NOC: Introduction to Airplane Performance
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Lecture - 05
Concept of Life Aerofoil: Wing: Complete Aircraft

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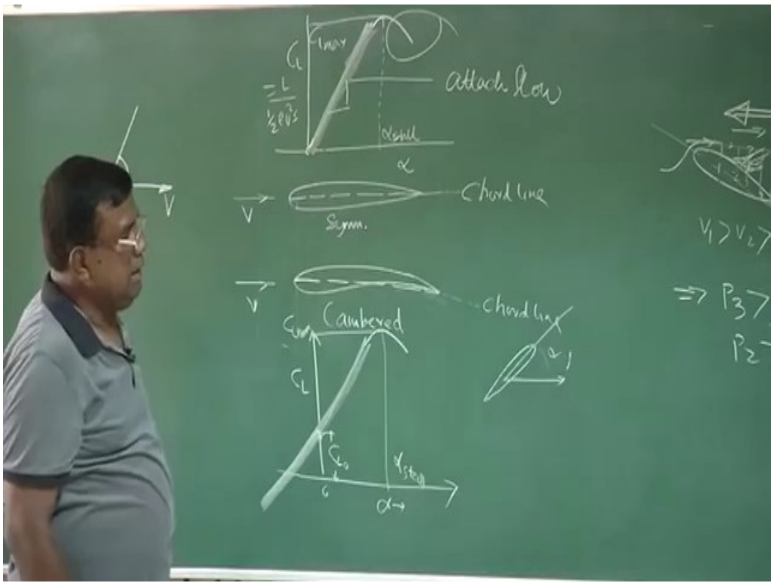

See for a symmetric aerofoil, if this is the v, this is the v, and this is the chord line and if I join this, this is the chord line. What do we notice here? Remember, chord line we defined as if we join leading edge and trailing edge by straight line that becomes the chord line. So, when here the leading edge and trailing edge are joined and that become the chord line for a cambered aerofoil, this is cambered and this is symmetric.

Now, what is in your mind? If the air is coming like this, do you think this will produce any lift? The answer is no, because we remember as far as George Cayley's guideline to us, On the Concept of Generation of Lift. Lift will be generated when there is an angle between the velocity and the surface. In this case, it this there is no angle, so it will not generate any lift. So, if I draw C L which is nothing but, lift divided by half rho v square S, $C_L = \left(\frac{1}{2}\rho V^2 S\right)$ versus angle at alpha (α) equal to 0, I will also get a 0 lift or a 0 CL. But, for cambered to see, you verify this chord line, it is no more parallel to the velocity vector.

You could see actually effectively it has an angle with velocity vector, even if it is at alpha equal to 0, similar condition, Ok. So, in this case what we will see that, even at alpha equal to 0, there will be some lift. I repeat this, one way to explain is this chord line and there is an angle between the chord line and the velocity vectors. So, what we talked about alpha equal to 0, a same velocity condition it will produce a 0 lift, but here because of this camber, because the camber is there, the some angle some surfaces is there, which is making the angle with the velocity vector.

So, even the alpha equal to 0, it will produce lift. As I increase this angle, what will happen? As I increase this angle what will happen, there will be more lift, because from this George Cayley explanation, the lift will be function of angle for a given area, given other conditions. So, as I increase the angle the lift will increase, so you will find upto certain point lift will go on increasing as alpha is increased.

But, beyond this certain point you will see that, beyond a certain alpha you will see that lift is no more increasing. In fact, it is going down like this, similarly for here going down like this. What exactly is happening? Typically, here if I see there is a limit which you called alpha stall (α_{stall}) and this value you called C L max ($C_{L_{max}}$). Similarly, here you call it alpha stall (α_{stall}), and this CLmax ($C_{L_{max}}$). What do you say that beyond a certain angle called alpha stall the flow will no more remain attached. There will be a separation and there will be a stall and the lift will reduce and drag will increase, right? Flow will no more remain attached or there is some sort of a flow separation. What is actually, loosely happening?

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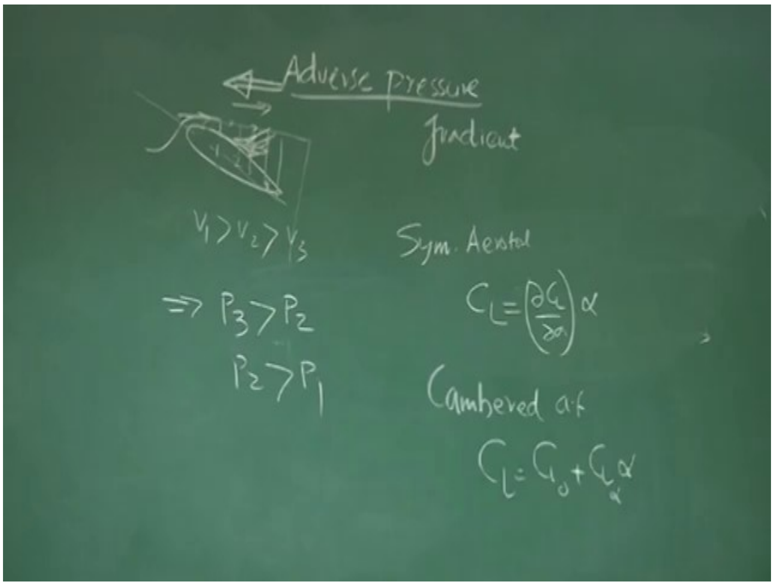

If you see, if an aerofoil is an angle, now see what is happening here. The air flow is coming like this, now as it comes backward, it's relative air. What again you are seeing that, if I take a control surface? I see here, as the air particle moves in the backward direction, the area at 1, 2, 3, 4 all these stations, the area is going on increasing for a given control surface. So, area increasing means the velocity V1, V2, V3 (V_1, V_2, V_3).

So, V1 is greater than V2, V2 is greater than V3 $(V_1 > V_2 > V_3)$, that is as area is increasing as, so maintain the same amount of fluid flow, the velocity has to reduce. So, V2 will be less than V1, V3 will be less than V2, V4 will be less than V3 and this implies, the pressure at 3 is greater than pressure at 2, pressure at 2 greater than pressure at 1 $(P_3 > P_2)$, $(P_2 > P_1)$. So, it will experience an adverse pressure gradient, ok! or at this stage, we only talk about adverse pressure and this adverse pressure will try to discourage the fluid particle to move in the direction and moreover, because of skin friction already some part of energy of the fluid is taken out.

So, there will be an angle at which the, the flow will not be able to move backwards, at some point it will separate, right! and that is, we say the flow is no more attached. Stall is much more than this, but we need to know that when we are talking about this zone, where it is almost linear. I am talking about attach flow and when I am talking here, I am talking about separated flow. There is a subsidiary aerodynamics to explain all these things, we are not going deep into it.

We only need to understand one thing, that for symmetric aerofoil I can write model CL as dCL by dalpha into alpha .

$$C_L = \left(\frac{\partial C_L}{\partial \alpha}\right)\alpha$$

What is dCL by dalpha? DCL by dalpha, because it is linear, straight line, almost straight line, right? so here this is a slope. What does almost straight line in practice will find? Beyond 6, 7 degree some sort of a non-linearity comes. So, but we are assuming here, after this point this is straight.

So, I can write CL as slope of this into alpha, but for cambered, for cambered aerofoil I will write CL as CL0 plus CL alpha into alpha.

$$C_L = C_{L_0} + C_{L_\alpha}\alpha$$

This is just a question of, because for cambered aerofoil I will be actually doing it like this, that alpha equal to 0, there is some CL which is I will be referring to as CL0, so I can model CL as this. In text book, when we try to distinguish between CL because of aerofoil and CL because of wing, we use strict nomenclature.

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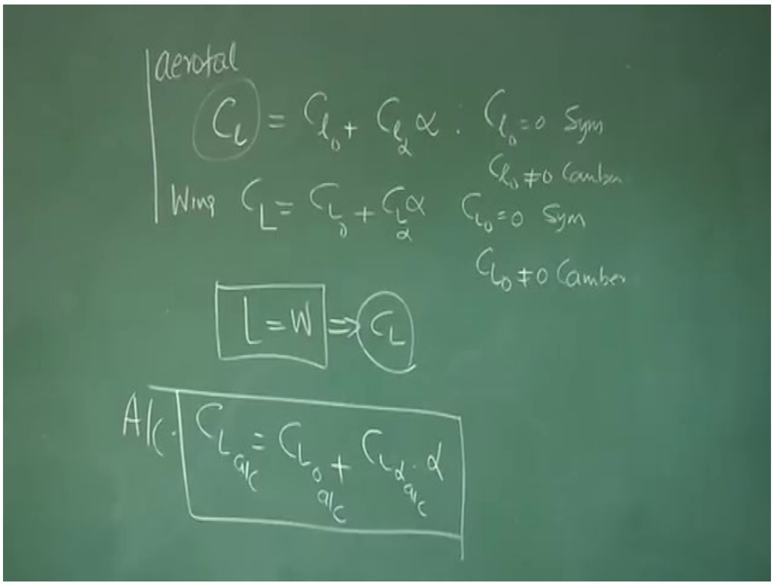

For aerofoil, we use C small l $\left(C_{l_0}\right)$ and for wing we use C capital L $\left(C_{L_0}\right)$ of course, in aerospace there is confusion, even for only moment we write Cl. So, let us be very clear, we are talking about lift coefficients. So, for aerofoil I will write Cl equal to Cl0 plus Cl alpha into alpha, $C_l = C_{l_0} + C_{l_\alpha}\alpha$

where Cl0 equal to 0 for symmetric and Cl0 not equal to 0 for cambered

$$C_{l_0} = 0 : Symmetrical$$
$$C_{l_0} \neq 0 : Cambered$$

and for wing, I will write CL equal to CL0 plus CL alpha into alpha, $C_L = C_{L_0} + C_{L_\alpha}\alpha$ again CL0 is 0 for symmetric and CL0 not equal to 0 for cambered, right?

$$C_{L_0} = 0 : Symmetrical$$
$$C_{L_0} \neq 0 : Cambered$$

Why this is important? You will soon realise that as far as flying the machine is concerned, if I have to maintain lift equal to weight, I need to fly at a particular CL which will be governed by the weight, speed, etcetera, etcetera. But, a question is how do I generate this CL when I am flying, the pilot how will you generate? That answer will come from here, if I know what is the CL0 of the airplane, if I know what is CL alpha of the airplane, then I know if I have to generate CL which I know priory, then I know how much angle of attack I should fly.

So, I know how much I should turn the airplane right, so this is the, that is why in performance this is important, the back of your mind, please. Now, I am talking about aerofoil and wing, for the whole aircraft we will try to find out, for whole aircraft we will try to find out, for CL aircraft, what is CL0 of the aircraft and what is CL alpha of the aircraft into alpha.

$$C_{L_{a/c}} = C_{L_{0_{a/c}}} + C_{L_{\alpha_{a/c}}}\alpha$$

Please, see the distinction. First one is aerofoil, second one is wing and third one I am talking about aircraft. But, now what is the difference between aerofoil, wing and aircraft? Why we are using these three terms? Let us see that.

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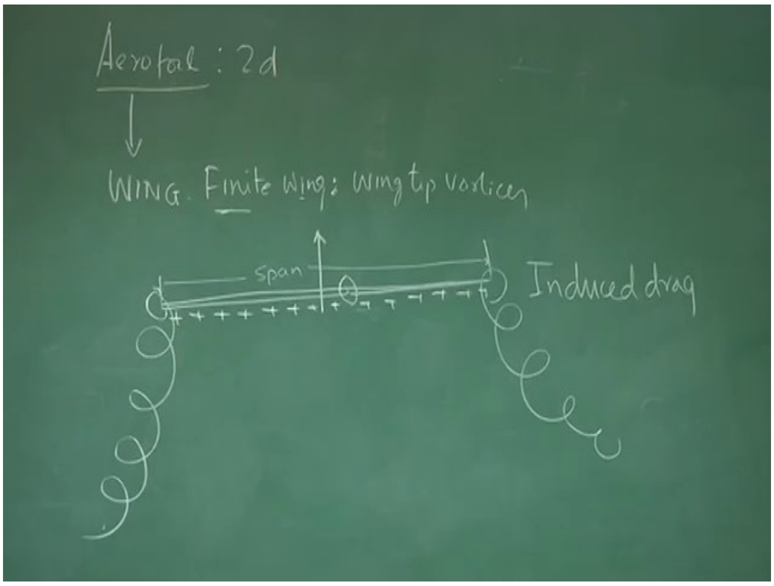

When I am talking about aerofoil, imagine aerofoil is basically a 2D concept, 2D that is imagine this is ,you have seen this aerofoil shape and imagine this having a span infinite, that is infinite here, infinite there. So, what is the basic message is, as the flow is coming like this, it has no way to go towards cross, right or left. So, always the flow is over the each aerofoil section. So, there are no flows around right or left, because these are infinite span.

So, that is why you called aerofoil is a 2D concept, but in actual practice what happens. See, when I come to the wing from, so this, from here when I come to the wing, many text books used a word called finite wing. This finite comes from here, the aerofoil, they are infinite span. So, no cross flow is allowed, only all the flows are along the chord of the aerofoil, right. Now, when there is a finite wing; that means, this is not infinite, let us see what happens.

If it is flying at an angle like this, the pressure here is more than the pressure on the top that is why there is a lift. As I come near the tip of the wing here, what happens? There the pressure is more and pressure is less, so air will try to come from the bottom to the top and they go on forming a vortex, vortices like this, right. And since, they go as a rotational; they go into a rotational motion, the vortices are formed. So, rotational kinetic energy is required and that comes at the cost of the energy of the airplane, so we call this actually induces drag.

So, that is why in the finite wing we have wing tip vortices. If I draw the diagram, if you see this is the wing cross section I am drawing, this is the span, right. Let us say, this is the fuselage, now what is happening. Because, if there is a lift; that means pressure at the bottom they are more as compared to pressure at the top. So, what is happening at the tip, because it is high pressure, so flow will go like this and they form vortices, which draws rotational kinetic energy from the energy of the airplane and hence, energy is lost and it affects the speed, so we call it drag, because of this vortices or we call these are induced drag.

Induced drag, because of these vortices, so many times we are called vortex drag, many time they are called lift induced drag. Why lift induced? Because, these vortices are formed, because of the lift, because of the lift there is a pressure difference and because of finite wing, there is a vortices form at the tip and so it gives a vortices drag, induced drag, lift induced drag.

Note here, if I make it infinite, then this situation will not come, so it becomes 2D or it becomes an aerofoil. That is why aerofoil will never encounter conceptually any such vortex or lift induced drag, right, so that is why it is finite wing.

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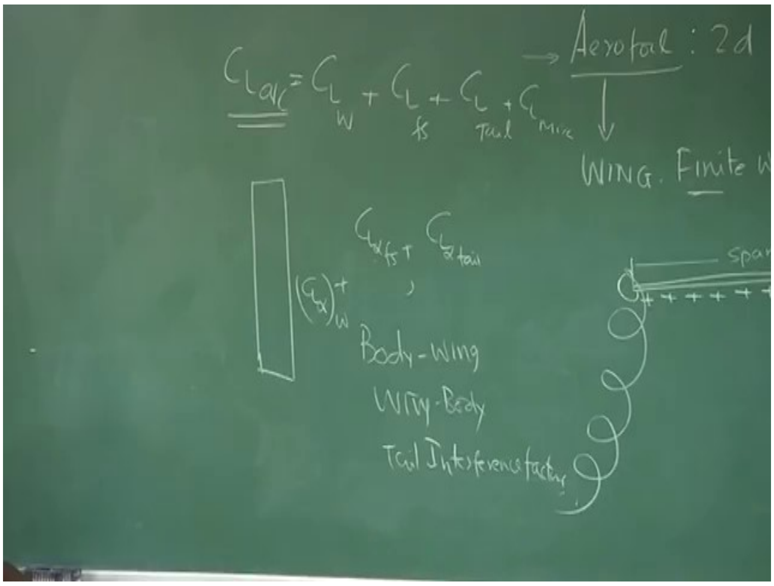

Then, we use for this aircraft, CL for the aircraft. So, we started with aerofoil where infinite span, wing, finite wing, we found there is the vortices and because of vortices, there is a drag and because of the vortices, we will find the vortices will be coming like this. So, there will be inducing a downward component of velocity to the local angle of attack at the wing or even at the tail will be changed, will be reduced, so effectiveness will go down.

So, in a language of airplane, aerodynamic modelling, we can always say that the lifting characteristic gets changed because of downwash or the CL alpha may also changed because of downwash, right. Now, for CL alpha of the aircraft or CL of the aircraft we write CL, because of wing, CL because of fuselage, CL because of tail plus CL because of miscellaneous component, that is to say.

$$C_{L_{a/c}} = C_{L_w} + C_{L_{fs}} + C_{L_{tail}} + C_{L_{misc}}$$

If I know what is the CL alpha of the wing, if I know what is the CL alpha of the fuselage, if I know what is the CL alpha of the tail and assume that, all of them are based on same reference area.

Then, if I simply add these things together, then I am expecting that, that will be CL alpha of the whole aircraft, you may not be correct. In fact, we are not correct if we are deal like that. For simple reason there are many such reasons, but one of those, see this is the CL. When I compute the CL alpha of the wing, this is the wing in isolation.

But, in the actual practice, what happens? When there is a fuselage, it gets let us say if this is the fuselage, then I am attaching it like this on the wing. So, there is a body to wing interference here, right. Similarly, if I calculate for body alone, when I bring near the wing there is a wing body, body wing interference, same thing happens with the tail.

So, when I am talking about total CL of the aircraft, I need to take care of body wing, wing-body. Similarly, for tail interference factor which modify the overall CL of the aircraft. So, that is why we use three terms distinctly.

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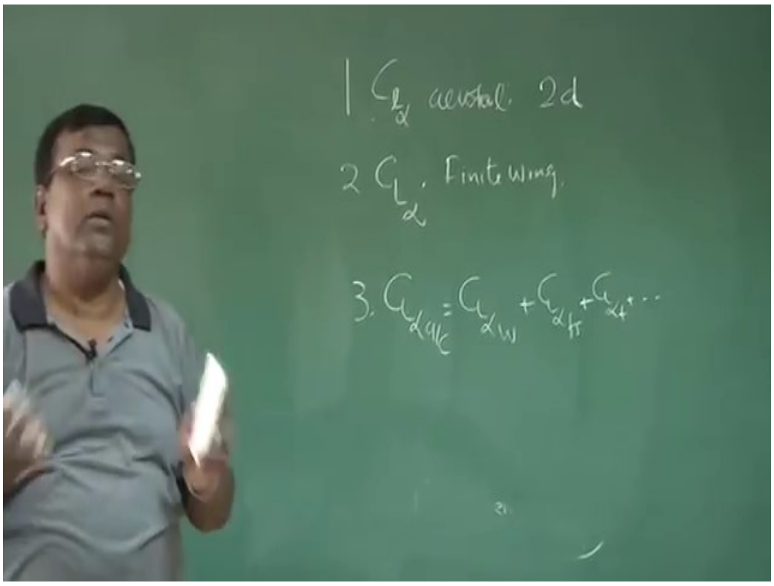

One was CL alpha of the aerofoil, understanding very clear this is 2D, 2D flow that is no cross flow like this, right. Second thing, where we use small l CL capital alpha for a finite wing, because of the finite wing there will be vortices, that will affect the drag and the lift curve slope depending upon the size and third one this is CL alpha of the whole aircraft which will be summation of sum of CL alpha of the wing plus CL alpha of the fuselage plus CL alpha of the tail plus any other components are there. But, we should be very careful that we have an appropriate be taken, no interference factors between body wing, wing body, tail body, body tail all in together, right.

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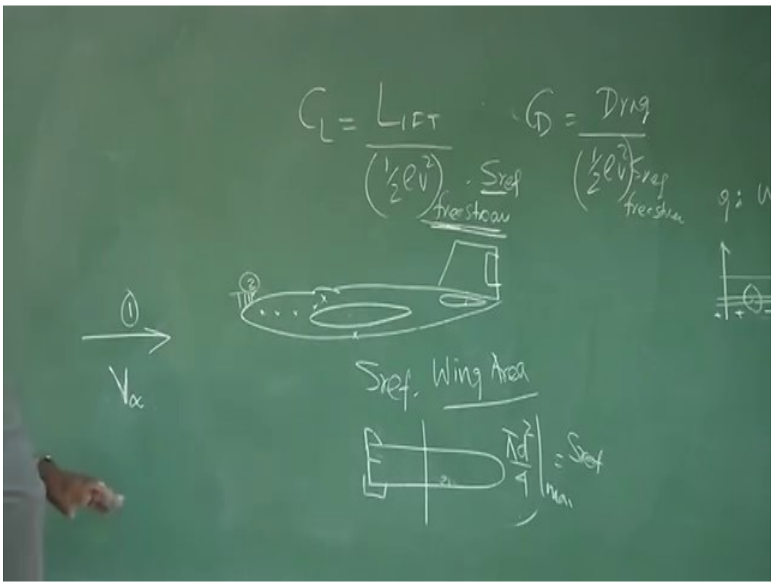

If I come back to CL again, I said this lift by half rho V^2 free stream into S reference.

$$C_L = \frac{Lift}{\left(\frac{1}{2}\rho V^2\right)_{freestream} S_{ref}}$$

Similarly, you said CD as drag by half rho V^2 S reference, again this is freestream.

$$C_D = \frac{Drag}{\left(\frac{1}{2}\rho V^2\right)_{freestream} S_{ref}}$$

We have being using the word free stream, right, I am talking about S reference, let us understand what are these things. Let us draw an aircraft and lets say, this is the relative air speed. Text book, always you see V infinity is written to explicitly mention that is the free stream condition. What is the meaning of free stream condition? We are tried to adjust that.

If I come close to these bodies, so here, here, here, here, what will happen, you see or here, for that matter. If this is V, 1 and if I take this point 2, will the velocity at point 1 and point 2 to be same or speed at point 1 and point 2 to be same. We could see very clearly, at point 2 because of the contour of this fuselage, there will be a change in speed here, because you could see at this point, again the area goes on decreasing if I take contour surface.

There is a natural tendency for the flow to accelerate. I am talking about low speed flow, right. So, local velocity and the velocity of air further away from the body are different, similarly if I come here, I come here by the time flow reaches here, there is a viscous effect on it, what about the contour effect and then, the velocity will never remain same. So, when I am try to non-dimensionalised , this lift or drag which velocity should I take?

Because, at each point the velocities are different, that is why we take the freestream velocity and the understanding is this, it is at a very far away from the body. So, that there is an influence on the speed of the freestream, so it remains constant, that is why we talk about free stream, so that we can define a non dimensional quantity consistently.

Now, S reference for aircraft, S reference is the wing area. Wing is the primary component, which produce the primary thing the lift and wing area is the S reference for aircraft and for a missile, you will find, missile is wingless missile. Just it has got like this, you may find this maximum cross-sectional area max that sometime becomes reference area. But, coming back to aircraft let us not forget when I talk about S reference, it is the wing area, ok.

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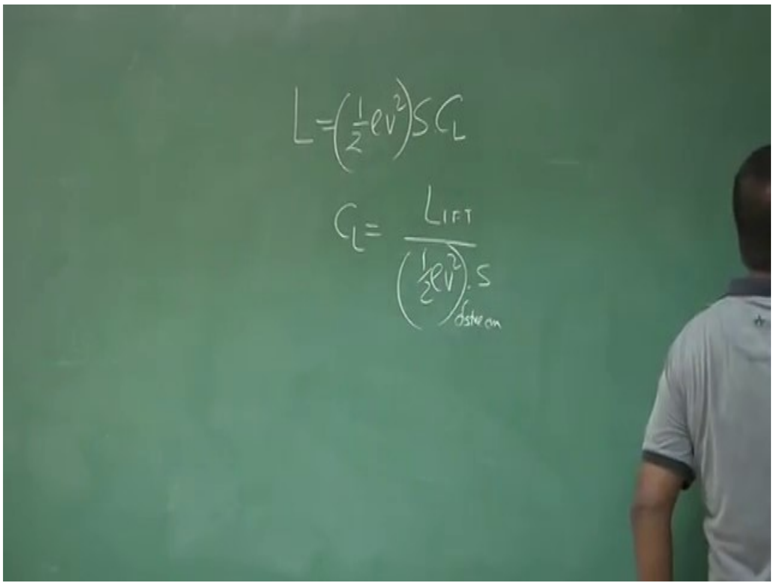

So, what we have very quickly learn that, when I try to model lift, I will model it like this half rho square S, S means S wing, V means free stream.

$$L = \left(\frac{1}{2}\rho V^2\right)S$$

Half rho V^2 is freestream dynamic pressure into CL, ok? and CL is basically lift by half rho V^2,

$$C_L = \frac{Lift}{\left(\frac{1}{2}\rho V^2\right)_{fstream} S}$$

which is the free stream into S, which is wing area.

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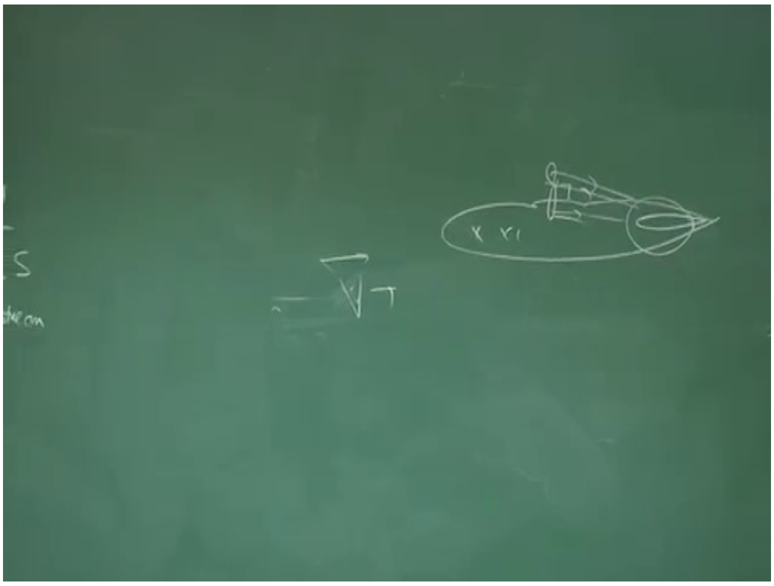

There is another observation on freestream, dynamic pressure, if I draw an airplane again, if you see a tail here and this is a V. The freestream pressure here, here, here and here could be different, because the flow, the energy might have lost. However, if I have an engine here, mounted here, this propwash can create or augment, change the freestream dynamic equation on the tail.

So, that is why we need to be very, very careful when I am talking about lift and drag, that when I try to finally, talk about overall CL of the airplane. Then, it has to be non-dimensionalised with the free stream dynamic pressure ok, but local lift could be on the local dynamic pressure.