# Introduction to Aerospace Propulsion

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## Lecture No. # 38
## Various space vehicles and their engines

We have been talking about rockets, missiles and space crafts and their engines. So, fundamentally we are talking about the rocket engines. We have covered some ground regarding the fundamental parameters that define the rocket engine performance.

We have also tried to look at how some of these engines actually they look like, and to some extent, how they are designed. What are their fundamental configurations that allow them to perform under various operating conditions? How they lift off from the ground? And to some extent, what you could expect their behavior to be when they are at a high altitude or even in vacuum; that means they are in space, not working under earth's atmosphere.

So, these are the various issues that we have looked at from fundamental point of view, and in this course, we are looking at some of the very fundamental issues related to rocket engineering. Today, we will look at some of those issues including the propellant characteristics of specially the liquid propellants and how they fair in terms of their fundamental parameters. We have already seen that liquids and solids. One has a large number of choices regarding the propellants.

Unlike in aircraft, where, you have basically only one kind of fuels. Here, you have a large number of fuels in terms of propellants and oxidizers, and as a result of which, they are characterized by some of their characteristic parameters or numbers and we will have a look at some of these, especially of the liquid propellants, and later on, we will look at how the nozzle of these engines are actually designed and how their characteristics are predicted or computed from fundamental nozzle theories.

So, some of these things would use the fundamental theories of aerothermodynamics. That you have done earlier in this course and may be a little bit of fluid mechanics or gas dynamics, you many have done in some other course. So, all of that put together, we will have a look at a very simple and fundamental nozzle theory and that will allow us to take a good look at how the nozzles behave, and fundamentally, how they contribute to the making of thrust of rockets.

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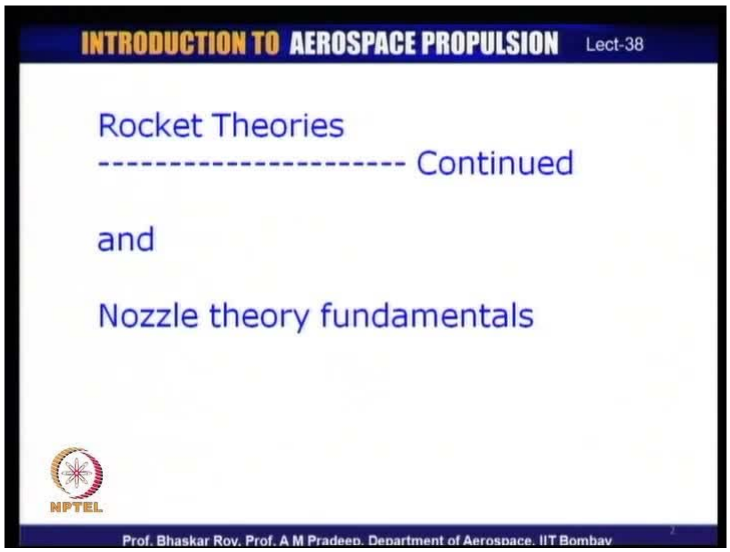


So, to begin with today, let us start off with some of the fundamental issues all over again. We are talking about rocket theories and we will continue with those rocket theories in terms of their liquids and then we will go on to the nozzle theory fundamentals in today’s lecture.

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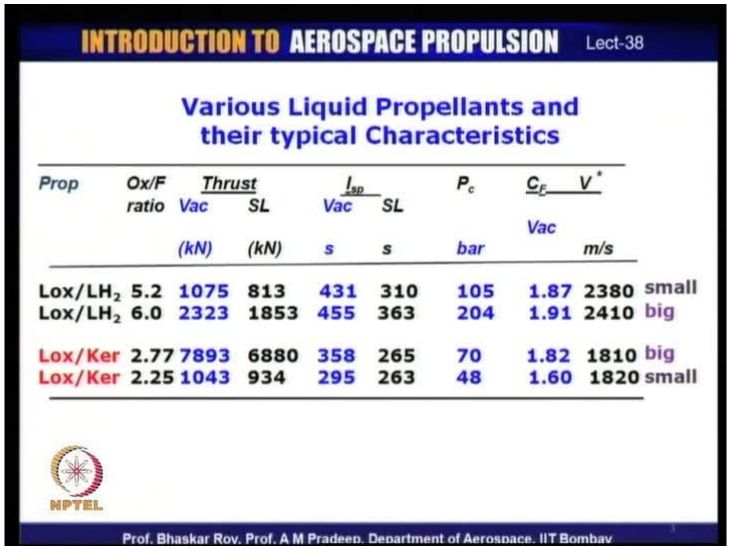


| Prop | Ox/F ratio | Thrust Vac (kN) | Thrust SL (kN) | $I_{sp}$ Vac s | $I_{sp}$ SL s | $P_c$ bar | $C_F$ Vac | $V^*$ m/s | |
|---|---|---|---|---|---|---|---|---|---|
| Lox/$LH_2$ | 5.2 | 1075 | 813 | 431 | 310 | 105 | 1.87 | 2380 | small |
| Lox/$LH_2$ | 6.0 | 2323 | 1853 | 455 | 363 | 204 | 1.91 | 2410 | big |
| Lox/Ker | 2.77 | 7893 | 6880 | 358 | 265 | 70 | 1.82 | 1810 | big |
| Lox/Ker | 2.25 | 1043 | 934 | 295 | 263 | 48 | 1.60 | 1820 | small |

Now, we have discussed that rockets are often characterized by number of parameters. Now, these parameters we have talked about thrust of course, is a total measure of the performance, but if you look at some of the characteristic parameters, those were the specific impulse - I sp, the specific thrust, that is C F and characteristic velocity in terms of meters per second - V star. Now, these are characteristic numbers and they are not necessarily connected to the size of the rocket. So, we can look at some of these numbers here and connect them to the propellants that they are using. For example, if you have a liquid oxygen and liquid hydrogen as propellants and their ratio oxidizer to fuel ratio is 5.2.

Let us say a comparatively smaller rocket would create a thrust of 1075 in vacuum and a little lesser in sea level 813. It would create I sp of 431 in vacuum and at sea level 310. The combustion chamber pressure, now, that is an absolute value and that could be 105. The C F, the coefficient of thrust is 1.87 and the V star characteristic velocity is 2380. Now, this is for a comparatively smaller thrust rocket engine. Somewhat bigger thrust rocket engine of liquid oxygen and liquid hydrogen gives us oxidizer fuel ratio of 6. The thrust of a much higher order in terms of kilo Newton’s of 2323 and this is almost double of the earlier thrust rocket engine.

And at sea level, it is 1853 which is more than double of what was given by the smaller engine. On the other hand, if you look at I sp is 455, it is indeed actually just a little on the higher side and pretty close to the earlier value of 431. At sea level, it is 363; a little higher than 310, that was for the smaller rocket, and as a result of which, the characteristic values are pretty close to each other; they are not necessarily dependent on the size of the rocket engine.

The combustion chamber pressure on the other hand is decided by the ratio, and hence, it is of the higher side; it is 204 that one expects or guess would be dependent on the design of the rocket motor. Again, if you look at thrust coefficient, it is 1.91 which is pretty close to 1.87 for the smaller rocket, and the characteristic velocity is 2410, which is also very close to the smaller rocket.

So, you can see that for bigger rockets, one may use a slightly higher oxidizer fuel ratio which could be different, but their thrust values would be different. They have to be higher to lift off a much bigger rocket vehicle, but the characteristic parameters like I sp or C F or V star would be more or less of the same order. Now, if we look at the other propellant combination, that is liquid, oxygen and kerosene, we can see that the oxidizer fuel ratio of a bigger engine is 2.77. The thrust is 7893 at in vacuum and at sea level is 6880.

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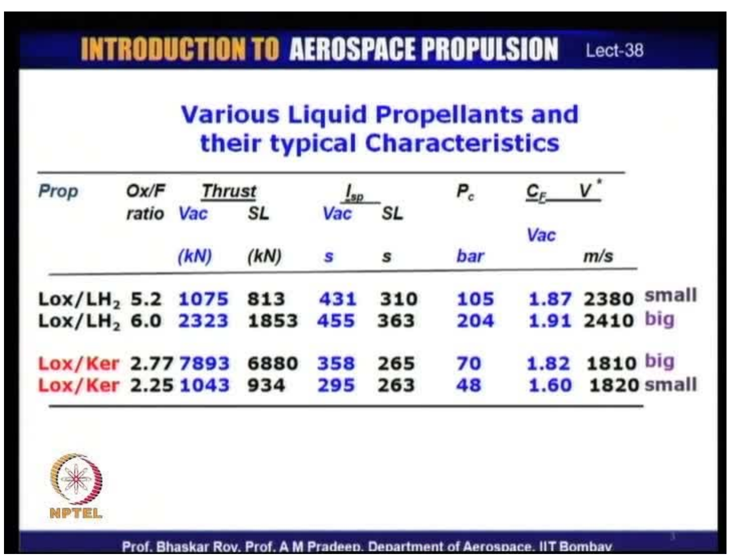


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The I sp on the other hand is 358 which you see is somewhat lower than the liquid oxygen hydrogen I sp, and at sea level, that is 265. The combustion chamber pressure as you can see is somewhat on the lower side, it is 70 bar. C F is 1.82 and V star is 1810. Now, this probably this is a big engine. As you can see the thrust value was very high even compared to the liquid oxygen hydrogen values and this was probably one of the bigger engines presumably that of Saturn engine of U S.

Now, if you take the other liquid oxygen and kerosene fuel ratio of 2.25 fuels to kerosene ratio, it gives a small thrust of 1043; it is a very small engine. The sea level thrust is correspondingly 934 only. On the other hand, I sp is 295, which is not much smaller than the 358 in vacuum of the bigger engine, and at sea level, the I sp is 263, which is pretty much close to 265 of the bigger engine.

The P c - the combustion chamber pressure - is 48 somewhat lower and the C F is 1.60 again pretty close and V star the characteristic velocity is 1820, which is actually slightly bigger or larger than that of the bigger engine using the same fuel. So, these numbers actually show that the size of the engine does not decide or not decided by the I sp values, they are decided essentially by the propellant combination that is being used.

So, similar propellant combinations of other propellants that we have discussed earlier would give characteristic values of I sp C F and V star and they would be valid more or less of the same value irrespective of whether you use them for small engine or for a big engine. They are somewhat dependent on the oxidizer to fuel ratio, but they are not dependent on the size of the rocket engine that you design.

So, this is something which is we have discussed before and these values essentially underline the fact that the characteristic values are decided by the propellant characteristics, propellant chemical characteristic, physical characteristics and their ratio and are not decided by the size of the rocket motor.

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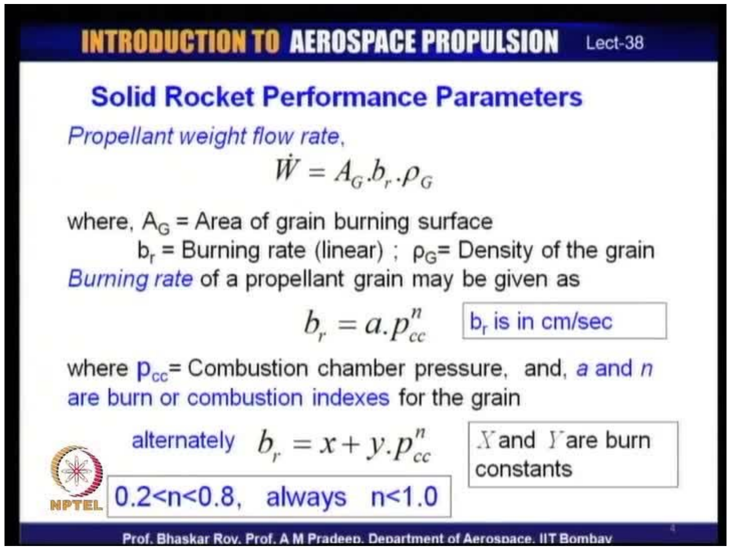


So, if we can now move on to some of the solid rocket performance parameters, if we look at the fundamental issues that we have discussed before, one is that that the rocket always uses propellant use in terms of weight flow. And if we use that, we get a weight flow of w dot which is as usual as for continuity condition is decided by the grain area of the burning surface. The rate of burning and the density of the grain at which, it is being housed inside the combustion chamber.

Now, the burning rate which is often a linear burning rate of a propellant grain is given in terms of b subscript r and that is a into p cc to the power n - where b r is normally measured in terms of something like centimeters per second; millimeters would be little larger value and meters could become the number could be a rather small. So, centimeters per second could be a proper choice of units for burning rate.

On the other hand, the value, the numbers given here in terms of a and n are the burn or combustion indexes for the grain. Now, these numbers depend on the grain design and the grain composition, and hence, they are decided by the grain choice of grain that one is using for this particular rocket. P cc is a combustion chamber pressure. Now, this could also be given in terms of another possible equation and that is x plus y into p cc to the power of n - where x and y are the burn constants of again dependent on the characteristics of the grain that are being used.

Now, the value of n is typically somewhere between 0.2 and 0.8 and must necessarily be less than 1. So, those values are decided by certain grain characteristics and quite often the known grains have their known values of a or x or y and n. So, those are available for the known propellants.

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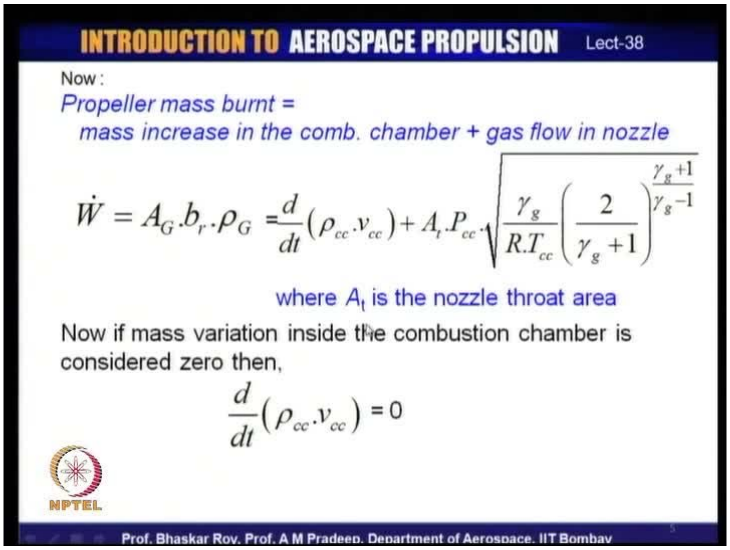


Now, the propellant mass that is burnt there the usage of the mass and that is what that comes out from the nozzle for creation of thrust. That can be given in terms of normally we are giving in terms of weight flow and that is normally the mass increase in the combustion chamber. That is the mass that is burnt from the grain and the gas flow that is happening through the nozzle.

So, you have two terms here - the first term is essentially representative of the gas that is burning inside the combustion chamber and continuous creation of that burnt gas by the burning of the grain. The other is the gas flow and it is possible at a certain point of time. You could have a situation that mass variation inside the combustion chamber is 0, because the amount of mass that is being created exactly same amount of mass is also being expended through the gas nozzle.

And as a result of which, the net mass that is being added to the combustion chamber could become 0; that means the entire rocket motor inclusive of the nozzle could reach steady state operation in which the mass created by the grain burning and the mass expended through the nozzle are equal, and as a result of which, one can say the operating rocket motor is operating in a steady state condition. Now, this is a simple way of calculating the amount of mass. One may say this idealistic way of calculating, what the ideal mass expenses would be during the performance of the rocket.

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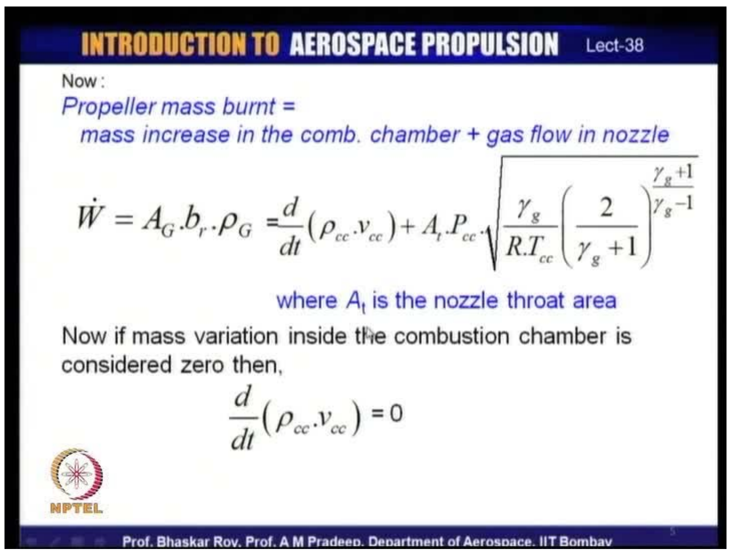


If we take this forward and one of the important parameters that one needs to find out with reference to the grain surface area that is burning. The other important parameter is a throat area of the nozzle. Now, grain area is what is the burning surface of the grain. Throat area is the area through which the burnt gas is finally flowing out. So, this ratio becomes an important figure of merit for the particular rocket motor, and as a result of which, one can characterize the particular rocket engine with reference to area ratio of these two areas.

And one can see that they are dependent on the fundamental performance parameters. The combustion chamber pressure, the grain density, a which we have used before and the basic characteristics gamma g of the gas that is being created the combustion chamber temperature, the R, the gas constant of the gas.

And these are the parameters that finally decide what the ratio should be. We can simplify this entire ratio, area ratio and write down simply that this is proportional to combustion chamber pressure P cc to the power 1 minus n or other way around P cc could be considered to be equal to this area ratio to the power 1 by 1 minus n.

Now, this expression means that the value of n. If it is large, the variation of burning surface A G will have large effects on the chamber pressure and on the propeller burning rate and thus the n should be low; that means the burning surface area, if it suddenly increases during the burning process, it will result to explosion and this is not what we would like to have, and hence, the value of n should be somewhat on the lower side.

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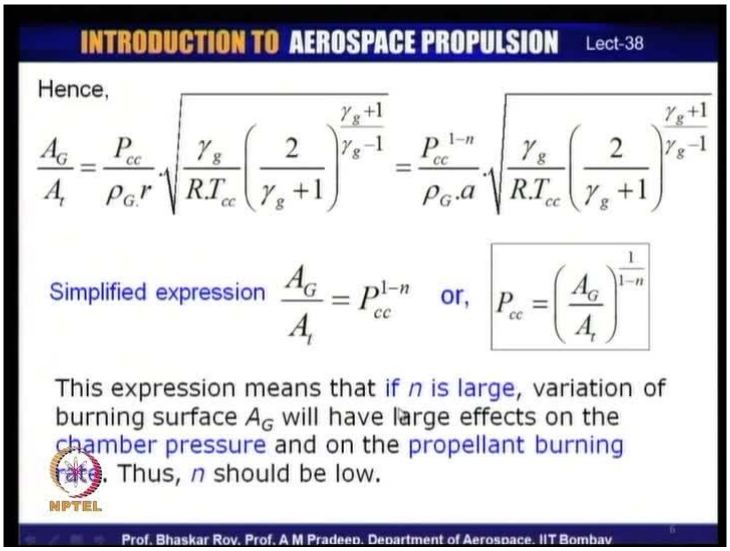


If we look at the last slide, we have seen that the value of n should be less than 1 and preferably it should be somewhere between 0.2 and 0.8 so that those are the numbers that we were talking about. And now, we see that n should be somewhat on the lower side and certainly not anywhere near one, which could lead to, possibly lead to explosion of the rocket motor. So, these are certain fundamental issues that can come out of the simple theory that we are doing.

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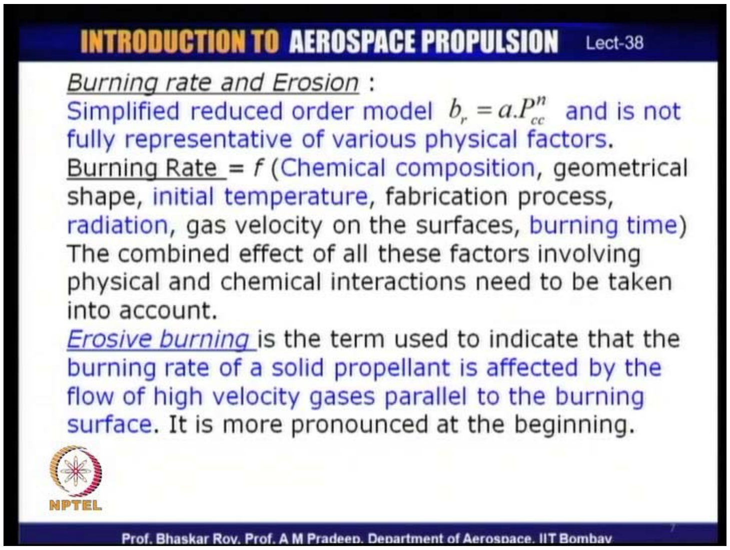


Now, the burning rate that we are talking about and we have written down simply in terms of b r equals to a into P cc to the power n. Now, this is a simplified expression. A simple way of mathematically catching the burning rate and gives us a first cut notion about what the burning rate could possibly be. In engineering, one always likes to have a number, a calculated number which gives us some idea about what is happening.

In reality, the burning is dependent on a number of parameters or number of factors. Now, as you can see here, the burning is a function of the chemical composition of the propellants, the geometrical shape of the solid rocket grain. The initial temperature at which, the burning was initiated. The fabrication process which is very important and there is a lot of mechanical engineering that we have discussed. Before that, it needs to be designed properly.

The radiation after the combustion is initiated, the combustion produces a radiation and this radiation could affect the chemical and physical properties of the grain. So, inside the grain body, since it is housed right in the combustion chamber itself, so, whether that is an important issue needs to be figured out before the grain is put inside the combustion chamber.

So, the radiation burning, radiation characteristics of the grain needs to be ascertained before it is put inside the combustion chamber and the gas velocity on the surface and we shall see just in a minute that this has a huge impact and the burning time, because the burning time tells you how much fuel is being burnt and how long the rocket will continue to perform, and combined effect of all these factors involving physical and chemical interactions are actual influential in finally deciding the burning rate.

So, all of it together burning rate, decide the burning rate and we have simplified, you know, reduced order model to capture the burning rate and we have shown it essentially dependent on the combustion chamber pressure which itself is dependent on all those parameters that we have written down.

So, the burning rate is indeed dependent on a large number of factors and some along the way a more rigorous process of finding out burning rate would have to be adopted to get a realistic number or value of the burning rate of any particular grain. Now, we have seen that when the grains burned, they create gas and this gas starts moving inside the body of the rocket chamber.

Now, when the gas starts moving and it picks of velocity, this gas may have on the surface of the grain a certain erosive capability. Now, this erosive capability obviously would harm the rocket grain and the erosive burning is one of the issues that needs to be dealt with and it is used to indicate that the burning rate of a solid propellant can be affected by the flow of high velocity gases on the burning surface and it could be parallel to the burning surface.

So, it flows on the burning surface; it flows towards the nozzle, and finally, the gas would go out of the nozzle, and this, when it first picks up right the beginning of the initiation of burning, the amount of erosion could be rather high. Later on, as we have seen the usage of the gas reaches a steady state, and at that stage, the erosion may not be a big issue, but in the initial phases, when the gases have burnt and they are accelerating towards the nozzle, the erosion could be somewhat on the higher side and some of these need to be studied very meticulously to find out what is the effect of erosion burning on the burning rate during the process of erosion.

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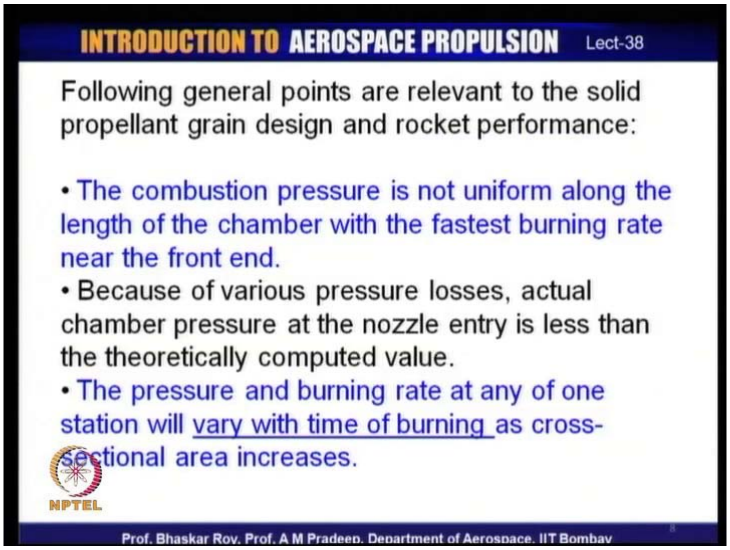


Now, some of the general points that we have discussed with reference to solid propellant grains, how they are designed and how they finally perform can be put together. The combustion pressure is not uniform along the length of the chamber with the fastest burning rate near the front, end. So, the combustion chamber varies from one end of the chamber to the another end. For some time, it may reach a steady state, but that is probably for a very short time depends on the rocket size and the chamber design.

We have to accept the fact that the chamber pressure is not uniform. One of the reason we have to keep this in mind is because we have used combustion chamber pressure as a figure of merit for calculating various rocket performance parameters. We have to keep in mind that this chamber pressure itself may not be a constant parameter, it may be a variable parameter, and hence, the basic performance will have to be then recomputed with a variation of combustion chamber pressure.

The other is the gas. When it is starts moving, it incurs pressure losses. So, the chamber pressure that is created; some of it is lost in the process of the movement of the gases inside the rocket grains, and by the time, it comes to the nozzle, certain amount of pressure may have been lost, and hence, the chamber pressure at the nozzle entry is quite often less depending on the grain design than the theoretically computed chamber pressure.

So, this also needs to be ascertained grain by grain, and hence, for every particular grain, the amount of pressure loses inside those grains when the gases are flowing inside before they come to the nozzle needs to be quantified and factored into the nozzle performance and thrust computation.

The pressure and burning rate at any one of the station will vary with the time of burning, because the size and shape of the grain would keep on changing the cross sectional area. That is shown is quite often changing, and as a result of which, the time of burning and many of these issues connected to the grain burning would actually change, because a grain shape and size are probably changing.

It depends on what the original shape and size was. If you have a simple cylindrical end burning, it will not change a great deal, but if you have all kinds of other sizes, it is entirely possible; they will change quite a lot during the process of burning, and hence, the pressure that is created and the burning rate at which they are burning would actually vary with the burning time, which means during the operation of the rocket, the pressure and the burning rate would continuously vary.

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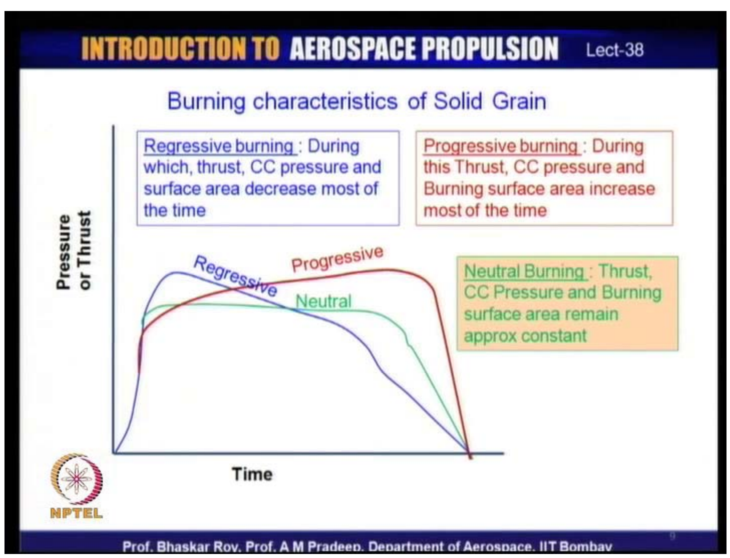


If we factor in this variable burning rates into three simple possibilities - one in which you have progressive burning, which is shown here in red that the burning rate evolves in such a way that during this process of burning, the thrust, the combustion chamber, pressure and the burning surface area increased most of the time with during the burning of the rocket.

Now, this you know pressure combustion, chamber pressure or thrust has been plotted against time and it shows that after the initial peak, it progressively shows more and more pressure and more and more thrust during the process of burning which we mentioned could reach a possibly a steady state. In this particular progressive type, it continuously increases till the rocket. The grain starts getting exhausted, and finally, the thrust and the pressure sort of fizzle out and rocket finally performance gets over, rocket performance gets over.

But till such time is there, it shows a progressive characteristic of rocket. The other possibility is a regressive where this performance in terms of thrust, combustion chamber pressure and surface area continuously decrease most of the time in a certain manner depends again on the grain design and the way the grains are used and their performance and their composition, and hence, we could have a regressive burning.

The third possibility obviously then is the one where you have neutral burning, which means the thrust combustion chamber pressure and the burning surface area remain approximately constant during most of the burning of the rocket grain. Now, this constancy has to be designed into the rocket grain design. So, as we can see here, one could possibly have three different fundamental possibilities of burning characteristics.

It needless to say that the most preferred one is the neutral burning capability of the rocket chamber and many of the rockets or the grains are indeed designed to create neutral burning capability. You may have slightly regressive capability or you may have slightly progressive capability, but very fast progressive or very fast regressive both are definitely not preferred choices, and hence, out of the three, the most preferred would be the neutral burning capability.

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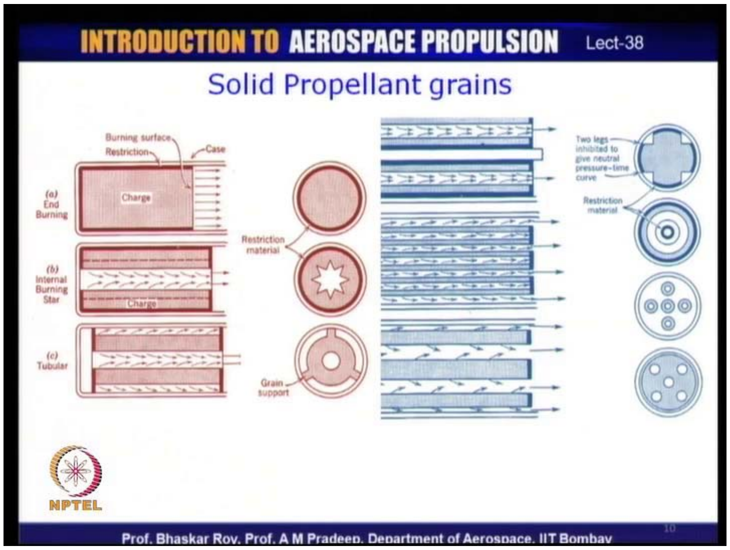


If we look at the grains that we had looked at before and we can look at them again, we can see that some of these grains have been designed to create certain neutral characteristics. Most of these grains indeed have more or less neutral characteristics, and to do that, for example, some them we have seen, have these restrictions imposed on their surface areas. So, those surfaces do not burned, only certain open surfaces are allowed to burn.

And these are the methods by which certain amount of neutral burning capability can be built into the design of the rocket grain. So, these grains have been designed to give certain kind of burning characteristics and most of them are pretty close to the neutral burning characteristics.

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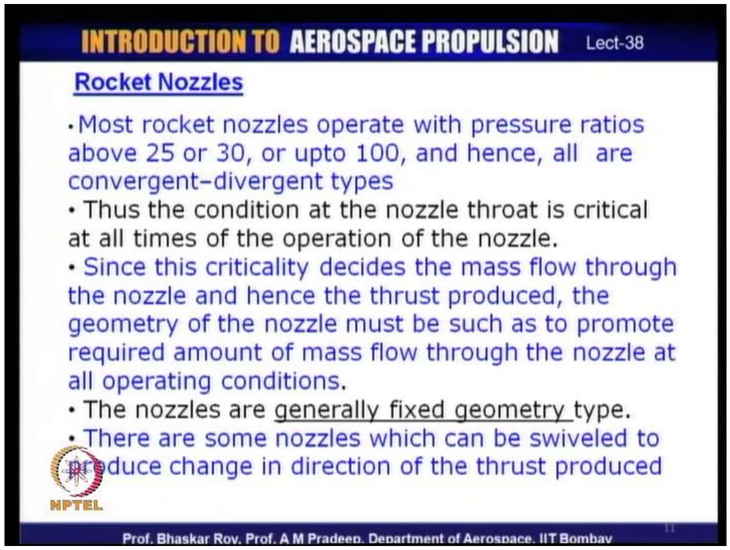


Now, we will look at what we can call the nozzle characteristics. We have been talking about the liquid propellants. We have been talking about the solid propellants. All of them finally create thrust with the help of rocket nozzles or the exhaust nozzles. Now, we have seen that most of the combustion chamber pressures that are reached are indeed very high, and the rocket nozzles normally operate with pressure ratios of the order of 25 to 30 at the minimum and some of the rocket nozzles could very well be operating with pressure ratios of the order of hundred.

And now, this indicates that the nozzle which we have operating with that kind of pressure ratio would almost invariably be convergent divergent type of nozzle. Now, this condition imposes that you have always critical flow in the nozzle throat; that means that the nozzle is always going sonic somewhere around the throat and the exhaust is almost always supersonic. Very rarely you have a rocket nozzle which is just sonic or subsonic.

So, the rocket nozzles almost invariably are supersonic exhaust created by convergent divergent nozzles. Now, the flow as we can see would always be critical. Now, this means that the mass flow through the nozzle, and hence, a thrust produced the geometry of the nozzle, etcetera, they must be such as to promote required amount of mass flow through the nozzle at all operating conditions.

Now, at all operating conditions, it is operating at maximum mass flow for that particular operating condition. Now, the criticality depends on the local temperature and pressure at the throat, and even though it is always sonic, even though the mass flow is always maximum for the particular operating condition.

Since the operating condition keeps on varying, since the back pressure, the ambient pressure at the exit of the nozzle is continuously varying and it is entirely possible that the combustion chamber pressure itself is also varying continuously; it means that the actual mass flow through the nozzle would vary even though it is critical all the time.

Now, this means you have to really optimize the nozzle design so that it caters to various kinds of mass flow, various amounts of mass flow which could be varying substantially during the operation of the rocket chamber. Now, normally these nozzles are fixed geometry, which means what we were talking about just now that we have to cater to variable mass flows under various operating condition. This cannot be done by variable geometry nozzle, not the way we have, for example, not that we can have variable geometry nozzle aircraft jet engines.

That kind of variable geometry nozzle is normally not used in rockets, and one of the main reasons is that in a rocket nozzles operating at huge pressures and under very high temperatures. At that high pressure and temperature, it is not possible to have a variable geometry nozzle. So, almost, all of them are fixed geometry nozzle. We shall see later on that some of these nozzles can be swiveled to produce change in direction of the thrust, and as a result of which, one can produce certain amount of variation of the thrust direction to change the course of the nozzle of the rocket body. So, some of those we will see in a few minutes from now.

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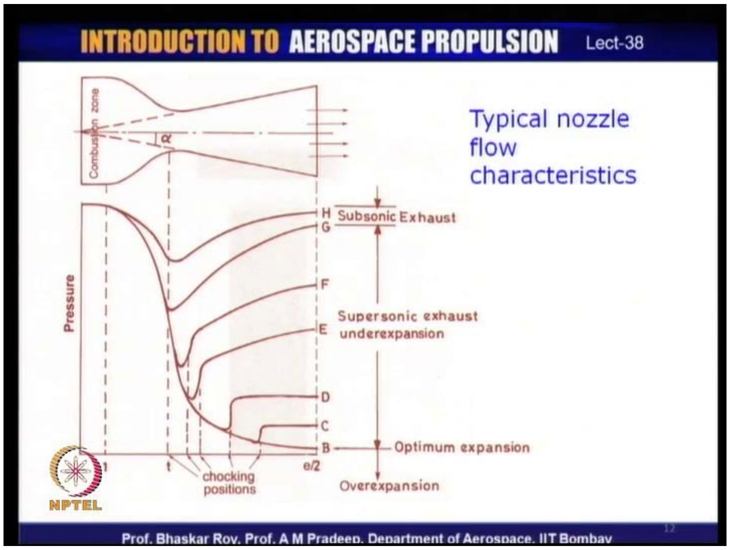


If we look at fundamental issues connected to the nozzle, if we go back and see, we shall see that a nozzle we are taking here, let us say very simple conical nozzle, most of the rocket nozzles are actually somewhat bell shaped, but we will come to that in a few minutes. The flow through them are always critical, and as we can see here, this is the throat, and at the throat, the flow if it goes critical, it will indeed continue to drop in pressure and it will reach the back pressure somewhere along here depending on where the back pressure is.

So, let us here, b c d e f essentially indicate the status of the back pressure. The y axis here is pressure; so, this is the status of the back pressure and depending on where the back pressure is the actual choking position would be decided, and if the back pressure is down here, that means almost near vacuum, the flow will continuously drop in pressure and flow along these pressure lines to finally reach this pressure.

On the other hand, if the back pressure is somewhere here, the flow will reach a choking condition somewhere over here, and then, reach a back pressure here. G is the lower back pressure at which we could have supersonic exhaust, and during which, the choking condition is reached here, and finally, the exhaust is at this pressure at G.

Now, this entire operation from B to G can be called supersonic under expansion and this is because the flow is not fully expanded; it is not expanding to the vacuum or the actual ambient pressure which could be somewhere down here. If it reaches B which is an exact ambient pressure, then we could say that it is reaching an optimum expansion or full expansion.

If, however it reaches a pressure even lower than B that is lower than the ambient pressure, then we say that it is gone into over expansion. There is a possibility that the back pressure could be so high that the flow is unable to actually reach. It is almost of the same level as combustion chamber pressure and the flow would simply flow through the convergent part of the nozzle, and then, again, go up in pressure.

And hence, this pressure zones between G and H would actually create subsonic exhaust. It is most unlikely that, that would ever happen but it is a theoretical possibility. In which case, obviously there is no point in having a divergent part of the nozzle. One could cut of the nozzle right here and create reasonable exhaust jet which would give some amount of thrust. So, the exhaust nozzle actually is dependent on the ambient pressure. It is dependent on the combustion chamber, and between these two, you have to create the nozzle shape. Now, this nozzle shape is an important issue in creation of the thrust and we shall discuss how this nozzle shape is now created.

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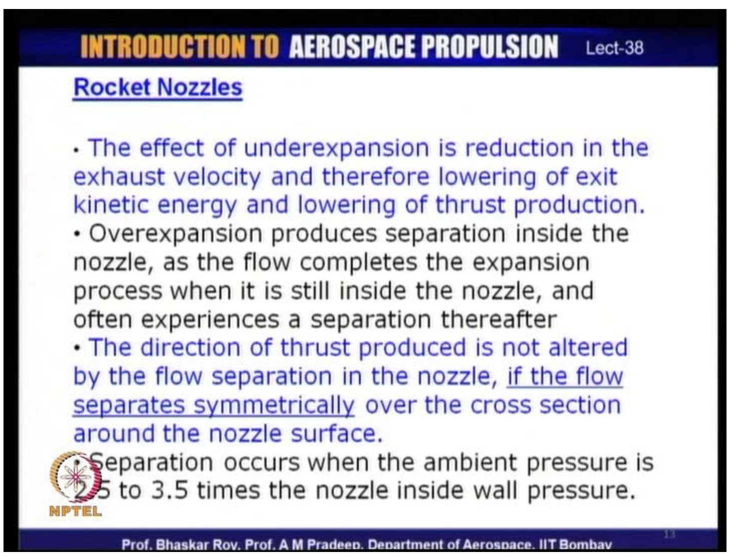

You see the effect of under expansion is reduction in the exhaust velocity, and therefore, the lowering of the exit kinetic energy and lowering of the thrust production. Now, this is something which we have to be really worried about that, if you have under expansion, the thrust is actually going to go down, and as a result of which, your rocket performance will be affected. On the other hand, if you look at over expansion, it could actually lead to situation where you have a separation inside the nozzle, because the flow has already reached the ambient pressure inside the body of the nozzle.

It cannot expand anymore, pressure cannot go down any more, and hence, the nozzle would actually show a separation tendency inside the body of the nozzle and this is something which again is a bother some affair that the nozzle would be highly underutilized and it could create certain amount of problem. For example, the separation is a symmetrical separation. The thrust produced would be affected but the direction of the thrust produced would not be affected. On the other hand, if you have asymmetric separation, what could happen is the thrust produced could be essentially asymmetric.

In which case, if you have asymmetric separation around the exit of the nozzle, you could have the situation that the rocket itself would lose its trajectory or it would take a different trajectory than what it was designed for and it could go in a different direction altogether. So, this is a problem that if you have the exhaust jet coming out of the nozzle, experiencing separation inside the nozzle body, and if that separation happens to be asymmetric in nature, the thrust produced would not be only in the direction of the initial travel, but it could start producing thrust in sidewise direction. In which case, the rocket would have a sidewise motion, which means it could have a trajectory different from what it was intended for.

So, this separation is normally not a very expected or definitely preferred kind of separation. If you have separation, ensure that it is symmetric. Now, having said that it is entirely possible to deliberately create asymmetric separation to actually aid the process of rocket trajectory and we shall see in a few minutes how this is done. So, the separation inside the nozzle body can indeed be controlled to control the rocket trajectory. One of the ways of a figuring out whether you are going to have separation or not is to see whether the ambient pressure is two to three times the nozzle inside wall pressure. So, that is a ballpark figure and it could give you some idea whether you are going to have separation at all or not.

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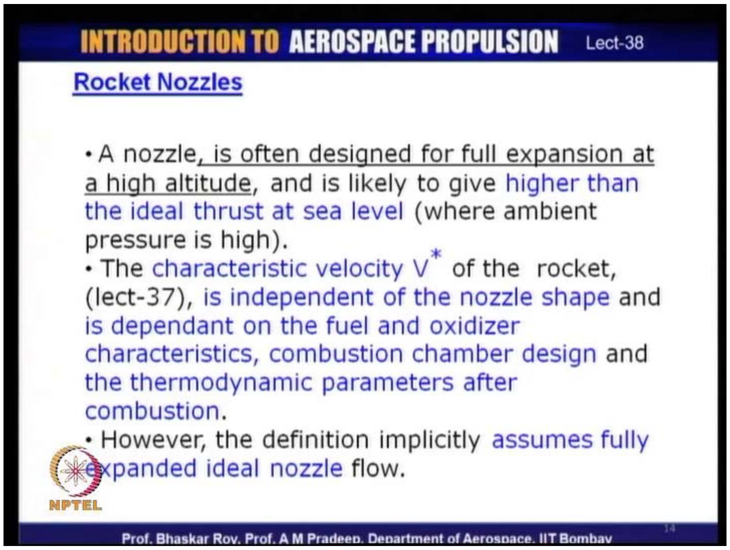


On the other hand, if the nozzle is design for a full expansion which it is often at high altitude, it is likely to give higher than ideal thrust at sea level, where the ambient pressure is actually on the higher side. So, you could have an ideal thrust calculated at sea level, but you could end up actually giving getting more thrust than you have calculated at the beginning. If it is design in such a manner, that the full expansion is actually at high altitude and it is often designed that way.

The characteristic velocity which we have defined in one of the earlier lectures is independent of the nozzle shape and is dependent on the fuel and oxidizer chemical properties, the combustion chamber design and the thermodynamic parameters after combustion. That is the temperature and pressure and gamma and other values. So, as a result of which, the characteristic velocity of the rocket is not dependent on the nozzle shape.

This definition implicitly assumes that we are talking about fully expanded ideal nozzle. So, the, whatever has been said about characteristic velocity, it is implicitly assumed that it is valid for fully expanded nozzle which is an ideal nozzle.

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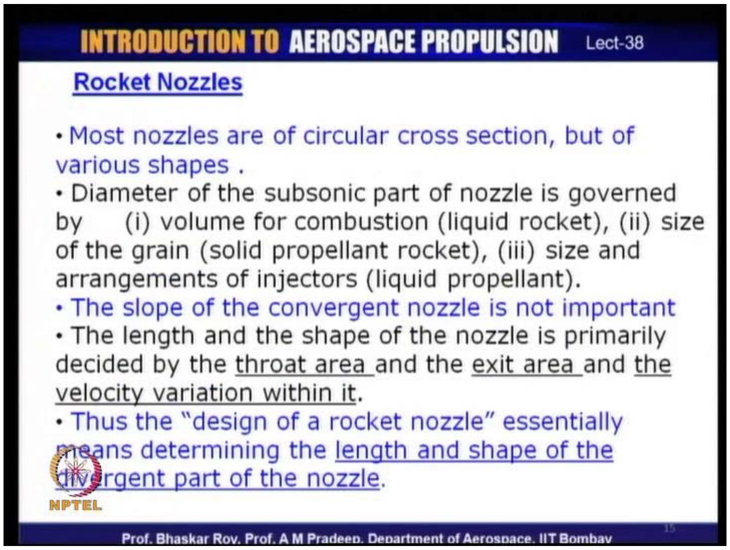


Now, in most of the nozzles that are used in rockets are circular. You will very really see a rocket cross section which is square or elliptic or rectangular or any other shape. They are almost invariably circular and one of the reasons is what we have just discussed that if you have a separation, it should be symmetric separation all around, that is, unintended separation. As I mentioned, we could have intended delivered separation, but that is a separate issue and we shall discuss that in a few minutes.

Unintended separation must always be symmetric, and for that to be symmetric, the rocket nozzle must have a circular cross section, but it could be of definite various kinds of shapes. Now, these shapes are decided by design between the throat area and the exhaust area, once you calculate, there are infinite possibilities of shapes. You have to create that shape depending on the aerodynamic and gas dynamic analysis of the rocket flow under various operating conditions.

So, the circular cross section is constant but the flow track shape could vary widely substantially from one rocket to the another. One of the parameters is the diameter of the subsonic part of the nozzle and that is governed by the volume of the combustion, because coming from the combustion chamber, size of the grain which is the combustion chamber of the solid propellant rocket.

And the size and arrangement of the injectors which is true for liquid propellant rockets; so, that means the entire combustion has to be completed before the combustion products of the gas, a properly mixed gas reaches the subsonic part of the nozzle, and hence, the subsonic nozzle has to be designed to accommodate the gas that is coming from the combustion chamber.

The slope of the convergent nozzle is on the other hand not at all important. You could have any slope of convergence. It is an expanding flow subsonic and it is not exactly a very important issue. The length and the shape of the nozzle is primarily decided 1 by the throat area 2 by the exhaust area and the velocity variation within it.

So, between the throat area and the exit area, the nozzle is shaped and this is a nozzle design which that we are talking about that the nozzle design is essentially creating the length and the shape of the divergent part of the nozzle where you remember the flow is always supersonic. So, if the supersonic nozzle design which is the most important part of the nozzle design.

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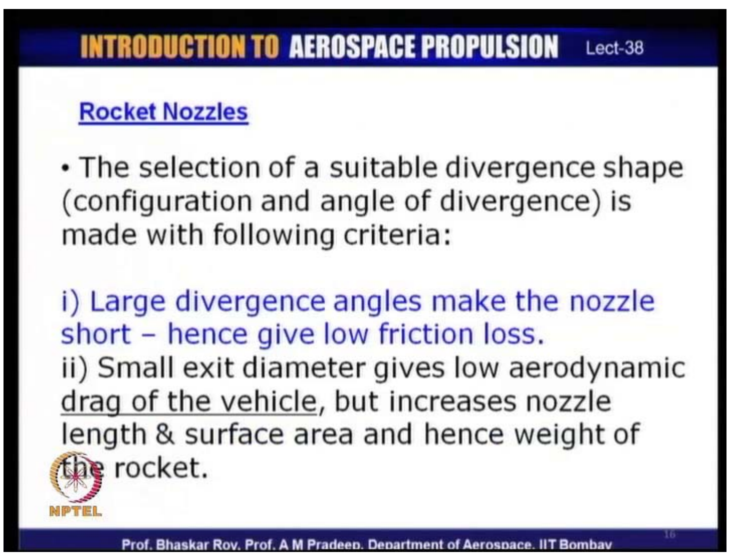


The selection of this divergent shape is can be made of certain simple criteria and we will write down here some of the simpler criteria that needs to be followed. You see the large divergence angle makes the nozzle very short, and hence, it gives very low friction loss.

On the other hand, if you give a small exit diameter, if it is not available or you want to give a small exit diameter, it gives a low aerodynamic drag of the vehicle; that means the vehicle could become thinner vehicle, a narrower vehicle, but it increase the length of the nozzle and surface area and which would increase the weight of the nozzle; it would increase the weight of the rocket engine.

Of course, it increase the surface area, which means it could increase the pressure loss through the rocket nozzle which could decrease slightly discharge coefficient of the nozzle. Nozzle being a duct has its own discharge coefficient. So, these are some of the simple issues that one needs to keep in mind in the rocket. If you have a large divergence angle, you see the exit produces a radial component of the flow. Now, if the flow is going out non-radially, the radial components of the flow will create forces. What we want is thrust going out absolutely actually so that you have an actual reaction and get an actual thrust.

If you have flow going out at an angle, it will have sidewise component of thrust. Now, if you have a symmetrical body of the rocket exhaust area, those are symmetrically distributed. Hence, they would cancel each other. However, those forces in terms of energy are lost and they are not available for thrust production. So, a lot of energy is expended in creating a sidewise thrust all around. May be they will not affect the motion of the rocket but the energy is being lost in creating those forces, and hence, those are non thrust producing components which are of no use for as far as the energy usage is concerned.

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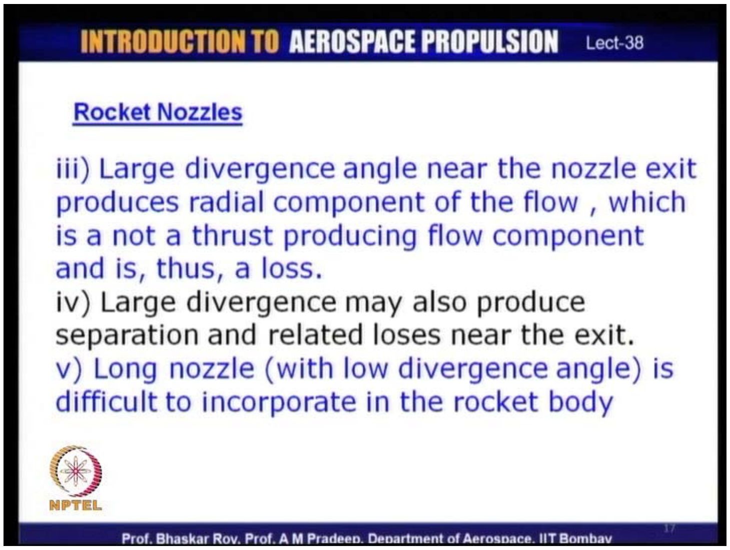


So, the large divergence angle is not exactly a very preferred nozzle shape. The large divergent has another issue and that is the flow under certain cases of over expansion. When it reaches very close to over expansion could produce separation and the separation would immediately create loses near the exit.

Now, remember, this separation is in supersonic flow. So, we are talking about supersonic flow separation. So, large divergence could produce a supersonic flow separation creating shocks and other issues over there inside the body of the rocket nozzle and something nobody would like to see happening inside the body of the rocket nozzle. It will affect the production of thrust; it will affect the trajectory of the rocket, the motion of the rocket itself. So, we do not want any of those things actually happening inside the nozzle body.

There long nozzle, very long nozzle is often difficult incorporating the body of the rocket and you would not like to have a very long. So, you do not want a very large divergence angle. You also do not want a very long nozzle, and as a result of which, most of the rocket nozzles often have some kind of a bell shape and this bell shape is to be defined exact curvature is to be designed by the rocket designer, the nozzle designer, and this is what the nozzle design is indeed all about.

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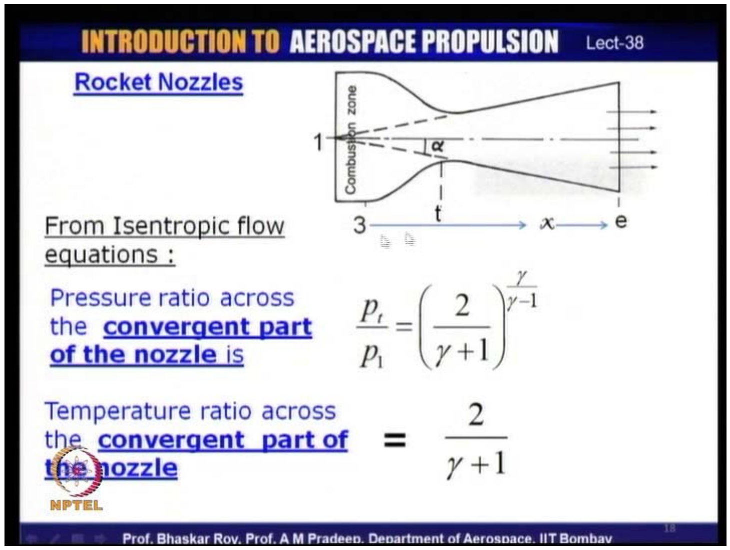


Let us quickly look at some of the nozzle design issues. If you look at the same diagram which we had looked at before, and suppose, this is your throat, this is your exhaust area and the distance from here is taken as x 3 is where you start off and one is the combustion chamber zone. So, the pressure ratio across the convergent part of the nozzle is given simply by the isentropic flow equations which you have done before in your thermodynamic courses done by professor Pradeep and this ratio is available for criticality.

Remember, flow is always critical in the nozzle. This is the temperature ratio across the convergent part of the nozzle again from the isentropic flow equations. We are assuming flow is isentropic and the flow gas is ideal gas.

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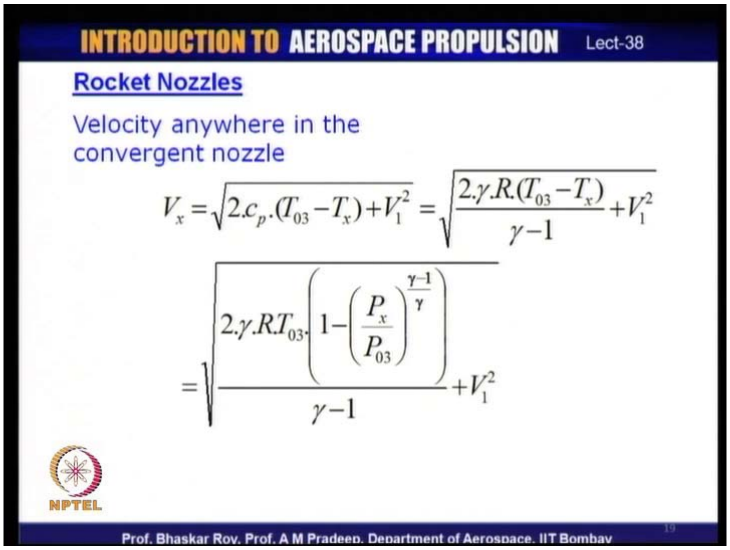


The velocity anywhere in the convergent nozzle can then be given by simple again isentropic flow relations and you could get them in terms of the basic parameters of pressure at any point which is P x and pressure coming from the entry to the rocket nozzle P 0 3 and then the gas values - T 0 3 and R and gamma. So, these things give you the velocity anywhere in the convergent nozzle.

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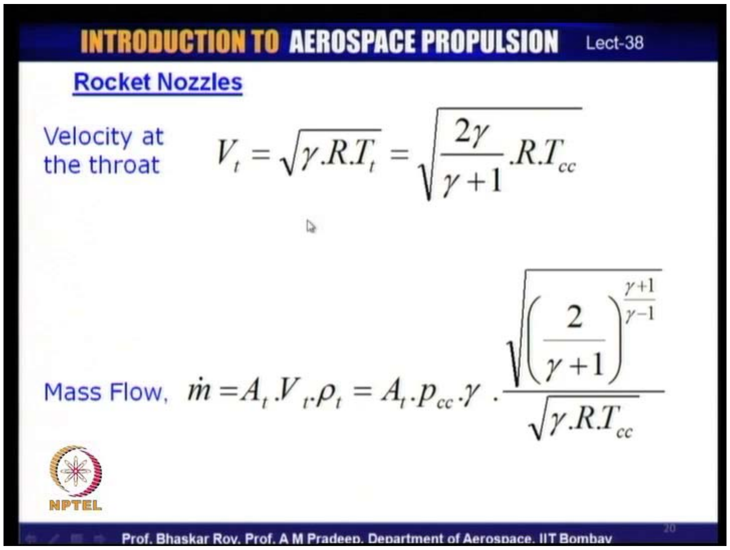

If you look at the velocity at the throats, that comes from your critical condition and you can related it to the combustion chamber temperature which is convenient, because finding out the throat temperature could be a difficult issue but combustion chamber temperature is often predictable and you can predict what the velocity at the throat would be under certain operating condition, that would give us the mass flow. That is the throat area is known; the velocity is known, and if the density can be computed, you can get the mass flow calculated for the flow that is flowing through the nozzle that is from your continuity condition.

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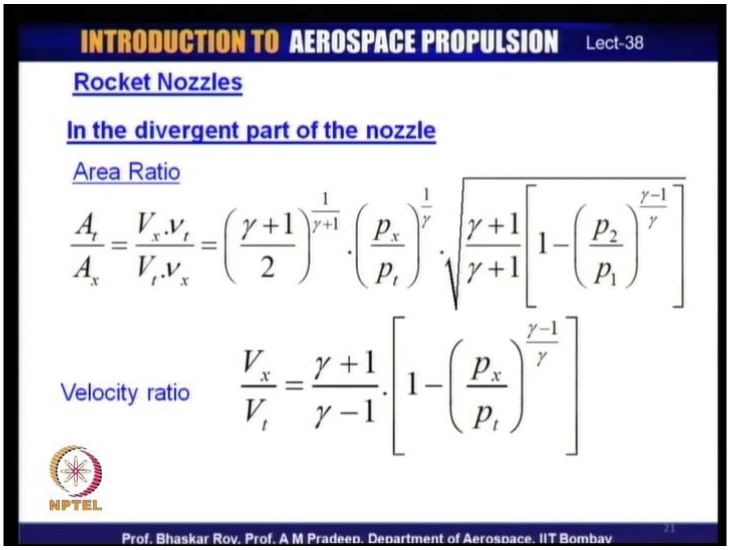


Now, if you continue to go into the divergent part of the nozzle, the area ratio between the throat and anywhere in the divergent part can be again found from using the isentropic relations. As I mentioned, we are continued to use isentropic relation. Assuming that the flow is indeed isentropic and it gives you these relations again in terms of the fundamental gas dynamic parameters. The velocity ratio at any point from the throat can also be found out from simple isentropic relations using the gas constant and the pressures at throat and at any operating point of the nozzle.

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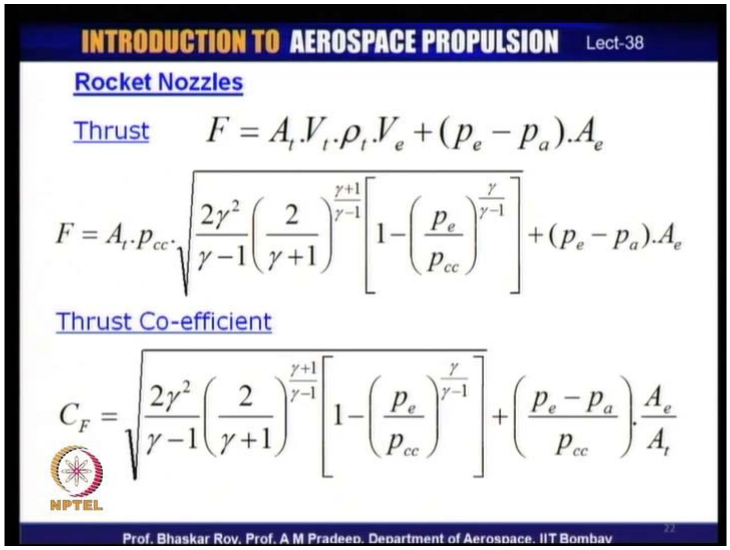


Thus the rocket nozzle thrust can be found out by looking at the thrust value that we have looked at before. The first one is the momentum thrust; the last one is the pressure thrust and this could be written in terms of the parameters that we were talking about, the nozzle parameters that we were talking about. So, the first part would be essentially dependent on the nozzle design and the nozzle parameters. The second part is the residual pressure, that is going out with and that gives certain amount of residual pressure thrust. The thrust coefficient that we have defined before can now be written down in terms of the simple gas dynamic parameters, that we are able to generate from the rocket nozzle operation.

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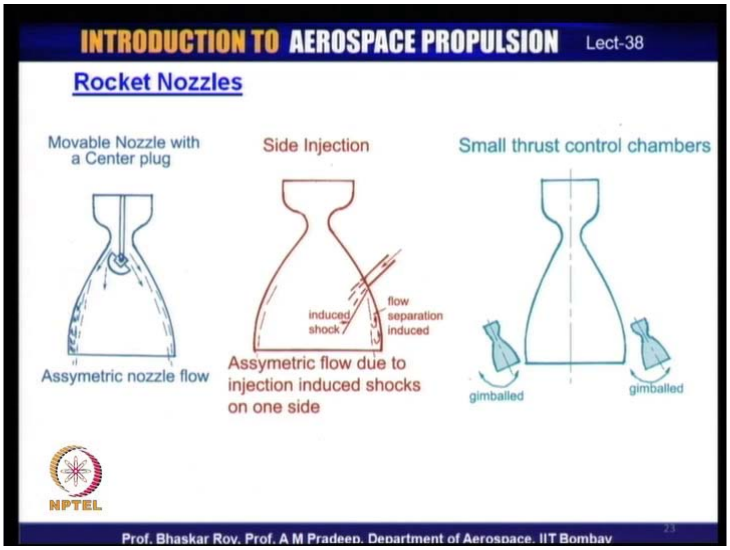


Now, if you see that the rocket nozzles could be of various kinds. Now, the equations that we have used or written down just now can be for any of these nozzles, what can be done is if you want to have some control over, what is happening in the nozzle? As I mentioned earlier, you could actually create an asymmetry inside the nozzle by creating a plug.

If you have a center plug and just move the central plug and it could asymmetry flow of one side compared to the other side and this could give you a sidewise thrust which could give the rocket a sidewise motion or a trajectory, which may be designed for the particular rocket machine. Similarly, you could have a side injection, somewhat small injection of some gas could create an asymmetry and this asymmetry could again induce shock.

Remember, the flow here is already supersonic and it could induce shocks on one side and this could create asymmetry to create a sidewise thrust for the rocket to change its course, but one of the most issued used ones is the gimballed small thrust rocket nozzles, which are the small side rocket nozzles and these ones can be gimballed.

So, as a result of the gimballing, you can create small amounts of thrust to take the rocket vehicle in one direction or the other and this is extremely useful whether you are in atmosphere or whether you are outside the atmosphere where a very small amount of thrust could indeed actually give you a certain amount of sidewise motion or a motion control of the entire rocket vehicle or the space craft, ballistic missile that you may have. Now, these are the ways by which you can use a rocket nozzle indeed to control the motion of the rocket.

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I will leave you with a picture of a rocket launch. We have discussed so many kinds of rockets. This is one of the biggest rockets probably ever created and this rocket launch essentially shows that for a huge big rocket to be launched, you need a big structure to hold it in vertical position. And before the rocket is launched, this is to be strapped there, and only when the rocket is launched, when the gas comes out from the bottom, these fixings are released and the rocket lifts off.

So, you need a huge structure which is a part of the launch pad and this is a launch pad on which the rocket is actually strapped till it actually lifts off. So, all rockets are mounted on a launch pad like this. The gas comes out from the bottom and it flows sidewise to create a soft lift off, and during that soft slow lift off, these fixings are released automatically and the rocket is released to go up vertically. And this structure may remain or it may actually fall off. So, this is a huge structure that is built for this Saturn rocket launch which was used for many of the Apollo missions that were conducted by U S quite some time back.

So, we have been discussing rockets, spacecraft's, missiles and we see that the fundamental issues are of the same order and we have just a discussed some very fundamental sciences that govern these crafts.

I hope it will be possible for you to go into other courses which take you to higher levels of rocketry and higher levels of rocket science, but till such time, I hope you would be able to look at any rocket today and see whether you understand how the rocket functions and how that particular rocket is actually used for any particular purpose. In the next lecture, we shall close down this lecture series on introduction to aerospace propulsion and we shall recount some of the things that we have done and we will be looking forward to what could possibly be the future of aerospace propulsion.