Design of Fixed Wing Unmanned Aerial Vehicles

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Lecture - 05
Generation of Lift and Drag

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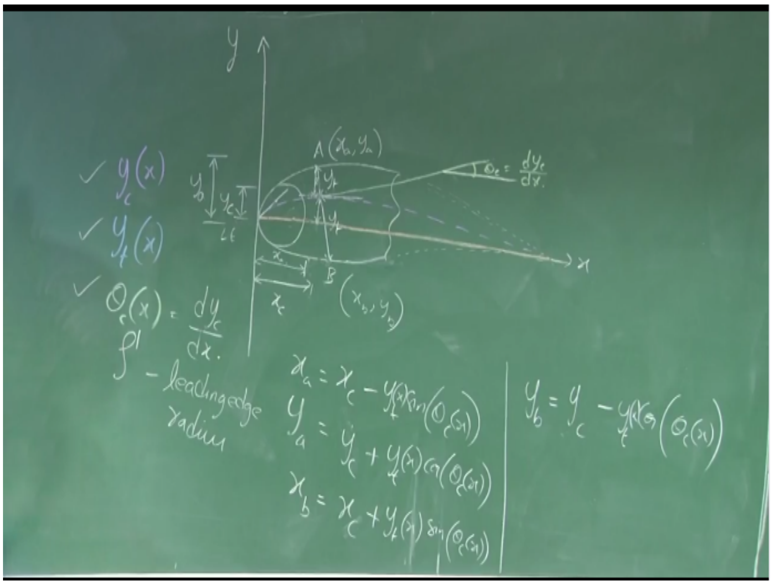

Good morning friends, welcome back. In our previous lecture, we discussed about how to plot an airfoil given camber as a function chord, slope of this camber as a function of chord and the corresponding camber thickness distribution about the mean camber line as a function of chord and also we require the leading edge radius to complete this airfoil. So what exactly this airfoil do? when you place it in a flow?

In other words, if I take that airfoil in my hand if I start running right, so what happens to that airfoil? So that can also be what you call duplicated or simulated by holding the airfoil the same effect can be simulated by holding the airfoil and allowing the flow at the required velocity right. So that can happen inside a wind tunnel. Now let us say we place this airfoil inside a wind tunnel.

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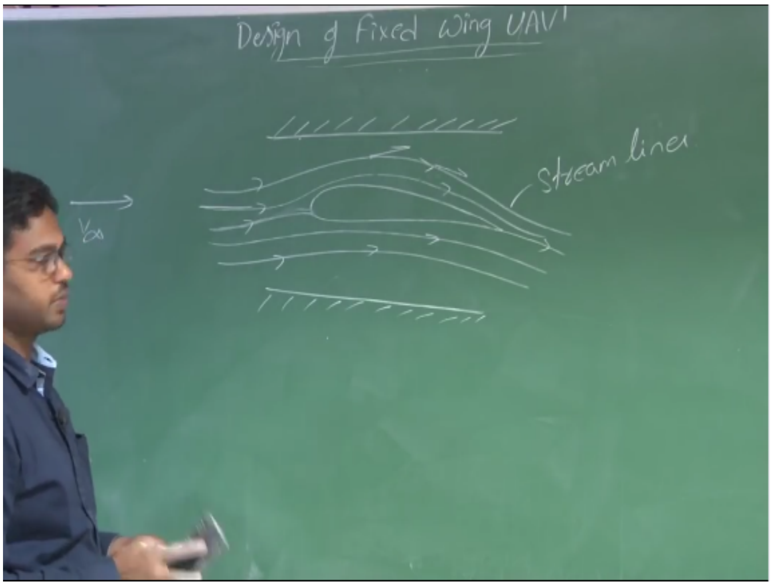

Now say there is a flow of this airfoil okay. So whenever we say flow, we draw these kinds of lines right. What are these lines? They are called stream lines right. So it is assumed that the fluid particle will move along this stream line right, it is a path traced by the fluid particle in which if you draw a tangent at each and every point so if you draw a tangent here it gives you the direction of velocity at that particular point right.

So the fluid particle tends to move in this direction at this particular instant and say if you draw a tangent here which means the fluid particle at this instant will have or at this location will be moving in this particular direction okay and the equation of stream line you know del cross v is=0 right, so we are not going to discuss about it. Now what happens when there is a flow? We say there is a lift but how this lift is generated?

We should talk some philosophy here, although there are many philosophies that talks about how the lift is generated, there is one fundamental like most people believe right we will talk about that philosophy.

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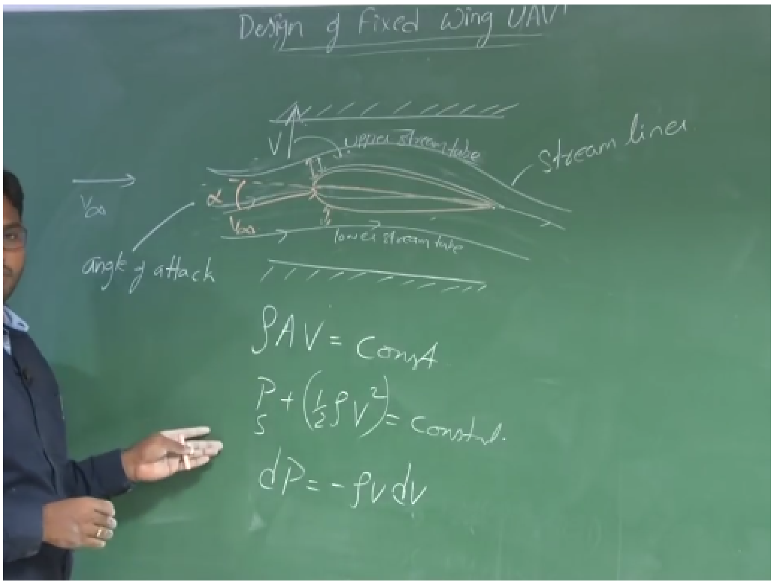

So let us consider a bunch of stream lines right called a stream tube right in which the flow actually happens right. Now let us assume the flow is happening in the form of a stream tube okay. Now as soon as this stream tube encounters the airfoil, it will be split into two stream tubes here right. So due to this radius of curvature right otherwise due to this curvature of about the leading edge curvature that is followed immediately after this leading edge right due to this curvature what happens is the cross sectional area here it decreases.

Let us say if it is a cambered airfoil, positively cambered airfoil this curvature is more above this chord line right. So the flow is assumed to touch the leading edge in the first case right, the flow is supposed to touch the airfoil at the leading edge right. Now there is an upper stream tube when the lower stream tube right okay.

Now due to this curvature, the cross sectional area of this stream tube reduces right. So considering an incompressible flow where when you say incompressible flow it is the density throughout the flow remains constant, density of this fluid remains constant right. Now using continuity equation, law of conservation of momentum right, rho AV is=constant. See if the cross sectional area decreases the velocity has to increase here.

So V increases, V increased here right. At the same time, if you have a symmetric airfoil, you have equal thickness distribution right which also has a similar reduction in cross sectional area on either side. If you have a cambered airfoil, you have greater reduction in cross sectional area of the stream tube above and when you have symmetric airfoil you have equal reduction in area above and below this chord line right.

Now this will accelerate the flow above the surface. Let us talk above the surface right now, so this acceleration will yeah so the dynamic pressure here increases, so from Bernoulli's theorem for incompressible flows of P+1/2 rho V square P static=constant right. When the dynamic pressure increases, there is a drop in the static pressure. So this particular phenomena is known as suction right.

So there is a drop in static pressure here. Yeah on the bottom surface, the cross sectional is reduced although there is an acceleration right, here also the cross sectional has reduced but the acceleration is not as high as that happens in the upper stream tube right. So because of which there is a greater pressure drop, there is also a pressure drop on the bottom surface but not as high as the upper surface right.

So this pressure difference will help in generating lift. This now the same symmetric airfoil if I use, so for symmetric airfoil say so it has equal thickness distribution about the chord line, so you have equal reduction in cross sectional area here. So the acceleration will remain same on upper stream tube and lower stream tube right. Hence there is an equal pressure drop on upper and lower surface, you understand this right.

There is an equal pressure drop on upper and lower surface, static pressure drop right. So eventually a symmetric airfoil at 0 angle of attack does not produce any lift right. So you need to now we have defined some term called angle of attack right. So let us say if this is my reference line of the chord line of this airfoil of the symmetric airfoil, the angle made by this reference line with respect to free stream let us say if I can represent the free stream like this.

This angle is known as alpha, see this V infinity is nothing but the velocity at which I am moving in a stationary air assuming there are no winds right, there are no gusts, winds and disturbances. If I am moving it in a stationary air at velocity V infinity that means the air is also moving towards me with the same velocity right. So this velocity V infinity represents the velocity of your flight vehicle right.

And the angle made by this with the reference line of the chord line here for a wing or an airfoil, so chord line we consider as a reference line for a wing and airfoil. For fuselage, we can say fuselage reference line or center line right. So this particular angle is known as angle of attack that is defined angle of attack.

So when there is an angle of attack then you have greater I mean the curvature on the upper surface or for the upper stream tube will be more compared to that of lower stream tube which eventually results in a greater pressure drop why because there is a greater reduction in cross sectional area which helps to accelerate the flow right and then the pressure drop and it is true for even compressible flows.

Like dP= -rho V dv Euler equation, if you consider Euler equation where if there is an increase in velocity right and density can never be negative for the fluid right, so there will be a drop in the pressure. So it is true irrespective whether it is compressible when there is an increase in velocity, there is a drop in static pressure. It is true irrespective of whether it is compressible or incompressible.

Let us see what is the pressure, how the pressure distribution will be there for an airfoil right. We say there is a pressure drop right, so how this distribution will look like?

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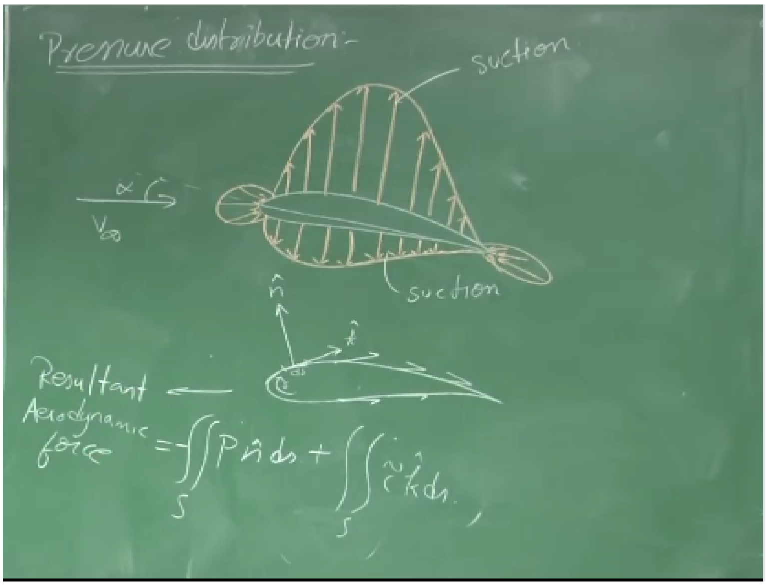

So you understood right, there is a pressure drop here right and there is a pressure drop but the drop in the pressure above will be higher for cambered airfoil even at 0 angle of attack right. So it is higher compared to that of lower surface, so this pressure difference will help you in generating lift right. In case of symmetric airfoil, you have to maintain certain angle of attack so that you can attain this pressure difference so that lift can be generated.

Pressure distribution, let us say this is my V infinity. So this is my airfoil right which is placed at an angle of attack alpha okay. Now so this is how typically a pressure profile for an airfoil right. So this is a cambered airfoil or you can also have symmetric airfoil let set an angle of attack right. So the arrows here pointing away from the surface right, so this represents the suction or negative pressure okay.

So even on the bottom surface we have suction on negative pressure but the difference is the suction is more here on the upper surface right. Now we are satisfied that there is a pressure distribution whenever we place an airfoil in a flow right and also the wing right. Since we are moving in a fluid, there is always a chance that I mean this body encounters some friction with the fluid right.

So that friction is the shear stress right, so shear stress will be if the body is moving in this direction say in this direction, so will be acting opposite to it right, will be tangent to this surface. So there will be a shear stress acting tangential to this surface due to fluid friction right. Let us consider ds be the small surface, so let n cap with the unit vector normal to the surface and k cap with the unit vector tangential to the surface right.

Now the pressure over the surface can be, it is a negative pressure right, P*n cap*ds the force due to pressure, so there is some aerodynamic force right+surface integral of tau*k cap*ds right. So these are the two forces right that results in a resultant aerodynamic force. One is due to friction, the other one is due to pressure right.

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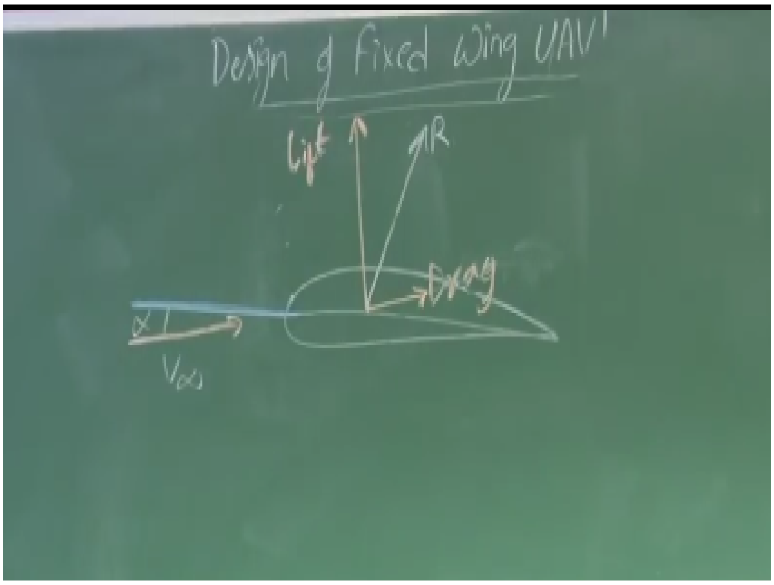

This is actually the extinction of this chord line right. This is my V infinity and this is my angle of attack right. So there is a resultant aerodynamic force R acting right, do you accept this? Because that is what we are able to I mean estimate by integrating this pressure force and shear stress distribution right. So now a component of this resultant aerodynamic force acting perpendicular to free stream is lift.

And the component of this resultant aerodynamic force which is acting parallel to free stream is drag. So if you say the arrows represents the magnitude then drag is much smaller than lift right, understand. So lift, we have defined lift and drag, so they belong to wind right. We also call at a later stage we see this is a wind access right when you talk in terms of lift and drag we are talking about wind access, forces in wind access is the lift and drag.

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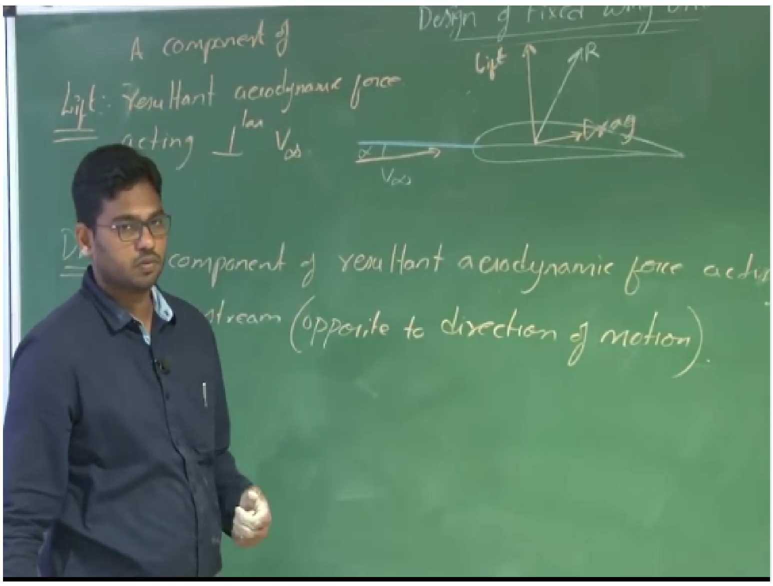

Now let us define lift. Say resultant aerodynamic force oh sorry, a component of resultant aerodynamic force acting perpendicular to V in free stream velocity right and now drag, this is also a component of resultant aerodynamic force acting parallel or along the parallel to free stream velocity and it is acting opposite to the direction of motion or in the direction of free stream velocity okay. Now we have defined lift and drag here right.

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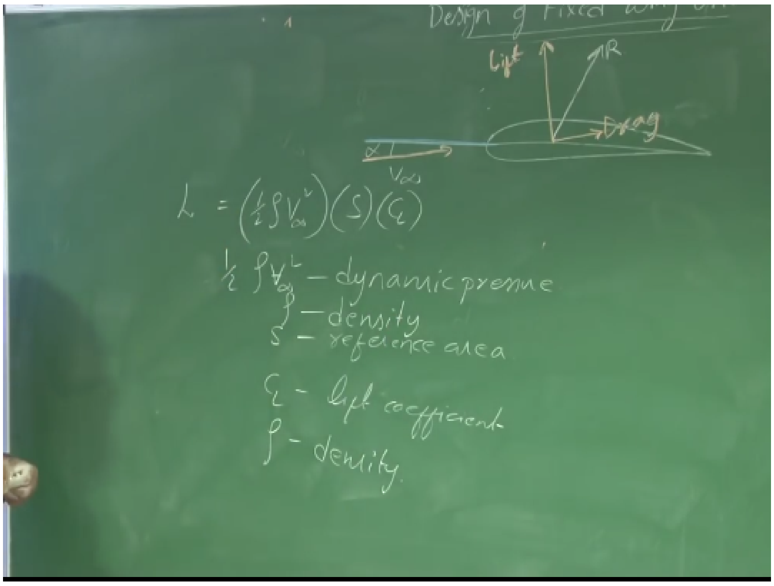

So let us denote L represents a lift here which is expressed as 1/2 rho V square dynamic pressure times reference area*non-dimensional force coefficient called CL, CL is a coefficient of lift, so 1/2 rho V square is dynamic pressure and S is a reference area and CL is a lift coefficient where rho is the density. Again, although you are moving at same speed with the same aircraft at the same CL right at different altitudes you will end up producing different lift because rho will change so density right.

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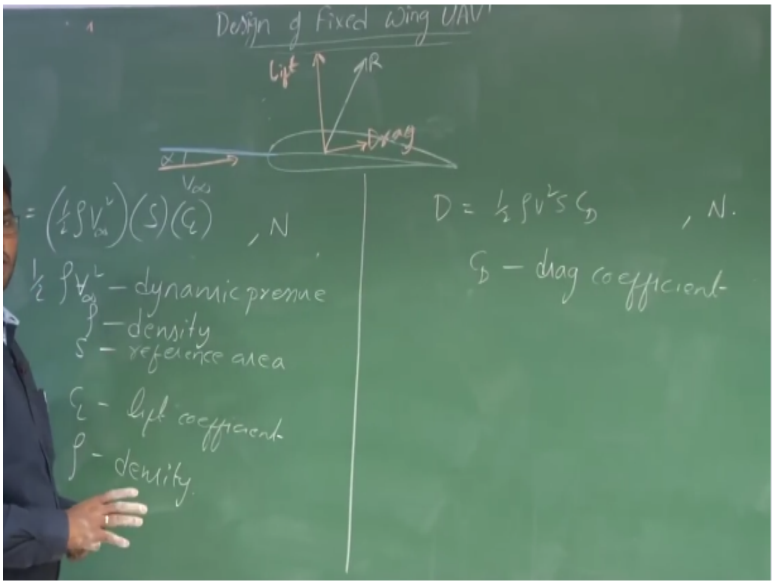

And similarly drag is defined as 1/2 rho V square S CD right where CD is the drag coefficient. So what are the units of lift? Newton right, units of drag is Newton. What about units of CL and CD? They are non-dimensional right, non-dimensional quantities. Let us now assume that we place this airfoil inside a wind tunnel right and vary the angle of attack and measure the corresponding CL right.

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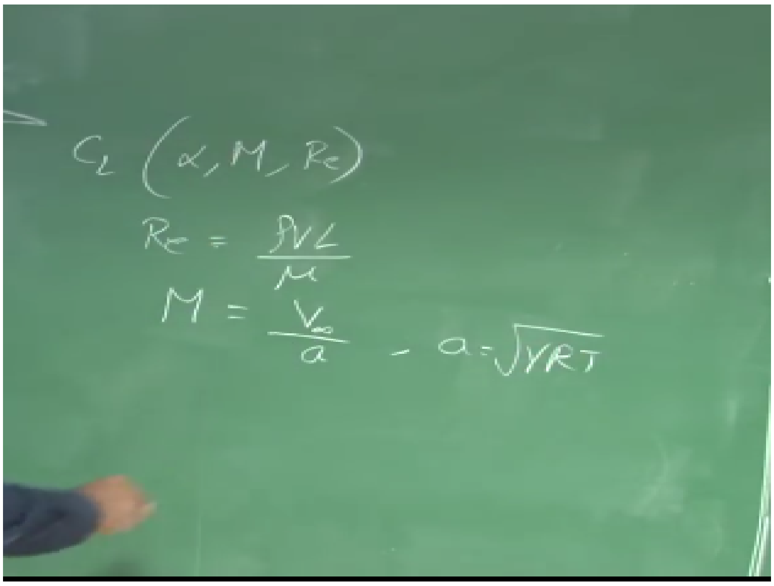

See here the CL is the function of alpha, Mach number, Reynolds number right, in general a function of this where Reynolds number is defined as rho VL/mu and Mach number is defined as velocity of the flight/velocity of sound where a is=square root of gamma RT right.

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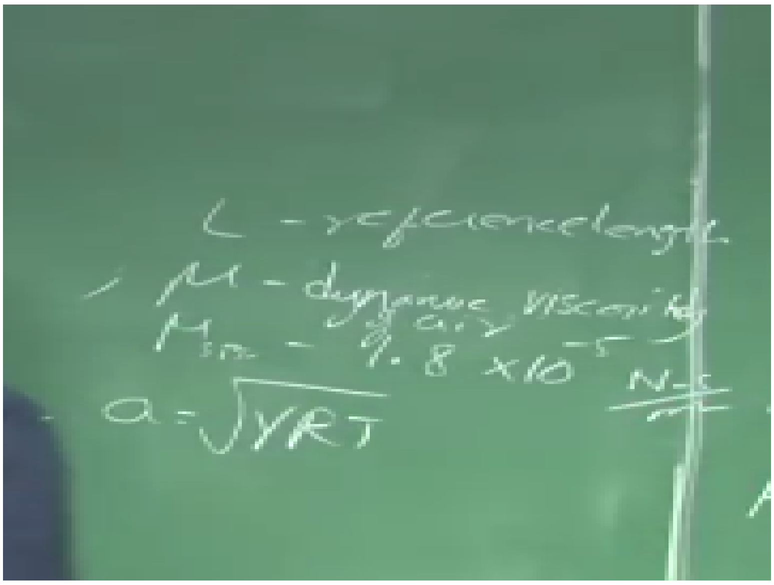

Mu here is L is reference length, ratio of inertial forces to the viscous forces right mu is the dynamic viscosity of air. At STP, mu at STP about 1.8*10 to the power -5 Newton second per meter square.

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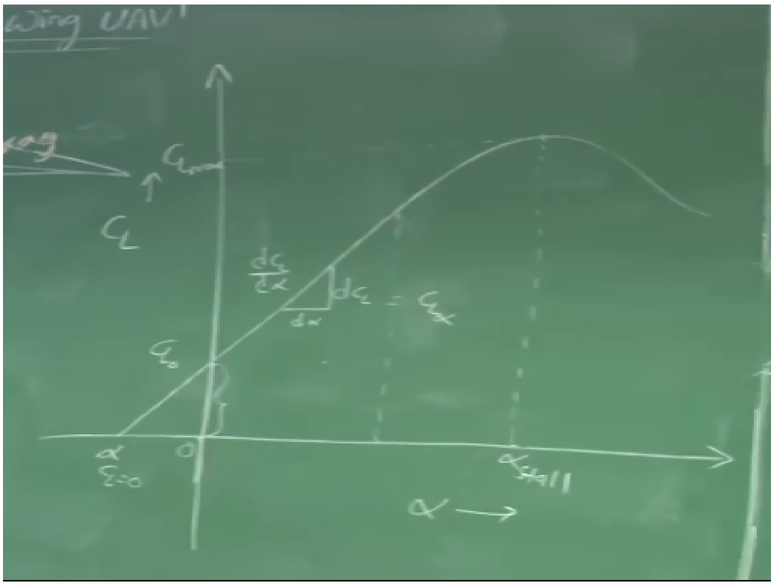

So now let us vary this alpha and see how CL varies okay. So CL is plotted along the y axis and alpha is plotted along the x axis right. So for a cambered airfoil, this is how a typically a CL versus alpha plot look like right. For a cambered airfoil, you have lift coefficient called CL0 even at angle of attack is 0 right is the y intercept at alpha=0 is the CL0 okay and you have alpha at which CL=0 right.

This is negative for a cambered airfoil and this slope see until certain angle of attack, this almost remains linear right within this region you can define the slope dCL/d alpha. So this slope is dCL/d alpha known as lift curve slope which is often termed as CL alpha right. You know beyond this there is certain nonlinearity and if you further increase the angle of attack, so the CL becomes maximum right.

And beyond that if you further in case that angle of attack beyond the CL max point, the lift suddenly drops, starts dropping right. So this particular point is known as alpha stall, the corresponding angle of attack at which CL maximum is alpha stall, so beyond stall you will start losing lift right, drag will increase so that we will see, how the drag increases. Now see in the linear region if I have to model this lift right, if I have to write an equation for lift with the variation of angle of attack.

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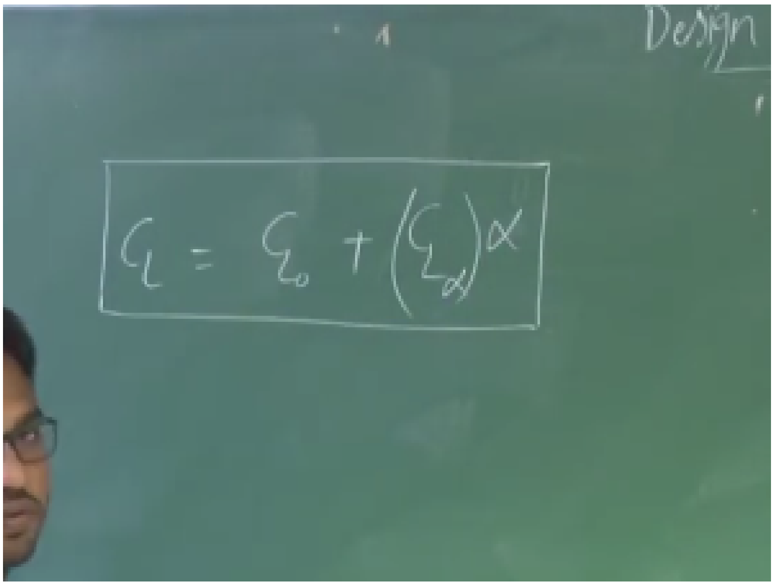

So since it is a linear curve up to certain angle of attack what I can assume is CL=CL0+CL alpha*alpha right. Since it is a straight line right y=mx+c, y here is CL and y intercept is CL0 that is C is CL0 here, m is slope CL alpha and this is alpha. So I can use this equation to model the CL as a function of angle of attack right up to certain region.

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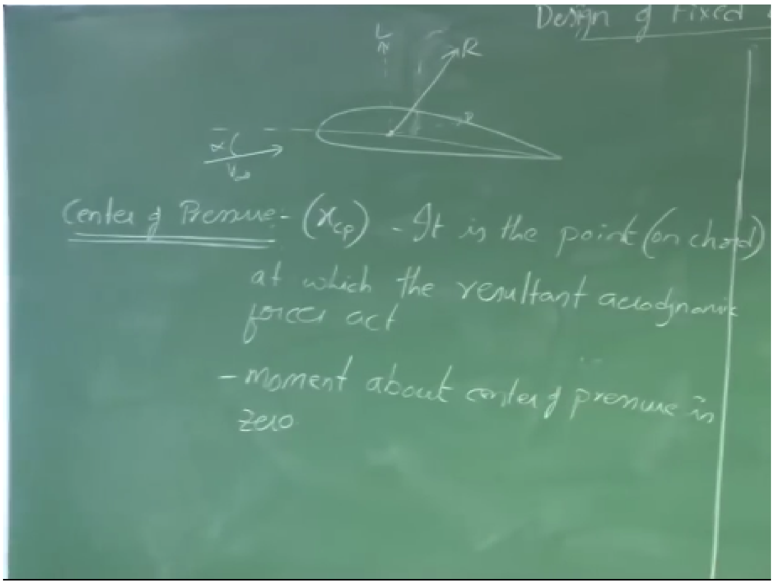

We took it granted that this resultant there exist a resultant force right when there is a flow right and we resolved it along the flow and perpendicular to the flow and we got lift and drag right that is the story till now. We got lift and drag but where does this resultant force act, we need to question that right we need to know, where does this point exist? Let us say if it exists, so where does it exist?

So the answer is center of pressure right. If you look at here so there is a pressure distribution here, so the average of this will act at certain point and there is a pressure distribution on bottom surface or the average will be acting at some point here right, maybe the centroid of this particular distribution right. So that particular point that lies on the chord is a center of pressure.

So let us define center of pressure right, we call xcp, it is the point mostly on chord line right on chord at which the total the resultant aerodynamic forces act or force act right. So at center of pressure if you take a moment about center of pressure, it will be 0, so moment about center of pressure is 0 right. Now we have defined something called moment right where this moment is coming from okay right.

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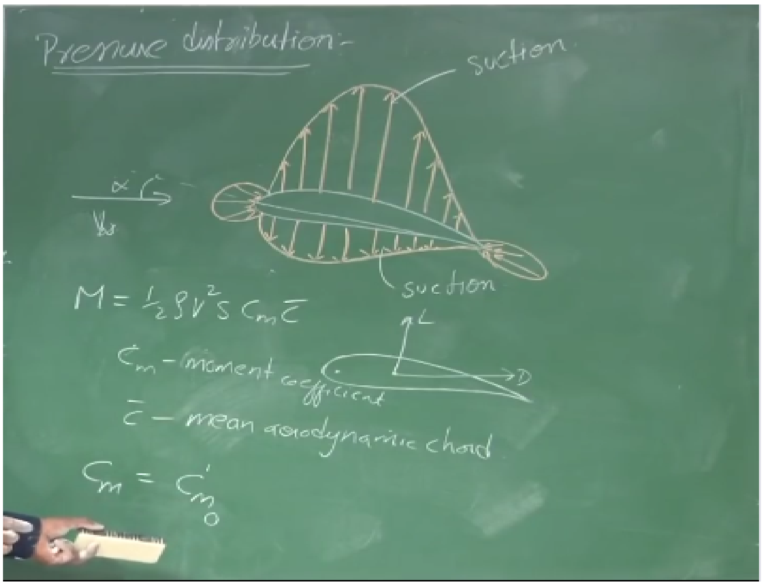

So consider an airfoil, so there is a lift, there is drag right and there is V infinity okay. So we have defined lift and drag. Now let us also define moment M right, M about a point, you have to know the reference point of this moment is=1/2 rho V square s*Cm*C bar right. So where Cm is a non-dimensional pitching moment coefficient or moment coefficient okay and what is C bar? It is a mean aerodynamic chord.

We will see what is this mean aerodynamic chord, chord for a wing or chord for airfoil. Now in this case, if I have to write moment about a point right and there is a force right, so let Cm be the pitching moment coefficient is=about a point any point say so moment at Cm prime is 0 which represents moment at 0 lift right and this is a pure moment that is acting when there is no lift right.

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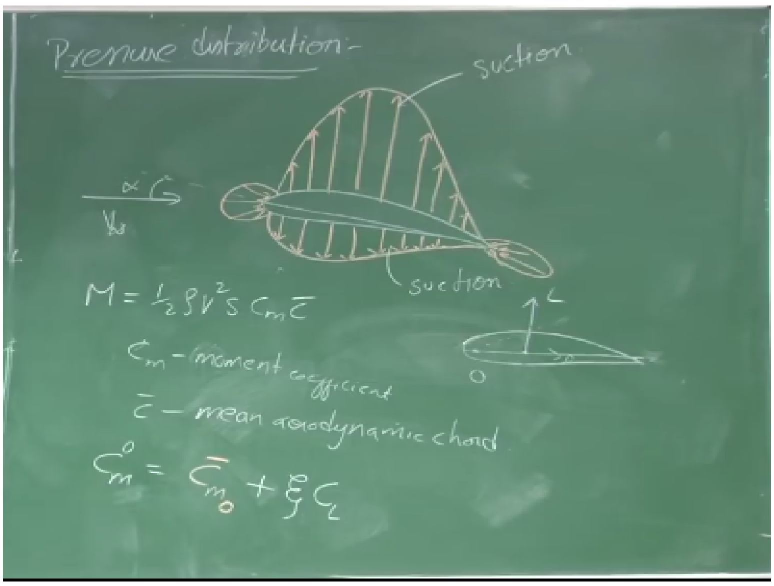

There is lift, there is drag right so say there is a point O right. So the moment about point O moment about so pitch up is considered positive, pitch down is considered negative right. What is pitch up here? Consider y axis into this blackboard right, stretch your thumb along the positive y so pitch up is considered negative, so the curl of fingers give you the positive moment, so pitch up is considered positive and pitch down is considered negative right.

So let Cm0 prime be the moment when the lift is 0, let us say if there is a moment right. Cm0 bar let us say Cm0 bar is the moment at CL is=0 when there is no lift right+some zeta*CL moment due to lift right. Can we write like this? Pitching moment or the moment is=moment when there is no lift+some which depends upon the location of your so this zeta depends upon the location of your moment reference point right.

Let us say Cm about O is=Cm0 prime bar+zeta*CL okay. Now let us consider this point O or the moment reference point at the leading edge right. Now definitely this xcp, so definitely there will be see xcp will be acting somewhere else that means the lift is acting at that point. Do you accept that? So lift will act at xcp right here.

So that means zeta becomes see we are considering moment about the leading edge, now xcp is somewhere behind so the lift will contribute a negative moment that means CL cannot be negative, zeta has to be negative right. So if we consider the most forward point, the zeta has to be negative here. So let us say if we consider this moment reference point is towards the trailing edge then zeta becomes positive right.

Now see zeta is varying from positive to negative right. So now there exist a point at which zeta is=0. Do you accept this? That point is known as aerodynamic center.

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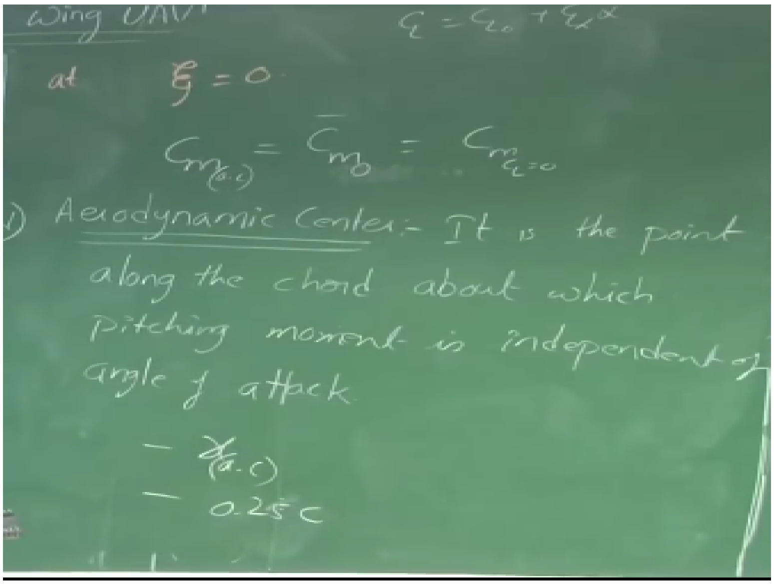

When zeta is=0 right so what is happening? When you consider this moment reference points towards the leading edge, you have zeta as negative why because CL will contribute a nose down moment here. So CL is positive, lift is positive right, zeta has to be negative and if you consider the same moment reference point at the trailing edge right, so zeta becomes positive, so there exist a point when zeta is=0 right.

At zeta is=0 you have the moment about that point that zeta is=0 is=moment when lift is=CL=0 right. This is what the same thing Cm0 bar=Cm where CL is=0 right. So this particular point is known as aerodynamic center. So this is like moment about aerodynamic center a dot c right. Now let us define an important variable aerodynamic center okay. It is the point along the chord about which pitching moment is independent of angle of attack right.

Now we witness that we have model CL right, CL as a function of angle of attack, CL is=CL0+CL alpha*alpha. The CL varies with angle of attack right. Here if I take moment reference point as the aerodynamic center, the pitching moment will remain constant that value is Cm at CL=0, so this is like Cm about aerodynamic center. So which remains constant irrespective of angle of attack.

Why Cm exist? So you call this as xac, aerodynamic center as xac right. For a low speed, it is observed that it lies at the quarter chord almost at 0.25 of c right almost at 0.25 c but what is this moment about aerodynamic center? Why does it exist even when lift is 0? So when can lift be 0 here? Let us come back to this figure. When can lift be 0?

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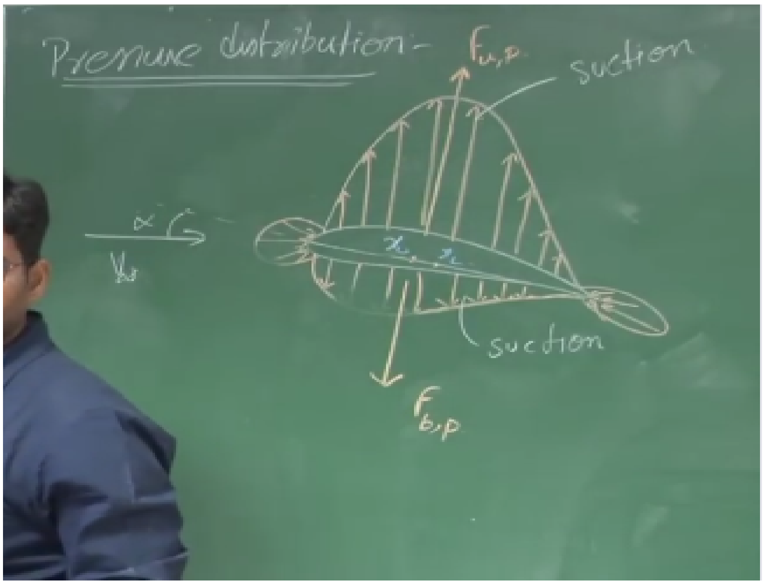

Let us say if the total resultant force if you integrate along the upper surface is=resultant force along the bottom surface right, you understand. If you integrate this pressure over this upper surface, you will get a force that is force on the upper surface due to pressure distribution, force on the bottom surface due to pressure. If these two becomes equal, then CL becomes 0 right at 0 angle of attack let us say.

For symmetric airfoil what happens? The upper surface distribution, bottom surface distribution becomes equal and it is 0 right and this resultant this integrated force like will be acting at a particular point on the upper surface right say along a point on the chord line right, so the bottom surface integrated force will also act at certain point right. Let us say this is from the upper surface, x upper surface, x lower surface right.

So for symmetric airfoil this pitching moment about aerodynamic center is 0. Why? Because these two points will coincide right. For a cambered airfoil even CL is=0 right. These two point will not coincide okay. So there is a pure couple because two forces are acting it in offset, there is a couple which is independent of moment reference point. So this Cm about aerodynamic center is the pure couple acting. So how should I relate this aerodynamic center and center of pressure?

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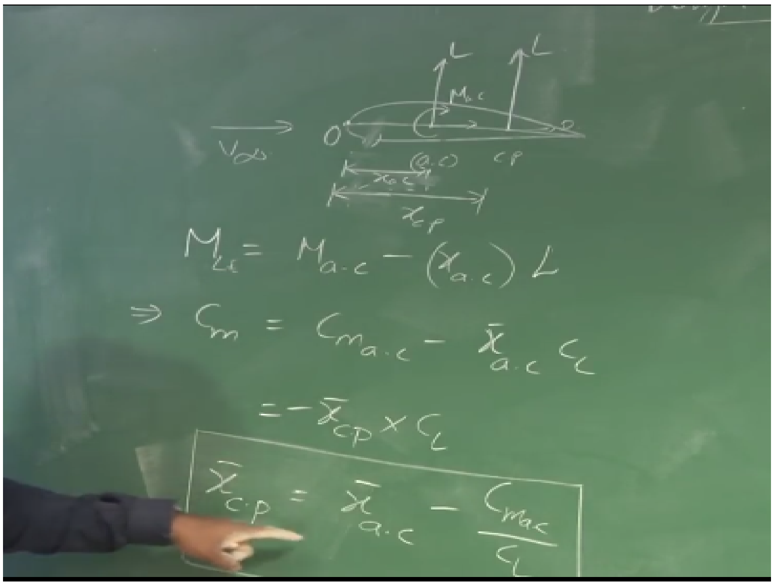

Let us consider a point O on the chord line, let this be aerodynamic center right and there is a center of pressure. So let this reference be xcp from this point O and xac right. Let xac be the aerodynamic center from point O and xcp be the center of pressure from point O right. So there is infinity. There is lift right and drag, the same lift and drag you can represent at the aerodynamic center followed by a moment that is moment about aerodynamic center right.

Now take the, now let us consider a moment abut point O right, let this is be point O, moment about point O right by considering aerodynamic center, so moment about aerodynamic center-xac right*L right. This equals to CL about point O is=Cm ac-x bar ac*CL. Now consider moment about point O with respect to center of pressure right. So both the moment should be equal, this Cm should be equals to -x bar cp center of pressure*CL, yeah -, x bar cp=x bar ac right -Cm ac/CL right, Cm ac remains constant.

This is a relation between center of pressure and aerodynamic center. Let us say this point OB is at the leading edge okay and please correct this, you can consider this as xac, otherwise you have to consider XO and take the difference between them right because I would like to use leading edge as a reference point to measure the length right.

So xap and xcp, now this becomes like moment about leading edge. Now xcp is=xac-Cm ac/CL. So this is the relationship between center of pressure and aerodynamic center. Now as angle of attack increases what happens? CL will increase, so this quantity decreases that means xcp becomes close to X aerodynamic center. So at higher angles of attack, this center of pressure will start moving towards aerodynamic center. Let us see why? Let us assume this is when the flow is completely attached.

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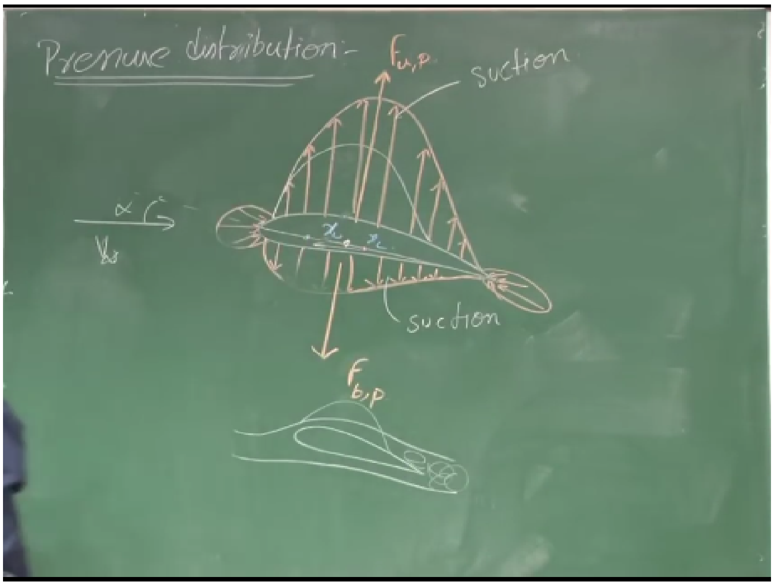

As you go to higher angles of attack, the flow may not remain attached here right, that means the effective pressure distribution will be for a smaller area of this side. So the resultant pressure will act somewhere in the centroid right centroid part of it, somewhere at the centroid. Say if this is your resultant pressure distribution after I mean at higher angles of attack, then xcp for initial case and this case will vary right.

So this may shift, this will shift towards the aerodynamic center, say this is your aerodynamic center, this xcp will slowly shift towards your aerodynamic center. So when this can happen? At higher CLs. When can you achieve higher CL, when you have higher angle of attack here right.