UAV Design - Part II

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Lecture - 05

Relation Between Aerodynamic Center and Center of Pressure

Yeah, hello all. Welcome back. In our previous lecture, we discussed about how aerodynamic forces namely lift and drag are produced on an aerofoil and we have defined what is lift and drag.

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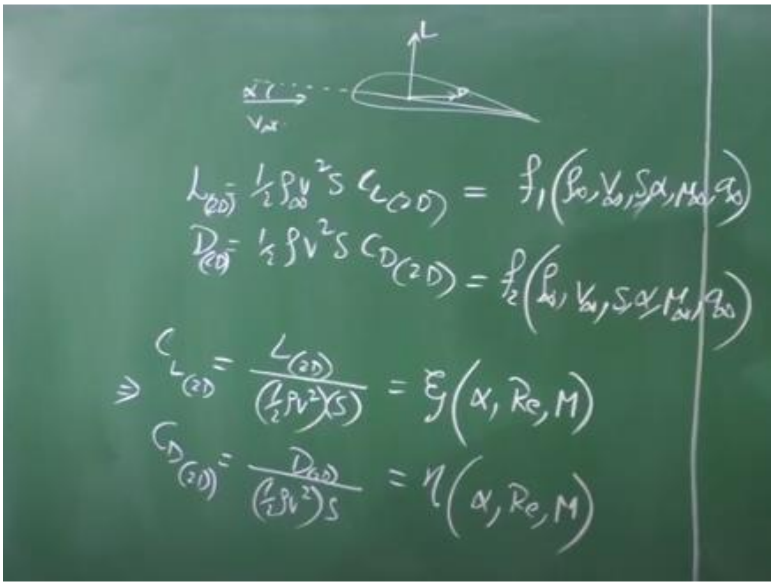

So let us consider an airfoil. So when there is flow, so the orientation of this airfoil with respect to flow is the angle of attack represented by alpha and lift is a component of the force, resultant aerodynamic force which is acting perpendicular to free stream and say drag is a other component of this force acting along V infinity. And we have defined lift is equals to half rho v square s C L right.

So and at the same time we have also defined what is drag, half rho v square s C D. So let us and this is 2D case right. So the corresponding lift coefficient and drag coefficient for this aerofoil is considered as 2D, why because aerofoil is also known as infinite wing. It is a sectional property right. So this lift and drag we are talking about is about a sectional property here, is it not.

So the section here is a two dimensional section of the wing, which is aerofoil also known as infinite wing or 2D wing, right. So if we look at the C L, C L 2D what we can write it as lift generated upon half rho dynamic pressure times the reference area. At the same time drag can be given by this expression, right.

So if you observe the C L, so in the first place lift is a function of say density right, rho infinity and velocity free stream velocity here and then size of the lifting surface as well as angle of attack and also dynamic viscosity as well as speed of sound, right. Where mu infinity is a dynamic viscosity of air, we will just define it very soon. So and f 2, let f 2 be the drag is a function of this same variables here, right.

So for example, if I have to deal in terms of lift, I need to if I have to understand the variation of lift, then I have to vary all these parameters and see how the lift is how the corresponding lift variation is with respect to the density, with respect to V infinity, with respect to surface area as well as angle of attack, dynamic viscosity and sound. So there are speed of sound here yeah.

So there are quite a lot of variables. So instead of dealing so many variables what we do is let us talk in terms of this non-dimensional parameters right. So let us say this is a function of angle of attack, Reynolds number and Mach number, right. Similarly, drag coefficient is a function of angle of attack, Reynolds number and Mach number.

So whatever we are talking about these coefficients here so the coefficients here we are talking about is two dimensional coefficients right. So not for the entire aircraft. You are talking about airfoil character yeah non-dimensional coefficients for airfoil here. Now apart from reducing the workload, let us say how to generate the C L with alpha in the first place, right. How do we generally get lift variation with angle of attack?

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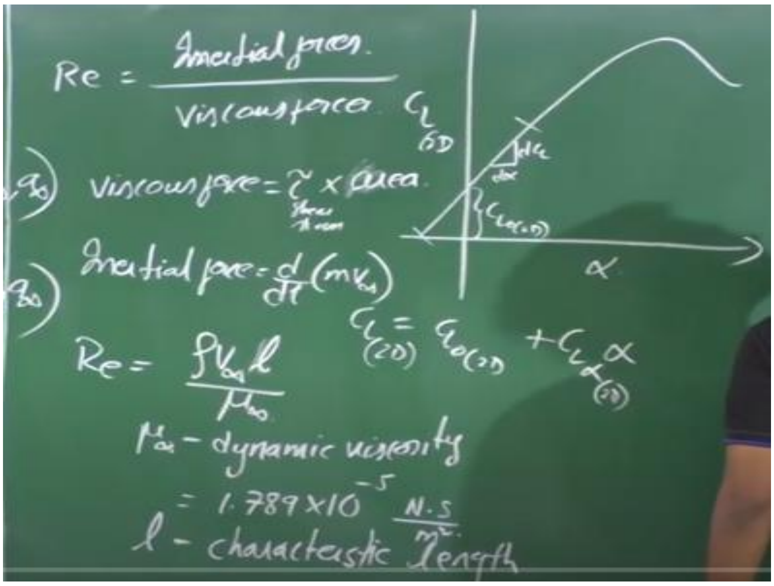

So previously we looked at how C L varies with angle of attack right. So for example, this is for aerofoil and within the linear regime we assume a straight line equation, right. So we can represent this variation of lift in the linear regime as C L naught plus lift curve slope which is C L alpha times alpha. So 2D, right. So how have we generated this data, how have we got this information?

By testing this airfoil in a wind tunnel, right? Wind tunnel is a setup, experimental setup where we can drive the air at the desired speed and test our airfoil and place our aerofoil inside the wing. So which is equivalent to airfoil moving at the same velocity in the static air, right. So we use that principle. So we place aerofoil inside that wing and try to measure the corresponding forces acting on it right.

So why do you want to deal in this with this parameters? One thing is if I want to know about the lift and drag characteristics of a particular aircraft, then I have lift drag characteristic of the airfoil for that matter, then I would like to then I have to vary all these parameters, say about six parameters. Say initially I have to keep density, reference area, alpha and all of the parameters constant and vary the velocity.

At different velocity what will be the corresponding variation of lift coefficient and at the same time I can keep velocity constant, rest of this parameter constant I can vary the angle of attack here right. And also for different for example, if I am using the same cross-section, but different, wings of different size let us say, right or say how should I compare.

Let us say if I am using different aerofoil here, so the lifting characteristics of the aerofoil will be for different, right. So and it changes with shape as well, right. So in order to decouple this or in order to reduce the number of variables as well as the complexity, we try to deal in terms of non-dimensional coefficients here. So let us consider a small example where a Boeing, a Dreamliner is flying at an altitude of 30,000 feet, which is approximately 10 kilometers right.

So moving at a velocity of 0.8 Mach number, right. So first of all let us define this Reynolds number is equals to inertial forces upon viscous forces right. So this viscous force is equals to so this shear stress times the corresponding area, reference area right times the area. And inertial force is due to rate of change of linear momentum right.

So if you solve this what you have Reynolds number as rho V l where capital so rho v l upon mu infinity right where mu is the dynamic viscosity. So for air it is at 18 degrees Celsius. It is about 1.789 times 10 raised to the power of – 5 Newton second upon meter square, right. And then l is a characteristic length. So when you talk about a aerofoil, you can consider the characteristic length to be the chord of the airfoil, which is the length of chord line, right.

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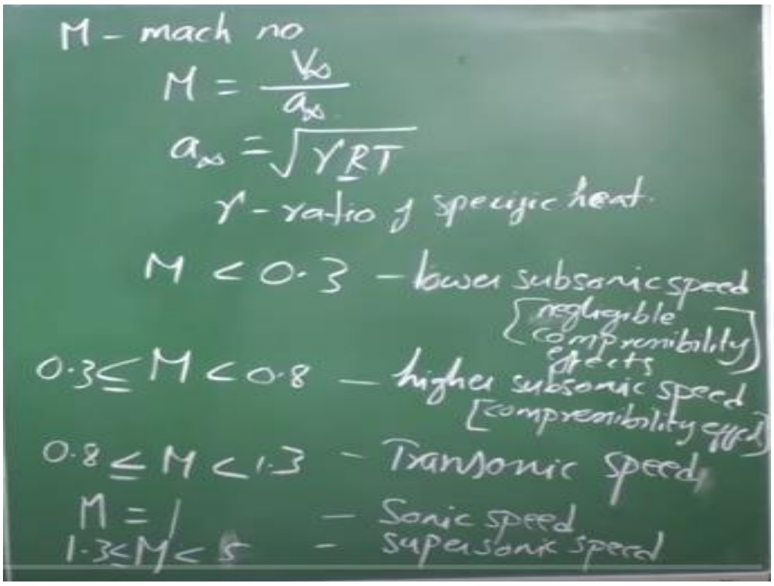

And similarly, the Mach number here, M is Mach number. So Mach number is defined as velocity of flight vehicle upon velocity of sound where a infinity is a velocity of sound which is square root of gamma RT right, where gamma is a ratio of specific heat which is 1.44 air right. And then R is the universal gas constant and T is the temperature, corresponding temperature ambient temperature.

So this talks about how fast we are moving, when comparison to time when comparison to sound speed of sound here, right. So if you are moving at a Mach number which is less than 0.3, we call it as low subsonic speed. So this is like negligible compressibility effect, right. So when you talk about when you are moving anything less than 0.8 Mach number and in between 0.3 Mach number.

So this particular flow, this particular velocity corresponds to a higher subsonic Mach number, higher subsonic speed which is where the flow is dominated by compressibility effects, right. So if you are flying in between 0.8 and 1.3 this particular regime is known as transonic flight regime or transonic speed, right. And the flow is, if the Mach number is equals to 0.1 sorry 1, so that means you are flying at sonic speed.

So effectively you are flying at velocity of sound. So you will cover about say close to 1200 kilometers per an hour, right. So 5 is supersonic. So the flow is mainly dominated by shocks right, shockwaves.

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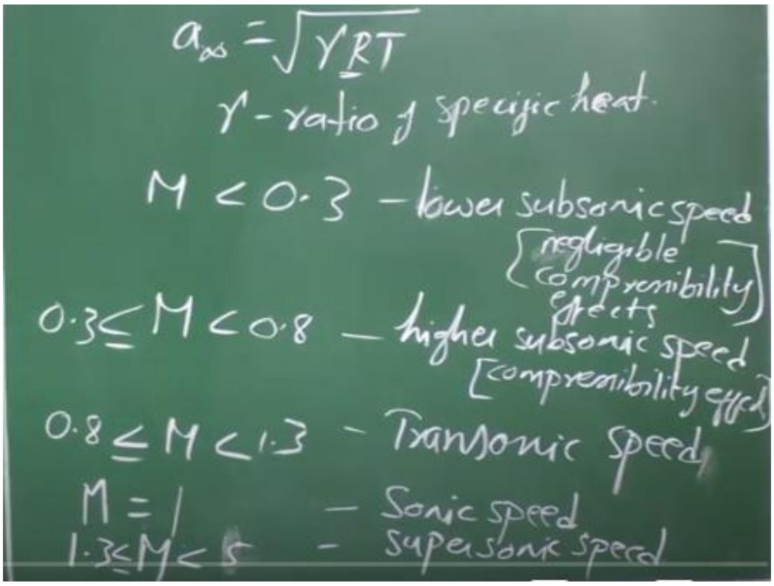

And then so the Mach number which is greater than 5 is hypersonic speed right. So where the concepts about rarefied gas dynamics will come into the picture, okay. So yeah now coming back to this discussion, so now we want to look into this non-dimensional aerodynamic parameters, which indeed a function of this non-dimensional variables right. So here Reynolds number is rho V l by mu is a non-dimensional variable.

And then Mach number is a non-dimensional variable here. So instead of handling six variables to find out the forces, we can able to find out their force coefficients with three variables. So apart from that, I will just get back to that example what we were discussing. So we have considered an Airbus dream sorry, Boeing Dreamliner, right 7879 or 8, whatever. So it is flying at an altitude of 30,000 feet which is approximately 10 kilometers right.

So and it is flying at a Mach number of 0.80. Now whatever the lift that is experienced by that aircraft, so in the first place, how can we allow that aircraft to fly, right. Can you dare to allow it to fly in like straightaway after manufacturing? You not you first have to simulate the conditions is it not, simulate on ground. So how do you simulate those conditions, similar conditions is by performing wind tunnel test.

So as I discussed wind tunnel is an experimental setup where you can place your scaled down models or place your models right inside the test section, where test section is the place where you actually test measure the data from the model right that it experiences from the wind and the tunnel is equipped with a fan which runs the air at a desired speed right. So fan will basically try to throw air at a desired speed.

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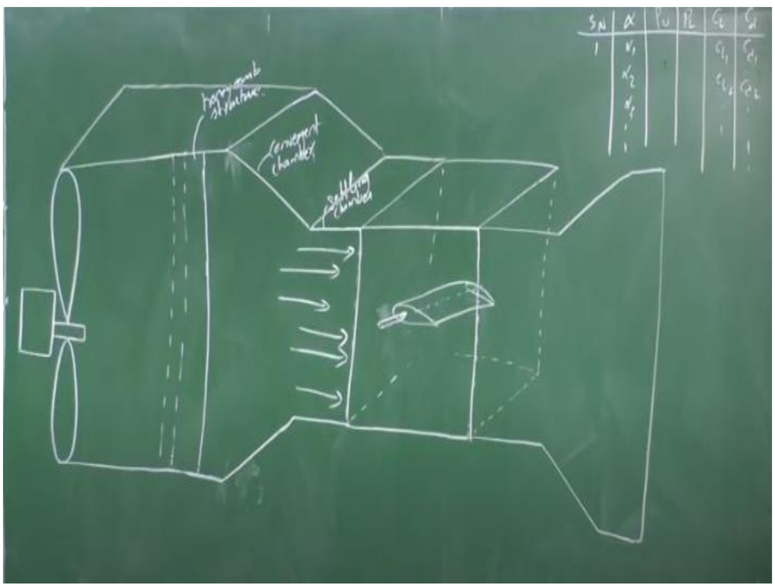

So I may not be drawing it to the scale right. So and then you have some filtering chambers, right. And then there will be a so say there is a honeycomb structure, it is not an obstruction here right. So this there can there is something called honeycomb structure to streamline the flow, right. And then there will be convergent comb or convergent chamber right.

Followed by this we will have a settling chamber where the flow now tries to settle in this particular chamber. Chamber, convergent chamber, right. And then you have your, you have the test section right. So this is where we generally place our models and perform the testing. So let us say this is the test section. So followed by a you will have a diverging chamber, right.

Okay, fine. So this particular setup runs the air at desired speed, okay. Now this motor will try we can vary the rpm of this so that the air inside will run at different forward velocities that is the aircraft faces right. So now, how should I generate this two dimensional data here, right. So what can I do is let us say this is my like side wall of this test section, one side vertical wall.

And then let us assume this is the vertical wall on the other side. So this is my front face and this is my back face right. So now let us consider an airfoil right and extrude this airfoil. What you have is a wing right. Now so extrude this aerofoil till the other face of this test section okay. So what happens by doing that is the flow anyways we know that it is always parallel to the test section here inside the wind tunnel, right.

The flow will remain always parallel to the test section here. So this is how the wind tunnel will be, right. So okay now what I can do is if I take a plane right which is in between these two, which is exactly the midway of the span of this airfoil right. And if I can tap, if I can put holes on the surface and then tap the corresponding pressure at each and every location, right.

I will be able to measure the pressure distribution over an airfoil at a given location right, is it not? So I have extended this aerofoil till the ends of this test section, walls of ends of the walls of this test section. So effect what I am trying to do is with respect to flow, it is like a continuous body, it is an infinite body, right. The flow happens within this confined regime, which is inside the test section right.

So till the ends of the walls we have we have covered it with the airfoils, n number of aerofoils you can issue. So there is an airfoil here as well. So this particular, you can imagine this as an extrusion of this particular airfoil say this, say if this is not to the scale, then I can reduce it a bit, right. So everywhere we have airfoil sections, right. So in the midway of this length at the midpoint of this length, let us take a vertical plane.

And in that plane, let us make holes on the surface of this object, right? So by tapping the pressure from the surface of this object, I will be able to find out pressure distribution across this aerofoil, on the top and bottom surface. By integrating that, I will be able to figure out what is lift and what is drag, okay. Now that is how we get lift and drag.

Now coming back to our example again, so if I have to test the wind, test this Dreamliner inside a wind tunnel, will I be able to do it? So I need, I need such a big test section right. So for an aerofoil we can definitely do it, but what about the Dreamliner? Can I bring it and place it inside the test section? That may not be a feasible idea, is it not?

So instead what I what can I think of is reduce it, scale it down, scale it down the dimensions and then bring it into the test section and perform the required test right. So in doing so can I generate the same lift, right. Why because if I have to generate same lift for example, the one which is flying at 10 kilometers right. So it experience certain lift right based upon its surface and the velocity of flight and the corresponding ambient conditions there right.

So can I get the same force what you call duplicated inside the wind tunnel, right? So that may be a bit difficult task, very difficult task. Instead, what I can do is I can find out what is the C L lifting characteristics at that particular lift non-dimensional lift coefficient at that particular flight conditions.

And if I can simulate the same conditions here like if I can measure the same C L by simulating the same Reynolds number what the aircraft is actually facing and then the same Mach number what the aircraft is facing in the real time. If I can simulate both Reynolds number and Mach number on a scaled down model, right, what actually the aircraft is facing, is it not or going through.

So if I can simulate the similar conditions, then I will be able to get the same lift coefficient for the actual flight as well as scaled down model inside the wind tunnel. So this similarity is, of flows is known as dynamic similarity, right. So if I can do the dynamic similarity then if I can achieve such condition dynamic similarity condition then I will be able to attain the same C L value as that of free flight value right, actual flight test value same.

Yes of course, you may have to maintain the same angle of attack right even inside the wind tunnel you need to maintain same angle. So but how do you get angle of how do you vary angle of attack here? For example, if I have a rod coming out of this aerofoil and I connect that rod to a motor, stepper motor here right, whose like the angle of stepper motor, I can control it, right. If say, there is a shaft.

So say there is a shaft running throughout this airfoil and the shaft is out here. Say if the shaft is out, say if I can rotate the shaft, this is outside the test section again right. So if I can rotate this shaft, I will be able to rotate the airfoil altogether, right the entire wing the section that we are using here. So and then, yeah, by rotating so I am trying to change the orientation of this object with respect to the free stream velocity.

Here in our case, we call it as angle of attack right. So at different angles, if I can rotate it and hold it at different angles and take the corresponding pressure readings from the pressure tapping there. So I will be able to figure out what is the pressure distribution at that particular angle of attack, and then I can find out what is the corresponding lift and drag, right lift and drag coefficient at that particular angle of attack.

So by varying the angle of attack, I will be able to generate the data, right. So for different alpha, so I am, I will change from alpha 1, alpha 2, alpha 3 so on. I will be able to generate pressure on upper and lower surface right, pressure on upper surface, pressure on lower surface. I know, physically I know the location of this ports, right.

I will be able to get pressure data from each port and I will be able to integrate them to figure out what is the lift coefficient as well as drag coefficient, right? So C L 1, C L 2 corresponds to alpha 2, right so on. Okay. That is how I will be able to generate C L versus alpha graph. So that is the advantage if you can deal in terms of non-dimensional force coefficients here.

Aerodynamic force coefficients, which are lift coefficient and drag coefficient, you will be able to right achieve this same characteristics as the that of real flight at a very less expensive and yeah, in terms of both time and money right, as well as your efforts, okay. So the main aim of the designer, wind tunnel designer is to achieve this dynamic similarity, right.

He wants to, so the ideal wind tunnel is the one which can generate the required Reynolds number as well as Mach number together. But in reality, it is very difficult. So the aim of wind tunnel testing will be either or the design is guided by either with an aim to achieve either one of these parameters, right. Either the Reynolds, desired Reynolds number or desired Mach number.

That is why in general this wind tunnel tests are performed in multiple wind tunnels. So the same model is tested in multiple wind tunnels where one wind tunnel can generate your desired Reynolds number, the other one can generate the desired Mach number and see how the characteristics of this yeah. Not average.

You want to know its variation with Reynolds number and Mach number, why do you take average there, right? It is independent, is it not these two?

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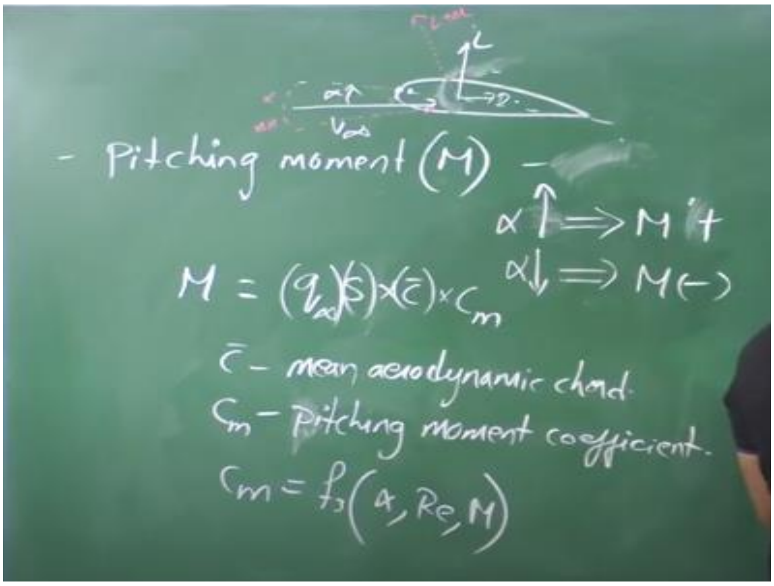

So let us get back to this figure again, where we know when there is flow. So there is lift and drag right. So if I consider any arbitrary point on this, so this flow will try because of this lift and drag or because of the lift majorly we will try to produce a moment as well, right. So this aerodynamic forces will also produce a moment when there is flow apart from the aerodynamic forces there is also called aerodynamic moment here right.

So let us consider one such moment called pitching moment. Very soon we will discuss about what are all the degrees of freedom what an aircraft have or a rigid body in space have and then we will define this yeah nomenclature in detail right, pitching moment. Let us say it is defined by M where if the aerofoil due to this moment, if the aerofoil increases its angle of attack it faces a higher angle of attack, right.

It will rotate in a direction which increases the angle of attack. Then we call it as positive, right. So alpha if it is the increasing direction of angle of attack, then we will call it as this pitching moment is positive pitching moment, which is also known as nose-up, right. So if the moment reduces the angle of attack we will call negative angle of, a negative pitching moment or pitch down moment, okay.

Pitch down of this aerofoil. So this is a convention that we have adapted right. Now let us assume yeah, so yeah, first let us define this pitching moment, which is the dynamic pressure times or Q infinity times the reference area will help us with the, so this quantity corresponds to the force, aerodynamic force right times the reference length here, gives you the moment here.

So multiplied by this is the non-dimensional moment coefficient, right where C bar is mean aerodynamic chord. So we will discuss about the C bar when we talk about that wing planform. And C m is non-dimensional or pitching moment coefficient which is non-dimensional similar to that of C D and C L. Yes and C m you can assume it, it is also a function of alpha, Reynolds number and Mach number.

Understanding the C m is also very important right in order to characterize a particular aerofoil as well as aircraft, right.

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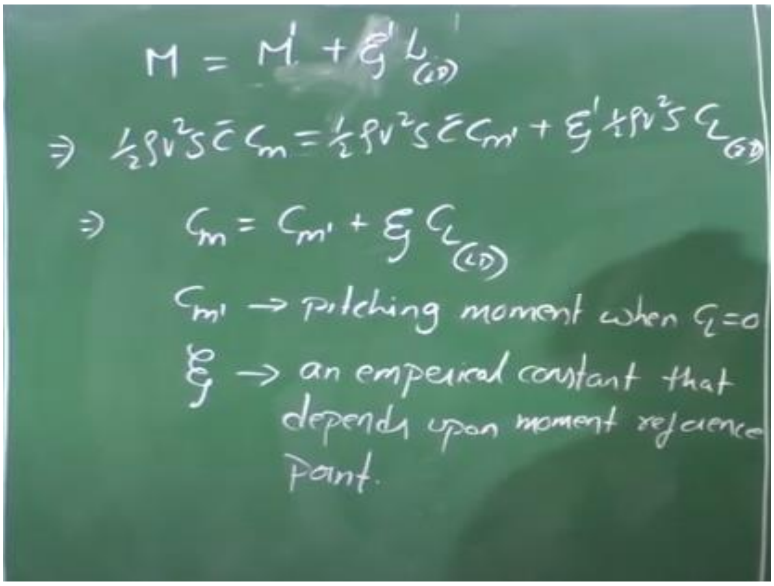

Now, let us say this moment about a point here. Let us write a moment about a point arbitrary point here is equals to moment let us talk about this moment which is independent of lift right. So independent, let that be M bar right, M prime is independent of lift plus some factor times lift. So this factor depends upon moment reference point, let us say okay.

So in non-dimensional form or say right, in non-dimensional form what I can mention this as is so this is half rho V square S C bar times C m is equals to half rho V square S C bar times C m prime plus zeta times half rho V square S times C L. This is 2D again right. So C L 2D. So until I mention this as an aircraft or wing this is always corresponds to 2D case, right.

So ideally I might have used the small case letter of l but I am used to this capital L 2d. So that is the reason why I am continuing. So if let us say if I miss at some place, you kindly consider this as a 2D case. Even C m is again the 2D case here we are talking about. So C m is equals to C m prime plus some factor zeta times C L 2D, right. Zeta prime upon those variables will be say zeta.

So where C m prime is pitching moment when C L is 0, okay. We will see whether such thing exist or not. And then what is the significance of the C m prime. And say zeta be the be an empirical constant that depends upon moment reference point, okay. Now, so what is this C m prime? What do you mean by that? What is the significance of the C m prime?

So let us say we said this is when pitching when lift is zero right. Am I correct or not? So we have discussed about symmetric as well as cambered aerofoils earlier, is it not? So for a symmetric aerofoil, when lift is zero for a symmetric aerofoil? When angle of attack is zero, right at zero angle of attack.

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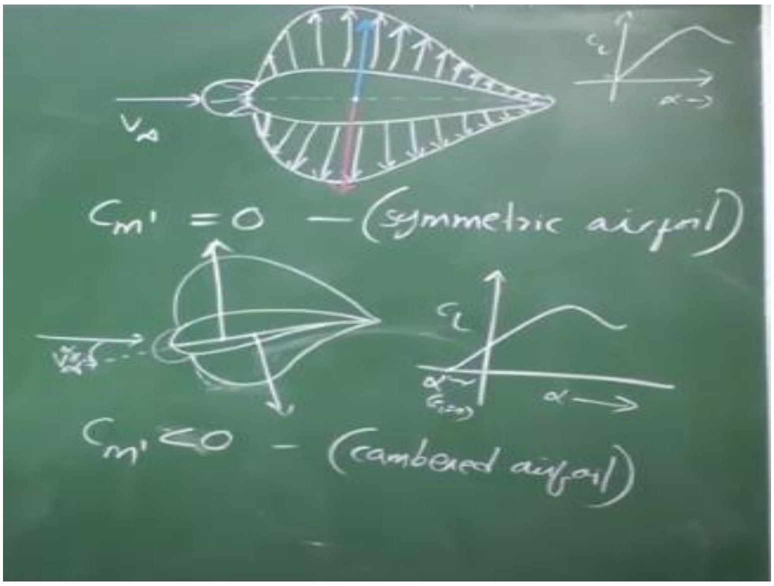

So what happens at zero angle of attack for a symmetric aerofoil? Why lift is zero? Right we said if you remember, so this is how the variation is right for C L and alpha. So at alpha is 0 C L is also 0 here right? What do you mean C L is 0? Lift is 0, right. So when can you achieve such condition? What do you think is happening? What do you think is happening?

When can we have such condition lift is zero? Oh, again yeah you can simply say that, when the body is not moving, right. But here we said at angle of attack zero, which means so we are defining an orientation with respect to flow. That means the body is moving alright it is moving ahead. So this is V infinity. So there is flow. But still there is zero lift. When that can happen?

So we know that there is pressure distribution when there is flow across this aerofoil. There is pressure distribution on top and bottom surface. So the pressure distribution on the top surface may result in a. So this pressure distribution of top surface, just forget about the boredom face right now. Let us talk about only this pressure distribution on top surface. So what happens here?

So there will be a resultant pressure on the top surface or there will be a resultant force because of this pressure distribution acting on this top surface is it not, top surface alone right? So say it may act at the centroid of this distribution, maybe. So that can be a easy guess right? It may be acting at the centroid of this pressure distribution on the top surface, right.

If there is an equal force that is acting in the opposite direction, right. So if there is an equal force which is acting on the, so there is a pressure distribution on the bottom face as well and then, so there will be a point about which there is a resultant force acting, is it not? Right, so let us say this resultant force is acting at the centroid on the top face, right. So this is the resultant force because of the pressure.

At the same time, if these two forces and there is a resultant force on the bottom side, so if the magnitude of these two forces are equal right, then lift is zero, that is when C L is zero here. Am I correct or not? So there is equal and opposite forces acting. So again airfoil is a rigid body. So for a rigid body, if I apply, if I am lifting this body with certain force F acting at this point, right, I am lifting this body.

That means there is some force acting on this. So the same force acts on each and every particle that is rigidly attached with this body right. So entire body will experience. So similarly, I am just transferring the force that may be acting on the surface just to undo the chord line here, right. Similarly, the bottom the resultant force acting on the bottom face is also transferred to the chord line here right.

Now when the top and bottom forces are equal for a symmetric airfoil, it happens that so these two act at the same point right. So when lift is zero, when these two are acting at the same point, that means there is no moment as well. So this moment when lift is zero, for a symmetric aerofoil is zero right. So in case of cambered aerofoil you have what should be the condition?

When can be lift, when lift is zero for a cambered aerofoil? Yes, you have to, so for a cambered aerofoil so I need to orientate a negative angle of attack. Alpha at which C L is zero is negative, right. This is alpha, this is C L. So this is my negative angle of attack at which I have to orient my aerofoil, this cambered airfoil with respect to flow, so that lift is zero. But in that case what happens is this is that alpha at which C L is equals to zero.

So though the lift distribution, the pressure distribution on the top and bottom they are equal in magnitude, but may not be acting at the same point, right. They may be acting at an offset point, right. So for a cambered aerofoil because of which there is a pure couple it is a independent of moment reference point here, right. So it is like pure couple, right.

So it is generally less than zero for a cambered airfoil, positively cambered airfoil, right. So let us get back to this equation. If you can yeah we now are clear with what is C m prime, right? It is moment when lift independent of lift here right and is equals to zero for symmetric airfoil and also discussed that it is negative for cambered aerofoil, right. So what about this zeta?

So let us consider a moment reference point towards close to the leading edge here right. So you know, when there is at certain angle of attack right there is some lift already acting on this as well as drag, right. And because of this lift and also now we are in a position to represent what is this moment in okay, let us not do that for the time being, right.

So because of this, when you change the angle of attack, right, so when you change this angle of attack the reason so let us say that angle of attack changes positive and there will be an incremental lift here right acting. So this incremental lift will try to produce a pitched on moment right. So this is negative why because there will be an increase in angle of attack here.

So let us say if I increase this angle of attack alpha plus delta alpha right. So because of which there is an increase in lift right, there is an incremental lift. That incremental lift will produce a pitched on moment here right. So if this negative moment, is it not? So the contribution of lift towards this moment is negative that means this particular factor is the moment factor right is it not?

So this particular factor has to be negative, which forces data has to be negative right.

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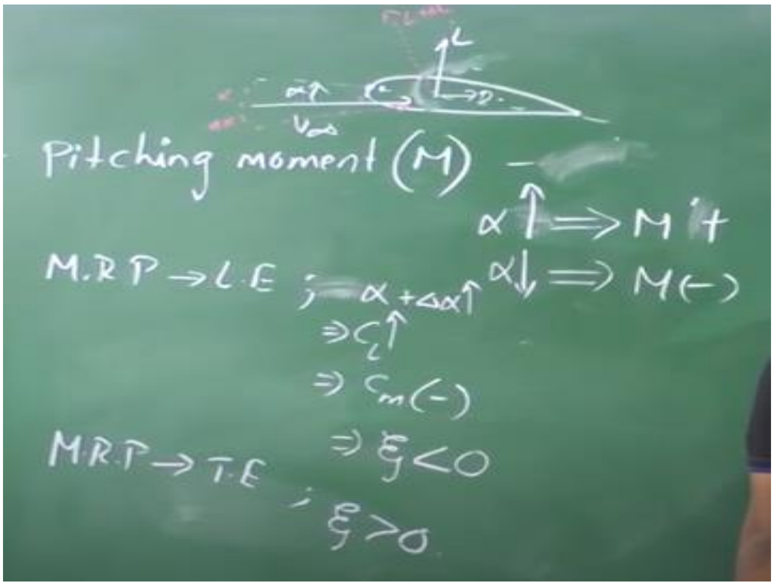

In that case. So if I consider, so moment reference point, so at leading edge, so this forces so when okay, so when there is an incremental alpha like right, so alpha is positive, so C L will be positive right. So which implies C L will also increase. This increase in C L will produce negative moment, right. So how this is possible? So this since, so this condition forces zeta has to this implies zeta has to be less than zero, right.

Let us consider this moment reference point at the trailing edge. And let us say we shift our focus to this trailing edge right now, and with respect to that point, so the any change in the lift right positive change in angle of attack will produce lift. So that lift will try to give a pitch up moment is it not, that gives a positive value of C m here is it not? So this C L this particular term contributes for a positive value, which means that zeta has to be positive.

So zeta has to be greater than zero right. So if we look at this so zeta again depends upon moment reference point. So the if the moment reference point is changing the value of the this particular empirical constant is changing right. And it is changing from negative to positive. So for a particular location on this chord line, this zeta value can be zero, is it not?

Am I correct or not? When it is changing from negative to positive this zeta can also be zero here so for certain location on this chord line here, right. So that particular location is known as aerodynamic center right. So what does it mean? So C m now becomes C m prime, right, which is independent of when zeta is zero, there is no change in C L.

That means, even if you change the angle of attack, C L will change but it will not affect the overall pitching moment about that particular point. So that particular point is known as aerodynamic center. So let us define it.

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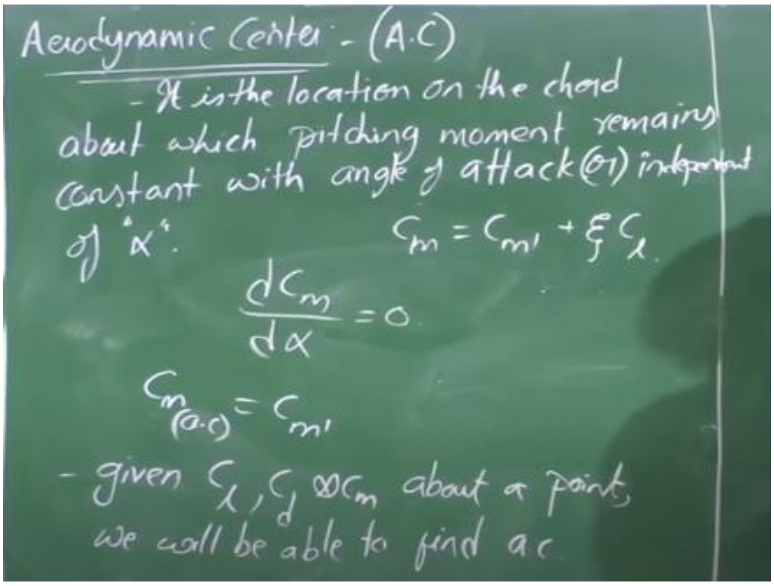

So aerodynamic center. So let us define aerodynamic center. So this place a crucial, so AC right, so this plays a crucial role throughout the, throughout our flight dynamic analysis, as well as in the coming course as well. So pitching moment remains constant with angle of attack right. Or we can say independent of alpha, right. Independent of angle of attack, which means change in pitching moment with respect to alpha is zero about that point, right.

So that is for the airfoil. So we call the corresponding moment about that particular point is C m a.c is pitching moment about aerodynamic center which is nothing but our C m prime here, right. We have our previous equation which is C m is equals to C m prime plus zeta times C l, right. So that is nothing but our aerodynamic center here, fine. So how do you find this aerodynamic center?

Given C l C d, so this C l again two dimensional C d two dimensional which is C capital C D times right that is one and the same here, right and C m about a point. So given C l, C d C m about a point we will be able to find aerodynamic center. So what do you mean by that?

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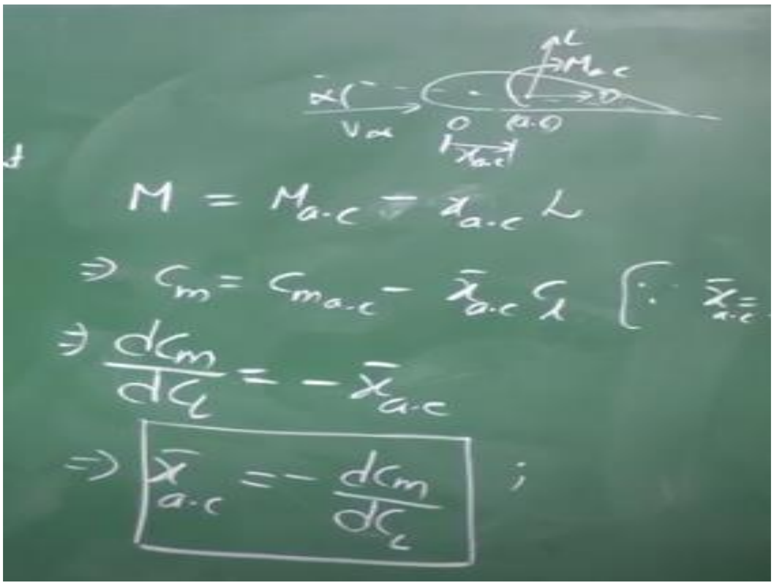

Let us consider an airfoil is at an angle of attack alpha, right. So let us say with respect to point O right I know what is C l, C d and C m, right. So and say there is an aerodynamic center a.c right. Let the positive, x a.c is positive behind this O, point O right. So let us say the offset between aerodynamic center and O is x a.c right.

So since as I told earlier if I know lift drag at a point these are forces I can translate to the aerodynamics as well as lift and drag here right. As well as there is certain moment about this aerodynamic center that we got to know. So for moment about aerodynamic center for a symmetric aerofoil is zero here. So which we witnessed earlier right. So and is negative for cambered aerofoil, positively cambered.

And is positive for reflex aerofoil, okay. Okay, so let us, right moment about O is equals to moment about aerodynamic center right plus x a.c times minus sorry, minus because this lift contributes will contribute for a negative moment here pitched on moment. So times the lift acting here right. So this C m is equals to C m a.c minus x bar a.c times C l, where x bar is equals x bar a.c is equals to x a.c upon c bar, right.

So what is the definition of this aerodynamic center? So dC m upon d alpha is equals to zero. So alpha when there is a variation in alpha there is a variation in C l as well, right. So this C l is again a function of angle of attack here. So what I can do is instead of alpha what I will say is dC m upon dC l, right. So C l with so we know C m a.c is independent of C l. So this turns out to be zero, right.

And what I have is minus x bar a.c. So I can find out the location with respect to that point right is equals to minus dC m upon dC l right? So if I know the lift curve slope of this sorry C m C l curve slope, if I know dC m upon dC l right it is a pitching moment and lift coefficient curve slope right, if I know that within the linear regime, I will be able to find out what is x bar a.c, right.

So if dC m upon dC l is negative, let us say if the slope is negative that means x a.c is positive, which positive is at. That means the aerodynamic center lies aft the moment reference point right. So if this is positive, then this is negative. That means the aerodynamic center lies ahead of the moment reference point, okay. If I know the data of pitching moment C m, C l, C d with alpha at a given point right.

If I am variation with at a given point, then I will be able to figure out what is the corresponding at least I need two data points there. To figure out the slope I need at least three to four data points is it not? Two minimum and then for accuracy higher number of data points. So within the linear regime, why because we are writing this equation for the linear regime here.

So what I have is C l x a.c is equals to minus dC m upon dC l, okay. Now, let us discuss about one more interesting right reference point. So have you ever wondered where this forces are acting aerodynamic forces are acting, right.

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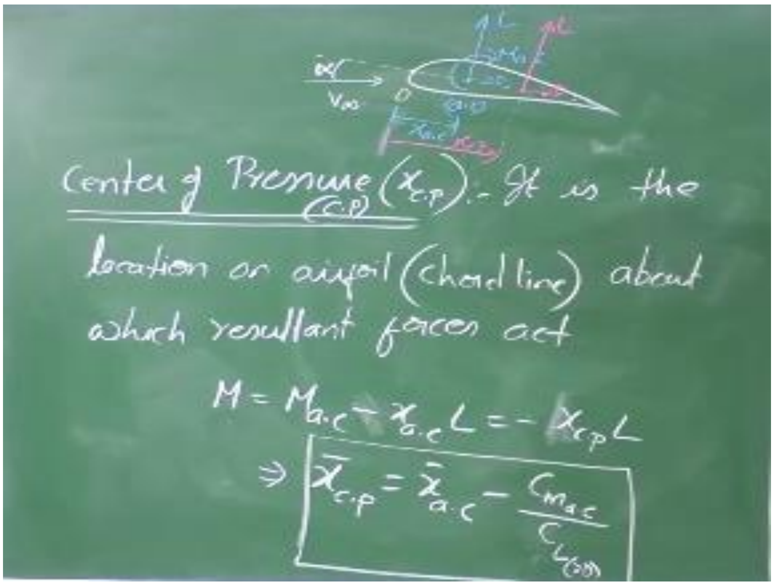

So there is some point called or reference point called center of pressure right. So let us say x c.p, its location is represented by x c.p right. It is the location on airfoil say close to chord line right about which resultant forces act right. So that means, there is no moment about that particular point, pitching moment is zero right. So how to find the relation between this x c.p and x a.c?

So let us say a.c and is represented by x a.c and then this way this is c.p represented by x c.p. So how to find the relationship between them? Let us consider an airfoil. So when there is flow if it is oriented at certain angle of attack. So there will be lift and drag, right. Let us say this is my moment reference point O, okay. So this is my moment reference point O and a.c be the aerodynamic center with respect to this O.

This a.c is at a location x a.c, right, location x a.c. And then let us say there is center of pressure. So about aerodynamics center what can we expect? Moment about aerodynamic center and then lift is it not? Lift acting perpendicular to V infinity right? And then at center of pressure, lift and drag of course, right.

So about center of pressure what we have is same magnitude of lift because it is a rigid body again lift and drag but there will not be any moment about this x c.p right.

So if I write pitching moment equation about point O, so the moment M is equals to moment about a.c plus x a.c times the corresponding lift minus I am sorry. It is a pitched on moment right, minus.

This is equals to. So the moment due to, so the same moment I can achieve by considering forces acting at x a.p, right. So that is equals to minus lift or x c.p times lift acting here, right. Okay, neglecting the drag components which are very small. So for the for this discussion, we are not considering them. So this talks about the same moment right. So because moment about O with respect to a.c and with respect to c.p.

So from here what I can do is x bar c.p is equals to x bar a.c minus of C m a.c upon C l right. Of course, this is 2D. C m a.c upon C l 2D, right. So this is the relation between aerodynamic center and center of pressure. So what happens as the angle of attack increases the C l value increases here. If the C l value increases, so this is not going to change with angle of attack of course.

So the C l value is increasing why because the C l alpha times alpha that is how you modeled it right, C l naught plus C l alpha times alpha. So in the linear resume this is approaching. When it approaches high angle of attack there is a higher value of C l before stall, right. This makes this quantity this particular ratio smaller. That means, so x c.p is trying to move close to this x aerodynamic center.

So let us say for cambered aerofoil C m a.c is negative, right, positively cambered aerofoil. That means, x c.p lies aft the aerodynamic center because this particular quantity is positive. So x c.p is behind x a.c.

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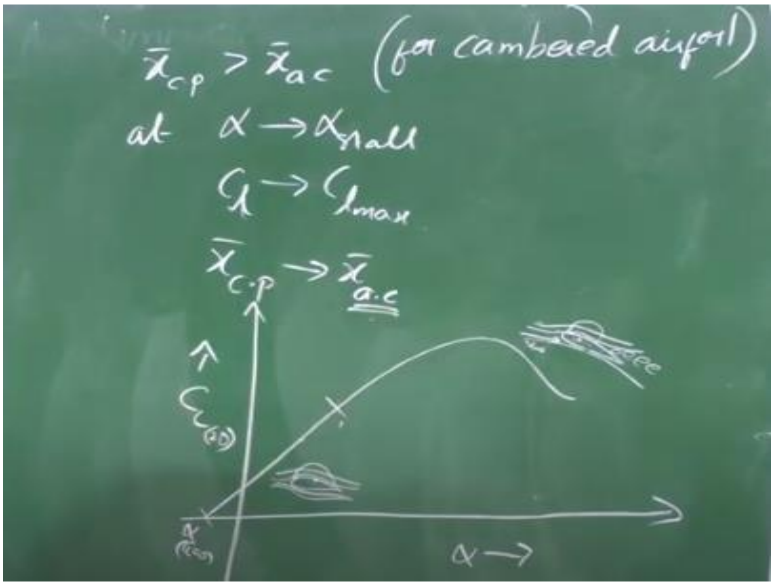

Otherwise x bar c.p is greater than x bar a.c for cambered aerofoil. So it is equals to zero this particular quantity for a symmetric aerofoil right, this particular ratio is zero because C m a.c is zero. So in that case x c.p and x a.c coincides. So and at alpha yes alpha approaches alpha stall higher C l. So C l approaches C l max, right? Which means x c.p approaches x a.c right.

So we will see why it is, okay. So the typical variation of lift coefficient with angle of attack right for an airfoil here, so this is 2D again. So this is where alpha at which C l is equals to zero right. So within this linear regime of angle of attack or say at low angles of attack so you have an airfoil right. We have flow, okay.

So the pressure distribution is the flow is completely attached so you have pressure mostly most of the part for the most of the part the flow is attached. So you will have larger pressure distribution is it not. That means, the centroid is more or less equals to the center of pressure here. You will have center of pressure almost like close to this centroid which is distributed throughout the aerofoil, right.

But at the higher angles of attack what happens? The same aerofoil now placed at higher angle of attack, right. So what happens is, so there is a flow separation right, is it not? So that means the flow the pressure distribution is mainly concentrated near the region where there is a test flow here.

That means, the centroid of this has moved forward compared to the centroid for the previous case, which is this one, right. That means the center of pressure is moving forward right close to the center of pressure, right. So sorry aerodynamic center, close to aerodynamic center. The center of pressure is shifting towards the aerodynamic center.

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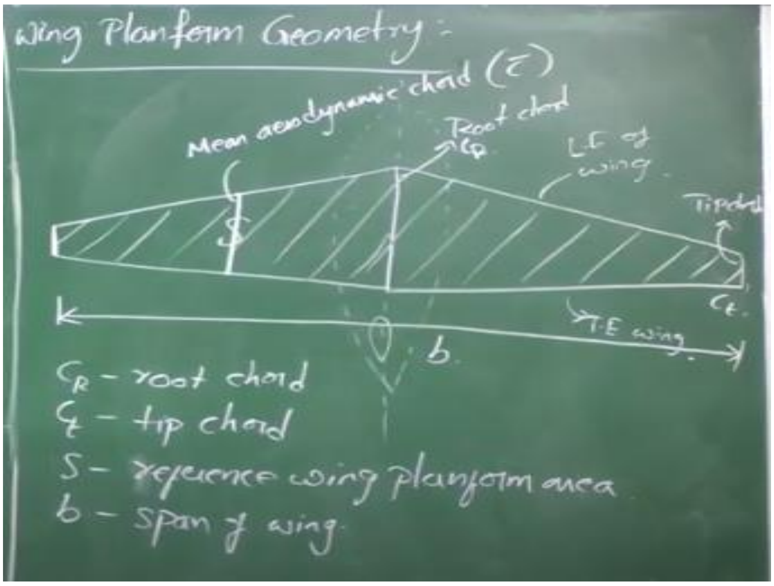

So let us now look at wing planform geometry. Let us consider the top view of an aircraft right. Say this is my fuselage. I have a vertical tail in the top view here. So the wing right we are talking about wing here. Say this is a center line of the fuselage and it is symmetric about this particular center line okay. So this is called the wing, leading edge of the wing, trailing edge of the wing.

We have trailing edge of wing, right. And what we have is, if you take a cross section here, we will have a airfoil right. And the airfoil characteristic length of airfoil is chord here, right. So what we call it as it is towards the root here right, root of this particular wing what we call it is root chord. C R, right. And we call this as tip, root and tip. What we have is root chord and tip chord which is C T.

So the chord near the tip is tip chord and the chord near the root is root chord, it make sense here. And let us define the span of this wing which is the distance between the two tips here, right. Let it be b and the reference planform, reference area of this wing is denoted by S right, S be the. C R is called root chord. C t is called tip chord. S is a reference planform area, right.

And b is a span of wing measured between tips, two tips here, right. And let us look at some of the non-dimensional parameters here that talks about the geometry, right. Again when we non-dimensionalize them, it will be helpful for us to compare aircrafts of different sizes, right.

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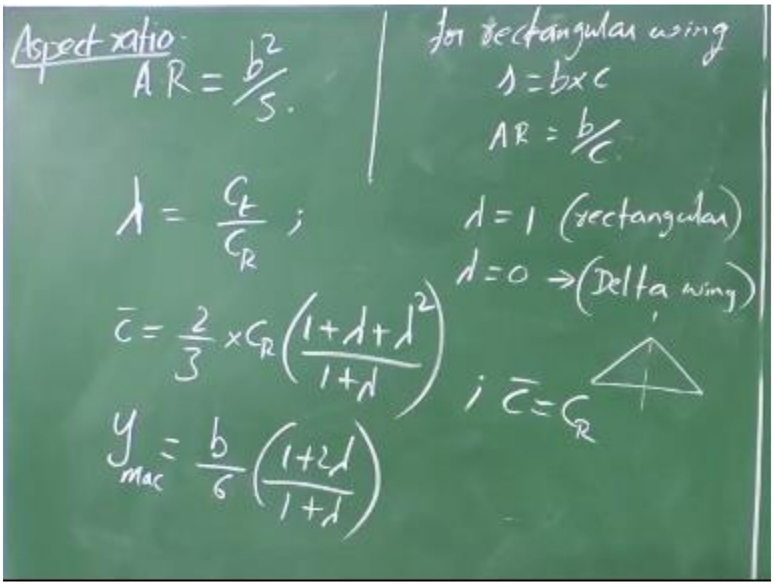

So the first non-dimensional parameter we will introduce here is aspect ratio, which is b square by S, how long your wings are, how big your wings are, right. Higher the aspect ratio higher is your wing span. So for a rectangular wing, this will be b upon c. Why because S is equals to b times c for a rectangular wing. So let me do it this way. So our rectangular wing say aspect ratio is b upon c.

And lambda is called taper ratio. This is called aspect ratio. So higher is the aspect ratio longer is the length of your wing, larger is the length of your wing right. So lambda is equals to C t upon C R right. So for a rectangular wing lambda is equals to 1 and is equals to 0 for triangular wing, right. So this is for delta, we call it as delta wing, right. It is a pure for a pure delta wing the taper ratio is zero, okay.

And then so this chord at tip and root are different is it not? So there can be mean, there will be definitely mean location for this chord, is known as mean aerodynamic chord right, which is known as C bar, right. So C bar for a rectangular tapered wings so it is two thirds C R times 1 plus lambda plus lambda square upon 1 plus lambda. So for a rectangular wing so lambda is equals to 1.

What you have is C bar is equals to C R or C t, both the same right for a rectangular wing. And then the corresponding y location of this mac, mac is known as mean aerodynamic chord right. So this is called C bar, which is known as mac. So y have this mac is b by 6 upon 1 plus 2 lambda upon 1 plus lambda. So these are some of the geometric parameters of this wing platform.

So let us look at what are the aerodynamic characteristics of this finite wings, right. So till now we are talking about infinite wings, which is airfoils. So will there be any difference between this finite wings and infinite wings? We will look at this during our next lecture, right.