the first stage, so therefore, the atmospheric pressure is the sea level atmospheric pressure which in absolute term is 14.7 psia, so that is the atmospheric pressure.

Let us say that this rocket, this is the most important parameter needs to produce thousand pound thrust that is the thrust this rocket needs to produce. We will give some more parameters, one constrain is that the combustion chamber or the thrust chamber should have a cylindrical combustion chamber, so the combustion chamber is cylindrical. Then, the chemical analysis of this as I have said that it will come from the chemical analysis, so from the chemical analysis of the product we get certain parameters for this mixture ratio, for this propellant. So, the chemical analysis results give that the T c naught, combustion chamber stagnation temperature is 4930 Fahrenheit which is also equal to 5390 rancid. Then the molecular weight of the exhaust at the burn product is given to be 25 pound per mole.

Now, when we are designing this rocket we have to produce certain I SP; we are designing it for that, so the I SP is specified, let us say the I SP is equal to 218 seconds; that is what we need, and again from the chemical analysis we get gamma equal to 1.22. So, for these conditions we have to design a thrust chamber that is the combustion chamber and the nozzle. We will first start with the nozzle, so first let us look at the nozzle configuration.

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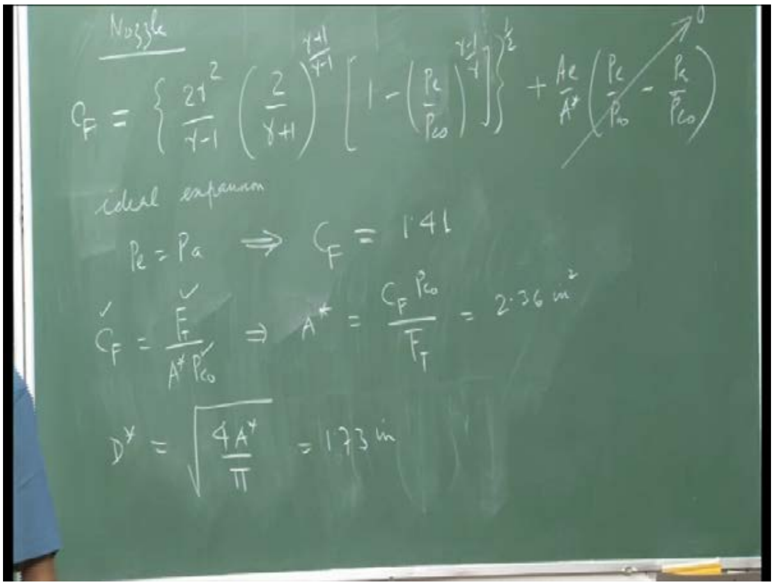

For the nozzle what is the important parameter, it is the thrust coefficient. So, first for this case let us estimate the thrust coefficient; we know the thrust coefficient is given as this plus, this is the general expression for thrust coefficient. Now, here is the design problem, we do not know the area ratio, so first thing is we start with an assumption that is an ideal expansion, so let us assume ideal expansion.

Now, with this assumption then P e is equal to P a therefore, the pressure term in the thrust coefficient goes to 0, so we are left only with the momentum term, and exit pressure is equal to the ambient pressure, so this P e is equal to P a. Now in this equation the value of gamma is given, P e is equal to P a which is given, P c naught is given, so I can directly calculate the thrust coefficient. So, if I do that, so therefore, P e is equal to P a, this implies the thrust coefficient after putting all the values will be equal to 1.41. Now, what does thrust coefficient give you, why first we are estimating the thrust coefficient because we are supposed to design the rocket nozzle, right, so design essentially means you have to estimate the area. How do I estimate the area, I get this thrust coefficient. Now, let us look at the definition of thrust coefficient, thrust coefficient is defined as the thrust by A star T c naught, right, that is how I have defined the thrust coefficient.

Now, let us look at this equation, the thrust coefficient I have estimated already, the thrust is given, the stagnation pressure is given. So, only unknown in this equation is A star; the throat area, so I can solve for the throat area from here as now the values are given, so I can directly solve for this, this comes out to be equal to 2.36 square inch. So, 2.36 square inch is my throat area therefore, I can get D star which is the diameter equal to square root of 4 A star by pi, this is equal to 1.73 inches. So, that is my throat area for this rocket 1.73 inches, now once we have this let us take the next step.

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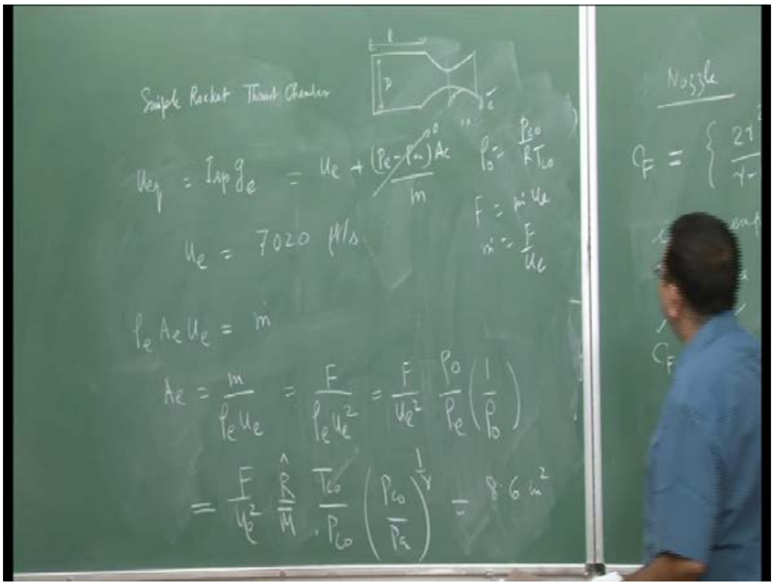

So, first now we are designed the throat, lets next look at the exit area. So, let me draw the stigmatic here, what we have to design is this A star, we have to get this exit area A e, we have to get length of this combustor l, and the diameter of this combustor D, that is what we need to design. So, first of all from here I get A star, next step is to get the exit area A e. In order to get the exit area we need to know the exit velocity, so next let us calculate the exit velocity. How do we calculate the exit velocity, lets again look at what has been given to us, what is given to us is the value of I SP, specific impulse. So, now, use the definition of equivalent velocity which is equal to specific impulse time acceleration due to gravity at sea level. Now, the equivalent velocity is U e plus P e minus P a upon A e by m dot. But, in this case we are considering ideal expansion, so P e is equal to P a therefore, this term is 0. So, exit velocity is equal to nothing but I SP times g e, I SP is given to 18 second, g e is the acceleration due to gravity which is known constant, so I can get U e, this terms have to be equal to 7020 heat per second, I have this value.

Next, we need to calculate the exit area, for the exit area what I will do is I will use continuity equation. So, from continuity equation exit density times exit area times exit velocity is m dot, total mass flow rate. Therefore, the exit area is m dot upon rho e U e, and what is m dot from the definition of thrust for this case, from the definition of thrust F is equal to m dot U e, so I can replace m dot as F upon U e, right. So, in this equation let me replace m dot by F upon U e, so I get F upon rho e U e square. If value of the thrust is given the exit velocity we have estimated, now we need to know the density. How do we get the density, for that what I will do is I will multiply and divide by rho naught; rho naught is density in the combustion chamber or stagnation density, so this is something 1 by rho naught.

Since, we are considering the gases to be perfect gas, rho equal to P by RT, right, so for perfect gas rho equal to P by RT, so rho naught equal to P c naught upon RT c naught. So, I can put it here, and rho naught by rho e can be given in terms of pressure ratio because it is an isentropic process. So, if I do that, this finally comes out to be equal to F upon U e square into, this is the value of R universal gas constant divided by the molecular weight; R was coming there, into T c naught by P c naught into P c naught by P a to the power 1 upon gamma; here this ratio comes from the isentropic relationship relating rho naught by rho to P naught by P. Now, in this equation my thrust is known, exit velocity have estimated, this is the universal gas constant which is the known value, this is the molecular weight which is given, this is given, this is given, this is given, so everything here is given, I can just plug in the values and get the final area that just comes out to be equal to 8.6 six square inch. So, now, I have this area also, I have this area, I have this area, I can estimate the exit diameter.

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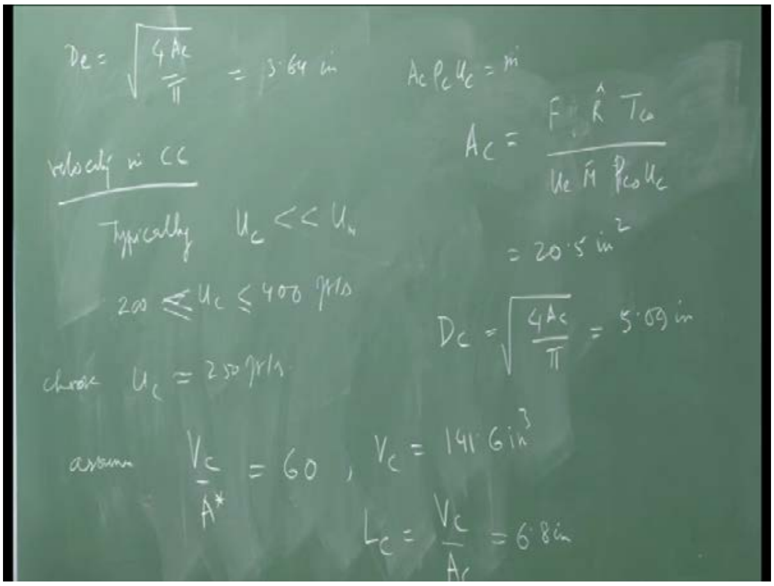

The exit diameter D e is equal to square root of 4 A e upon pi, so this is 3.64 inches; about less than 4 inches, so this is the small rocket we are designing. Now, we have these two parameters, so now, we have the nozzle configuration completely; one thing that nozzle is essentially, I am getting the throat area and the exit area. I would like to point out here that the shape of the nozzle still we are not talking about because we have not discussed that, later on we come to the shape of the nozzle. At present we are considering a shape nozzle but we are specifying what should be the shape, so the nozzle part is done.

Now, let us look at the combustion chamber next. So, for the combustion chamber what we need to know is the velocity in combustion chamber. Typically the velocity in the combustion chamber is given by U c is much less than that in the nozzle which is quite obvious because combustion chamber conditions are almost stagnant whereas, nozzle has a high velocity flow. Now, this is something that we assume, this is a design parameter that we assume, so typical values of U c is between about 200 to 400 feet per second, that is standard values of U c, U c lies between 200 to 400 feet per second, the combustion chamber velocity. For this problem, let us consider or choose that U c is 200 and 50 feet per second; this is value 250 feet per second velocity.

Now, with this velocity, then first of all we use the continuity equation for the thrust combustion chamber this is equals to rho c A c U c which is equal to m dot. Once again m dot can be replaced by F by U e equivalent as we have done here, so if I come to this equation now I can replace m dot by F by U e equivalent, this U e equal to U c, this is rho c. So, everything here can be replaced, I can get an expression for the area of the combustion chamber, this we can write it as this one. See, this term F by U e equivalent is my m dot, this term R by M T c naught, sorry this will be seen now by P c naught, this is my rho naught that is the density in the combustion chamber, and this is U c , right, so we have m dot by rho c U c . Everything here is known, so I can get the combustion chamber area, this is equal to 20.5 per square inch.

Therefore, the combustion chamber diameter is about 5 inches that is the combustion chamber diameter. So, once we have estimated the combustion chamber diameter next we need the combustion chamber length. Now, length is something that depends on characteristic time we are known, and the combustion instability parameters. At present what I will do is we will consider a length factor which is typically given by the ratio of combustion chamber volume to the critical area. So, let us assume combustion chamber volume two A star, is typically a given parameter which has been evolved through experiences, so typical value of this about 60, so let us assume this value is 60. A star we have estimated, so this gives us the combustion chamber volume to be 141.6 inch cube, so roughly about 140 inch cube.

Therefore, the length of the combustion chamber will be volume of the combustion chamber divided by the area of the combustion chamber this is about 6.8 inches. So, if I draw the schematic now with the proper dimensions, first let me write down this dimensions, I will try to make it as much to the scale as possible.

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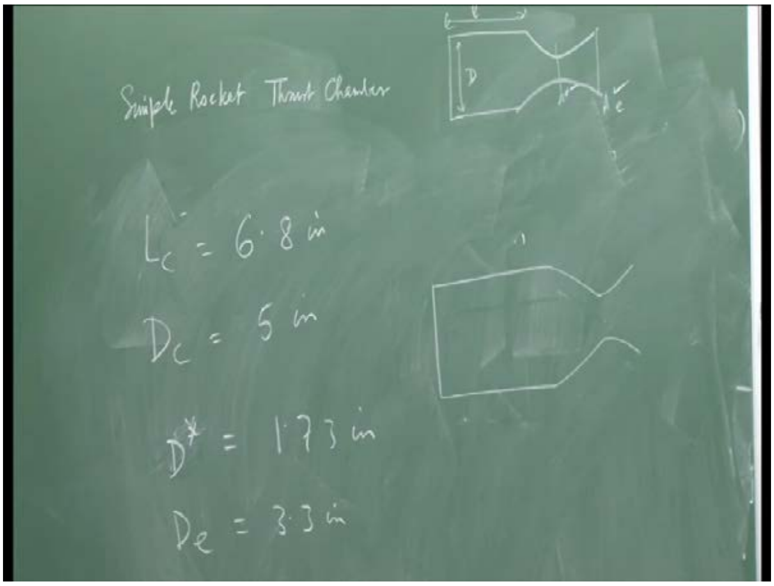

So, first let me write it that the length of the combustor I am getting is 6.8 inches, the diameter of the combustor is about 5 inches, the throat area sorry, the throat diameter is was about 2 point something, right, the throat diameter was equal to about 2.6, no 1.73 inches, and my exit diameter was 3.3 inches. These are the four parameters that we have designed now, estimated from the design, so sorry 3.64, so 3.64, no no sorry this is 3.3 inches.

So, if I draw it to the scale roughly about 5 inch, about 7 inch, here it goes down about 2 inches, around 3 inches. This is your rocket, with the propellant that we have about this size rocket will produce 1000 pound of thrust with the given flow rates and everything. So, that is a small rocket which will produce the thrust, actually this kind of rocket can be designed in a laboratory and can be tested, that is one thing I wanted to show that how we design a rocket. I like to point out here once again that for the nozzle we have considered only the throat area and the exit area, we have not considered the shape of the rocket, that will come later when we go into more detail of rockets, now rocket nozzles, so that is what we are going to take up next, the design of rocket nozzle.

Before that, first we define the efficiency parameters, then we go into the design of rocket nozzles, and then we see how this rocket shape is designed. Once that is done then we go back to the combustion chamber, and we will do the some more chemical analysis to get the composition, to get the temperature etcetera, we will discuss how those things are obtained. So, the next topic is continuation of our discussion on nozzles, and now we go into the shape nozzle. I will stop here today, in the next lecture I will start from there.

Thank you.