Design of Fixed Wing Unmanned Aerial Vehicles
Dr. Subrahmanyan Sadrela
Department of Aerospace and Aeronautical Engineering
Indian Institute of Technology – Kanpur

Lecture - 09
Interpreting Airfoil Data, Cl vs Alpha and Drag Polar, Selection of Airfoil

Good morning friends. Welcome back. In our previous lecture, we were discussing about how Cl alpha of an airfoil and wing differs. When we say wing, we know it is of finite aspect ratio.

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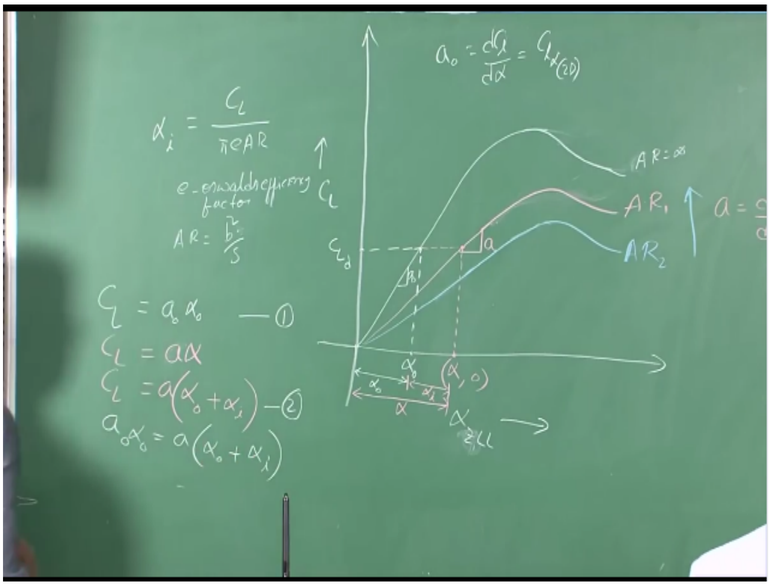

So we have derived a relationship between the CL alpha of airfoil and CL alpha of wing. Let us say this is your zero lift line, angle of attack is defined with respect to the zero lift line of an aircraft and say this is your y axis represents your CL variation and alpha is along the x axis here right. This is for aspect ratio infinite right say this curve represents the variation of lift coefficient with angle of attack for an airfoil right.

Now let us say this is for a finite aspect ratio wing which is AR1, so this represents the CL alpha curve for wing with aspect ratio 2 right, with aspect ratio AR2 right. Now say this is your increasing order of aspect ratio okay. Now let us assume we have used the same airfoil to build a wing right and what should be my angle of attack for airfoil to achieve a particular CL and what should be the corresponding angle of attack for.

Let us say if I want to achieve a particular CL what should be the angle of attack that I need to trim an aircraft for an airfoil and for that of a wing right? So let us say this is your required CL say design CL or CL required. So for this particular curve say I require this angle of attack which is alpha 0, so alpha 0 represents the angle of attack required by this airfoil to achieve this particular CL right.

So this is your alpha 0 right. Now let a0 be the CL alpha curve for this airfoil where a0 is=dCl/d alpha of airfoil that is Cl alpha which is 2D. Now let us assume we have built a wing using this particular airfoil without any geometric twist and aerodynamic twist right. Now what happens if I want to achieve the same CL here right, I need to trim the aircraft at an angle of attack right.

Let us say this be alpha right, this point represents alpha, 0 right so this point represents 0, CL design okay. Now for a wing, we have an increased angle of attack compared to that of an airfoil. This increment is due to the induced angle of attack which we have witnessed during our previous lecture alpha i right where this alpha i according to lifting-line theory is=CL/pi e AR where e is the Oswald's efficiency factor and AR is the aspect ratio b square/S, both are non-dimensional parameters here right.

Now the CL design of the CL you can achieve by airfoil since it is a straight line you can assume this is in the linear domain of angle of attack or the variation of CL with angle of attack is linear up to this say whatever the CL that we have considered is in the linear domain okay. So now from the straight line equation we have y=mx+c that Cl is=C0 here y intercept since we are defining this angle of attack with respect to zero lift line.

So a0 is the slope of this curve times the corresponding x alpha 0 here. So this is achieved by airfoil right. The same Cl I can achieve by the wing say a is the lift curve slope of this particular airfoil of particular wing, let a be the lift curve slope which is 3D lift curve slope where a =dCl/d alpha is CL alpha 3D. So capital CL here represents 3D lift curve slope CL alpha whereas small Cl represents for 2D Cl alpha okay.

So this = slope of this wing lift curve slope of this wing*by the corresponding x that is alpha here right. So this alpha is alpha 0+alpha i right, so say this is my equation 1 and equation 2 right. Now comparing 1 and 2 since we are producing that same design CL by using the airfoil and wing, so the lift is same in neither the case, so from this equation 1 and 2 what we have is a0*alpha 0=a*alpha 0+alpha i okay.

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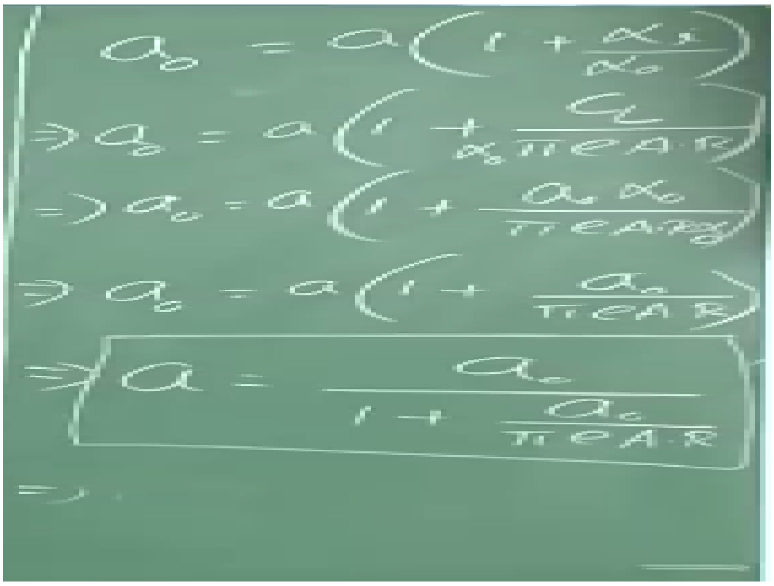

From here a0 is=a*1+alpha i/alpha 0 right. This implies a0 is=a*1+what is alpha i? Here is CL/pi e AR right. This=a0=a*1+what is CL? So we can achieve CL from this airfoil by trimming it at a particular alpha 0 right. So that means the CL that I can achieve here is by a0*alpha 0 from the airfoil*alpha 0.

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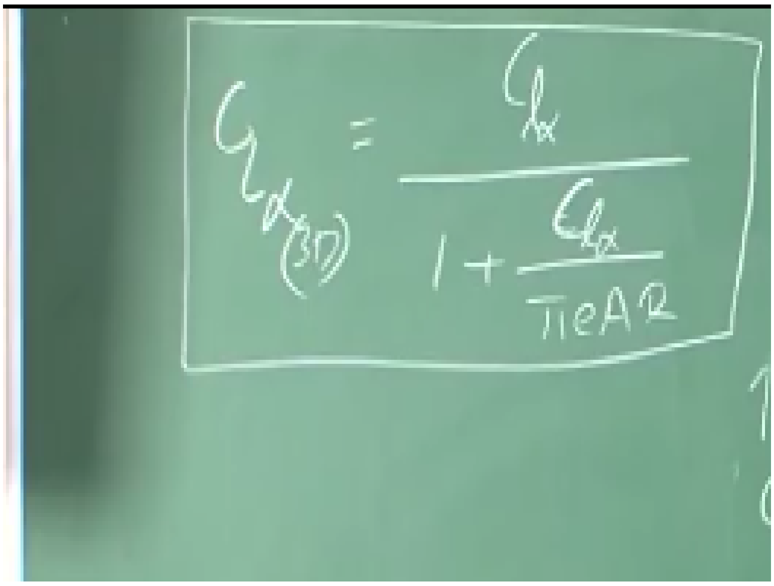

This implies a0 is=a*1+a0/pi e AR, a=a0/1+a0/pi e AR which is CL alpha 3D is=Cl alpha 2D/1+Cl alpha of airfoil/pi e AR. So this is one valid in the subsonic flight results right. This is the relationship between wing CL alpha and airfoil Cl alpha. Now let us look at some of these airfoils and their lifting characteristics right. So this data that we are going to present is obtained from the wind tunnel test.

So the reference that you can consider for this is Theory of Wing Sections by Abbott.

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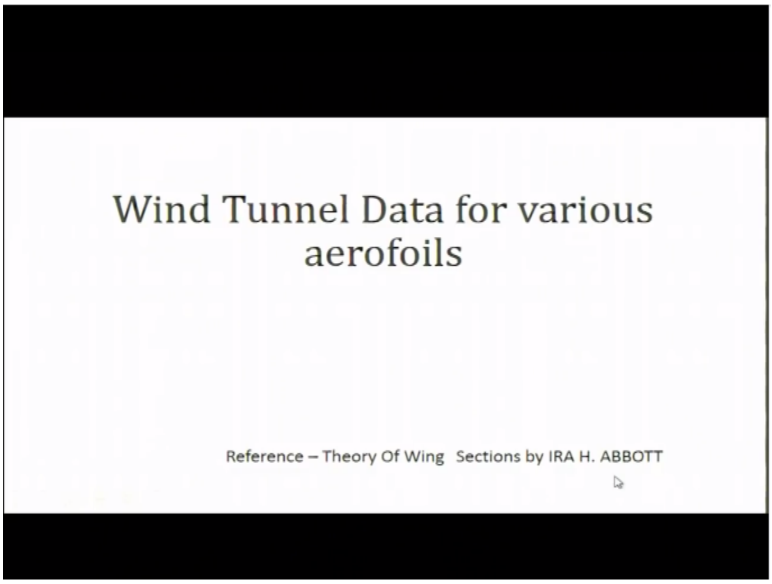

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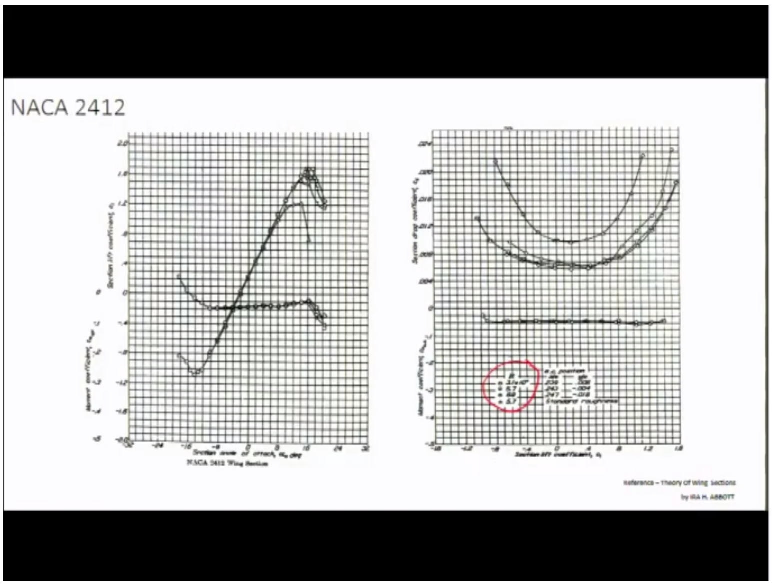

So the first airfoil here we also discussed this during our earlier example. It is a NACA 2412 airfoil. Now the question is how to calculate CL alpha right. How this data is obtained? You can see here, so this is obtained for different Reynolds number which means assume that you have used the same airfoil but you have performed this test at different velocities inside a wind tunnel.

So let us dedicate one lecture to figure out how to measure this CL, Cl, CD and Cm from wind tunnel test right or say how a wind tunnel test is performed, how these measurements were taken for airfoil as well as wing. Now here let us consider any one of this Reynolds number say it is 3 million Reynolds number here in the first, so the corresponding marker here is they are circle.

So we need to follow the circle data to achieve the corresponding characteristics at 3 million Reynolds number right. So now the question is how to find what Cl0, Cl alpha, alpha max, alpha stall, Cl max right, alpha at which CL is 0.

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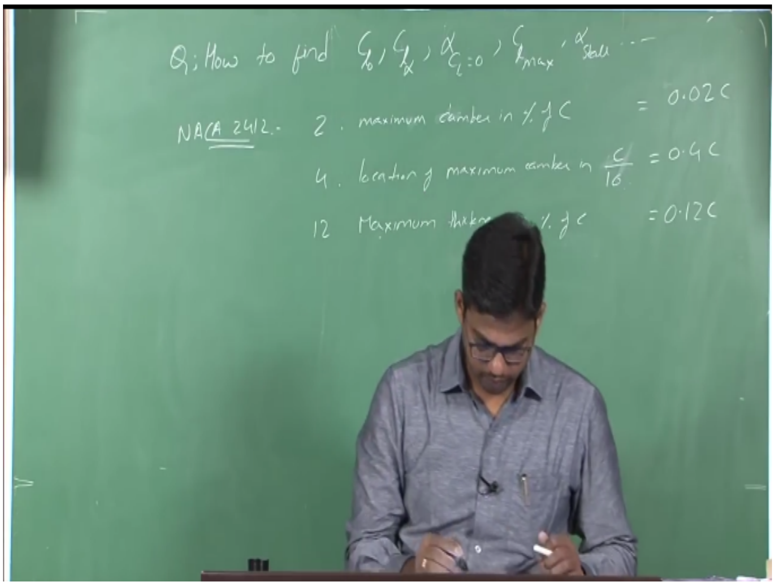

How to find Cl0 of airfoil, Cl alpha of airfoil and alpha at which CL is 0 and Cl max of airfoil and alpha stall and so on right. So given the data let us see how to get these parameters. Now coming back to this 2412 data. We know what is 2412 right, just a quick recap. The 2 represents the maximum camber, maximum camber in percentage of chord right, 4 represents location of maximum camber in tens of chord that is c/10 and 12 represents maximum thickness in percentage of chord.

See so this is=to maximum thickness location is maximum thickness is 0.2 c for this particular airfoil and corresponding location is 0.4 c and the maximum thickness is 0.12 c which is 12% here. Now what is Cl0? Cl at which alpha is 0.

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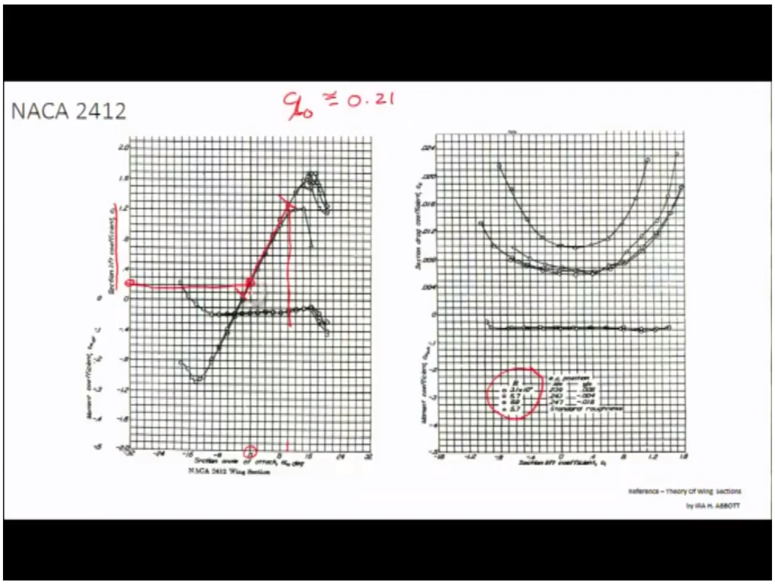

Now let us look at the angle of attack at which so this is your zero angle of attack. You can see here, now let us take the corresponding value of Cl that should be your Cl0, so the corresponding value is something here which say so this y axis represents the section lift coefficient here Cl right. So this is your 0 Cl and there is an increment to 0.4 here that you can see so till that increment there are 4 steps which means this each line corresponds to 0.1 Cl here.

So the corresponding Cl0 from here we can notice it as Cl0 so which is 2-dimensional, Cl0 is 0.21 approximately right and say how to find out the Cl alpha here right. So we will find out Cl alpha while solving some of these examples okay. Now let us say how far this Cl alpha is linear, so for this particular curve which is achieved by performing this test at 3 million Reynolds number.

So up to here so up to here it looks pretty linear which is approximately 10 degrees angle of attack right, up to 10 degrees angle of attack this curve looks pretty linear which you can model it as Cl=Cl0+Cl alpha*alpha right. So this particular regime you can consider the linear variation of Cl with angle of attack and now we have to look at alpha at which CL is 0. This is your origin say 0, 0. I am sorry.

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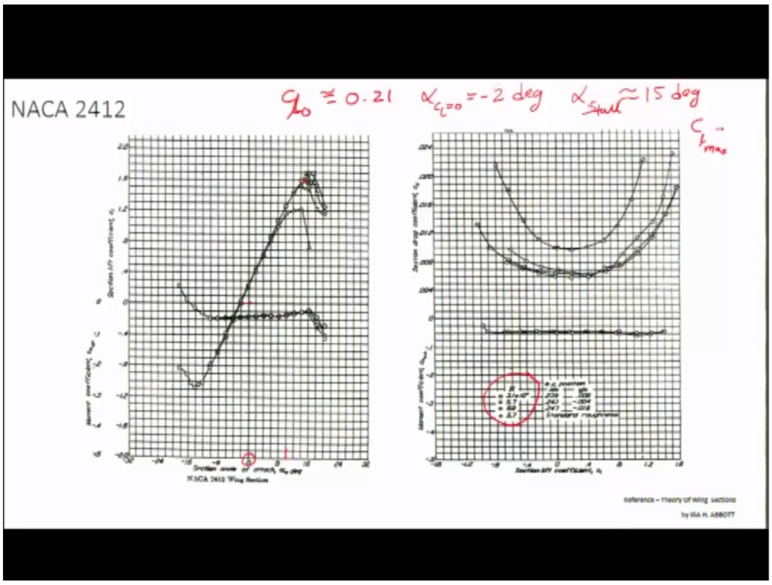

Now this is your origin, now you need to find out alpha at which CL is 0 that means this is the point where CL is 0 right, this is the point where CL is 0. So the corresponding angle of attack for this is approximately -2 degrees right so alpha at which Cl = 0 is -2 degrees. This particular airfoil stalls at an angle of attack approximately 15 degrees right. Alpha stall is approximately 15 degrees and the corresponding value of Cl is 1.6 so Cl 2D max is 1.6.

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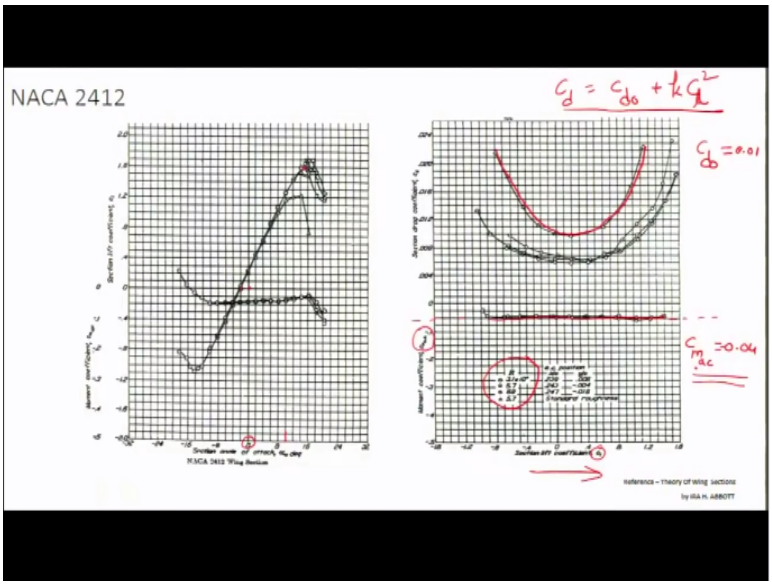

Cd=Cd0+k Cl square right. Now this particular plot represents the variation of Cd with Cl, see it is almost parabolic variation. So this particular equation or this particular formulation will definitely fit this data Cd whatever the data wind tunnel data that you have taken here. Now what is Cd0? The profile drag here which is at Cl is=0. So what is Cl0 here? So this is your Cl0 and the corresponding drag coefficient is 0.1 approximately.

So Cd0 of this airfoil is 0.01, see this is the moment about aerodynamic center, variation of moment about aerodynamic center with Cl right although the lift changes, when the lift changes when the angle of attack changes lift coefficient is changing when the angle of attack changes right. So here this we can infer as the angle of attack changes and as the Cl changes, the pitching moment almost remain constant.

See it is a constant value now, although there is a variation in Cl that value of this pitching moment remain constant at about C/4. This is the corresponding aerodynamic center for this is C/4, so this moment is about C/4, quarter quad of that airfoil. So the value of this is about -0.04, so Cm about ac of airfoil is 0.04 right, -0.04. Please make a correction it is -0.04 which means it is a positively cambered airfoil. So for positively cambered airfoils, the Cm ac is negative right.

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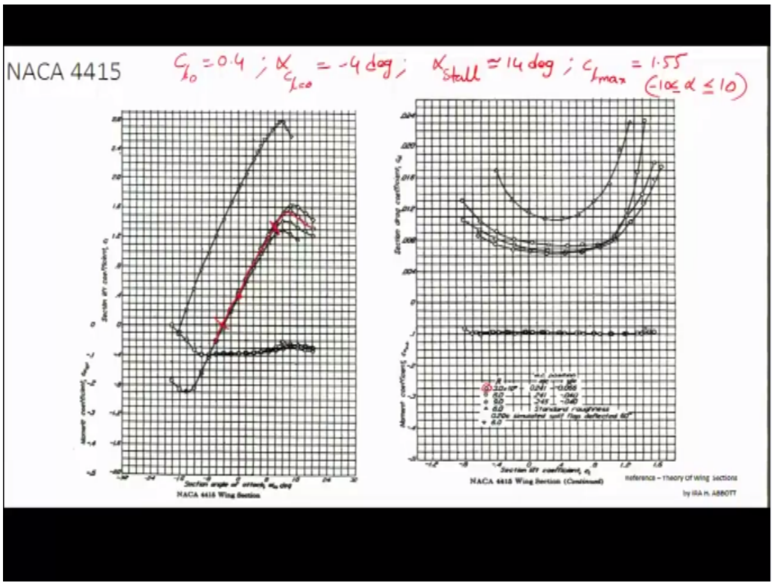

Now let us go to the next data set which is 4415. It is again a four series airfoil, so you can see here so this is the variation of Cl, Cm with alpha and here to the right you have Cl, Cd, Cm variation with Cl right. Here what will be the Cl0 for this airfoil, Cl0 of this airfoil is 0.4. So let us look at the next airfoil which is also a four digit NACA 4415 airfoil right.

So we know the maximum camber is 0.4 c here and the location of 0.04 c and the location of maximum camber is about 40% of the chord and the maximum thickness is 15% of the chord right. Now for this airfoil what will be my Cl0? So we need to select first the Reynolds number of our interest. Let us say 3 million is our current interest, 3 million Reynolds number is our current interest.

And the corresponding Cl at which alpha is 0 which is our Cl0 is approximately 0.4. So this 0.3 million corresponds to this data that is plotted with circular marker. So this corresponding Cl0 is 0.4. You know previous example and say what is alpha at which Cl is=0 for this airfoil? So this is approximately -4 degrees. This is the point at which the curve touches the x axis which is like Cl is=0 and the corresponding angle is -4 degrees.

And we have to figure out what is the Cl max of this. Now can we identify the linear regime of angle of attack? So for this circular marker the one which is above here, the one this is the Cl versus alpha data at 3 million Reynolds number right. So here we can observe this is linear up to say this particular point beyond which there is a slightest nonlinearity and then stall occurs is alpha of 15 degrees, 14 degrees say you can say.

Alpha stall is approximately 14 degrees and this is your point for Cl0 that is 0.4 alpha. You have Cl max which in this case is 1.55 around right and what is the linear range of this Cl variation with angle of attack? So from 0 if you can observe from this plot, it is varying from -2 in fact or -4 degrees to say approximately 10 degrees. So alpha even for -8 degrees and -10 degrees, it is still behaves linear here -12 degrees it still behaves linear, so <or= -10 let us say alpha 10 and -10, these are the boundaries of linear regime of angle of attack right.

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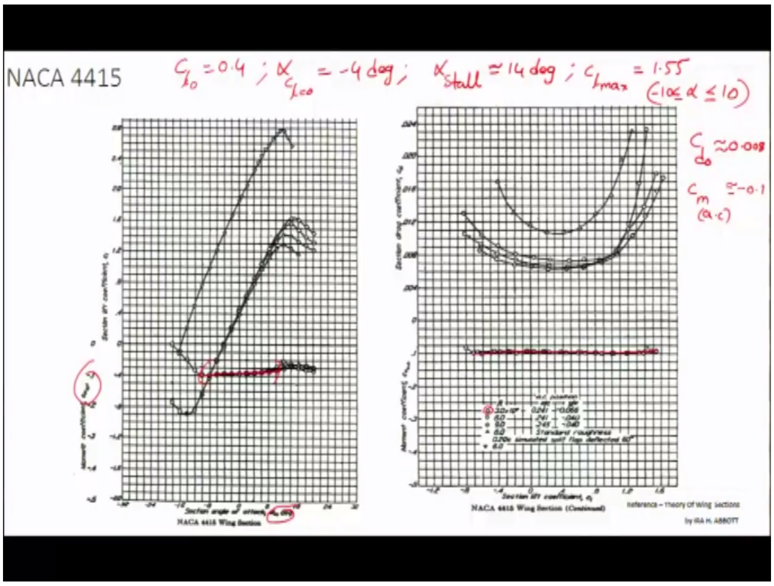

And the corresponding Cd value for this is Cd0 if you want to find Cd0, the Cl is 0 at this particular location. So this is approximately for 3 million Reynolds number it is 0.0079 or 8 that is CD0 is approximately 0.008 for an airfoil right and what is the Cm about aerodynamic center? Cm about ac is=-0.1 approximately right. So you can see here with angle of attack, in the linear regime of angle of attack the Cm almost remains constant with angle of attack right in this linear domain.

So the aerodynamic center above we can consider this moment reference point is the aerodynamic center, why because the pitching moment is remained almost constant with angle of attack in this particular regime. This is the Cm variation with angle of attack.

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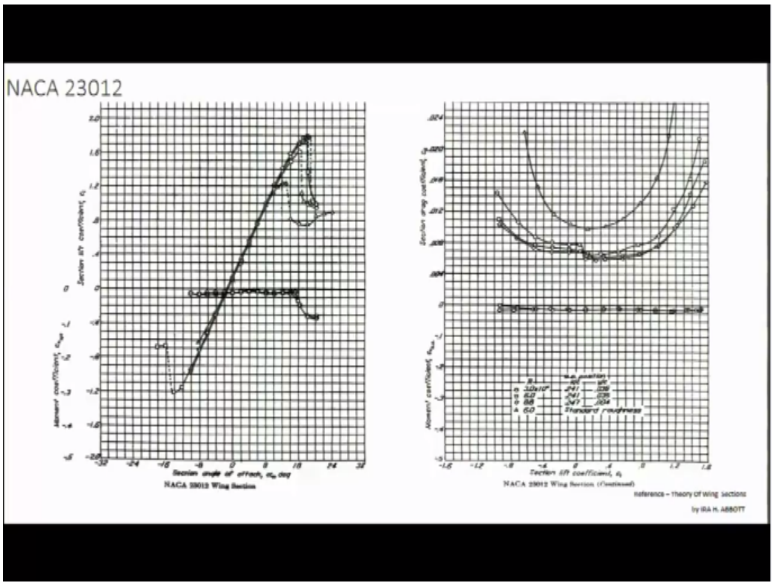

Similarly, there is a five series airfoil 23012, you can also find Cl0, Cl alpha, so the Cl0 for this airfoil is about 0.1 and Cl alpha is about 6.45, will figure out how to find Cl alpha when solving some examples. The alpha stall in this case is about 15 degrees and Cl max is 1.5 and the corresponding Cl max is 1.5 at alpha stall of 15 degrees and Cm about aerodynamic center is -0.0125.

So we are talking about 3 million Reynolds number, data at 3 million Reynolds number and Cd0 is 0.07 and alpha at which Cl becomes 0 is -2 degrees in this.

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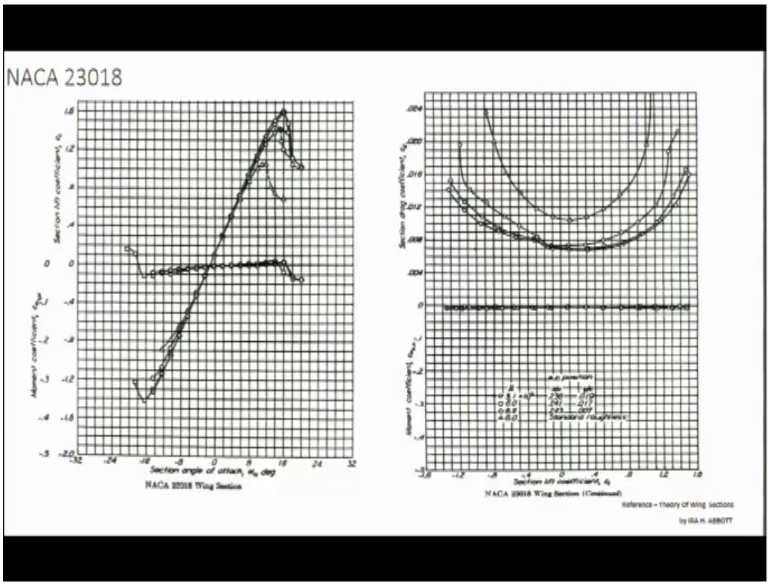

And there is another five series airfoil 23018. You can see the pitching moment is constant almost constant, Cm variation with Cl is constant here if you can see and the corresponding value is almost 0 here 0.01 say 0.01. The Cd0 value for this particular 23018 airfoil is approximately 0.007 which is almost same as 23012 and 23018. So although there is an increase in the thickness ratio 2/c but there is no increase in drag coefficient, profile drag coefficient.

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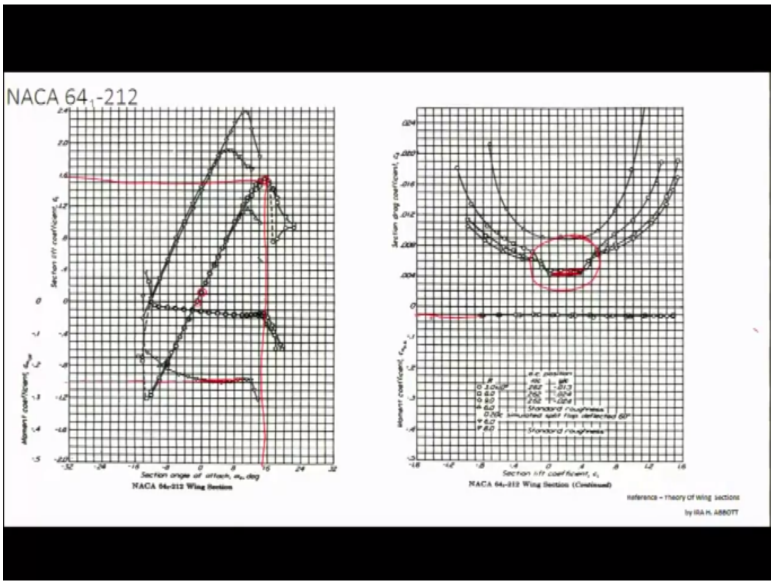

Now comes the six series airfoil. See there is a huge difference, the main thing that we can notice is that drag bucket. This is what we discussed in our earlier lectures. So in our previous case where the Cd versus Cl if you look at the drag polar plot, there is no significant drag bucket, you cannot witness a significant although for some portion it may remain constant almost constant but not exactly the same as what we have witnessing at six series airfoil right.

So right from here we do not see a significant drag bucket but for the six series airfoils you can see the Cd almost remains constant with variation of Cl or angle of attack here and similarly you can find out for this 641-212 airfoil, you have Cl0 is 0.1 and alpha at which Cl it becomes 0 is -1 degree and Cl max is approximately 1.5 and the alpha stall is 15 degrees, this is 1.5 and corresponding alpha stall here is approximately 15 degrees right.

And Cm about ac is -0.025, so the value that you get Cm at different alpha is this is Cm at alpha 0 right. This is Cm at Cl0.

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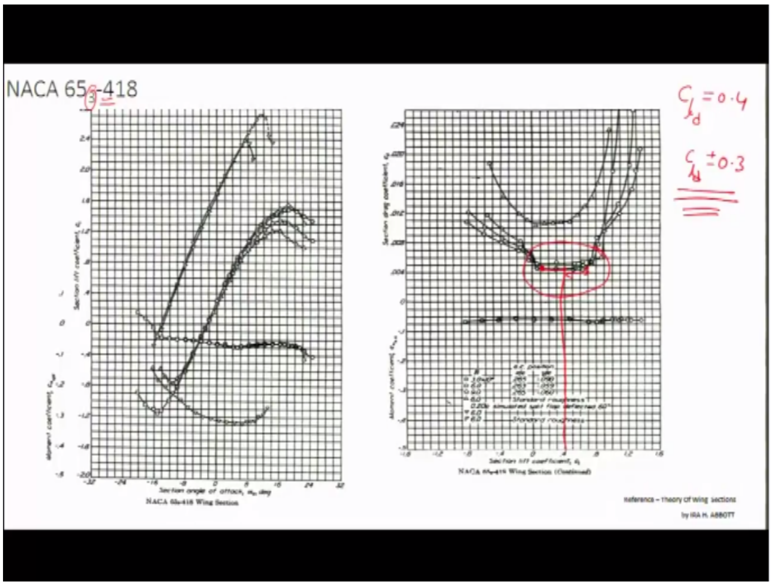

Now similar drag bucket we can also observe with 653-418 airfoil. Now let us look at this significance of this subscript 3 here right. Now from the nomenclature 65418 we can find out what is the design Cl for this particular airfoil which is Cl design of this airfoil is 0.4 right, this 4 represents in tens of I mean 4*1/10 represents a corresponding design Cl of this airfoil.

Now let us say take this 0.4 here, consider 0.4 Cl and say now the subscript 3 represents the regime in which Cd remains constant right for about this Cl design which is + or -0.3 of the Cl design, so in this regime Cd will almost remain constant. Let us see whether it is true or not. So 0.4-0.3 is 0.1 approximately, so up to this point the Cd is almost flat that is Cl design -0.3 and then say Cl design+0.3 is 0.7.

So we can also consider this regime as a constant CD regime right, so this is a significance of this drag bucket here right that happens, there is a significant drag bucket for 6 series airfoil and the nomenclature will help you to select the corresponding airfoil based upon your Cl design. Let us take up few examples to find the relationship between wing CL alpha and airfoil Cl alpha right.

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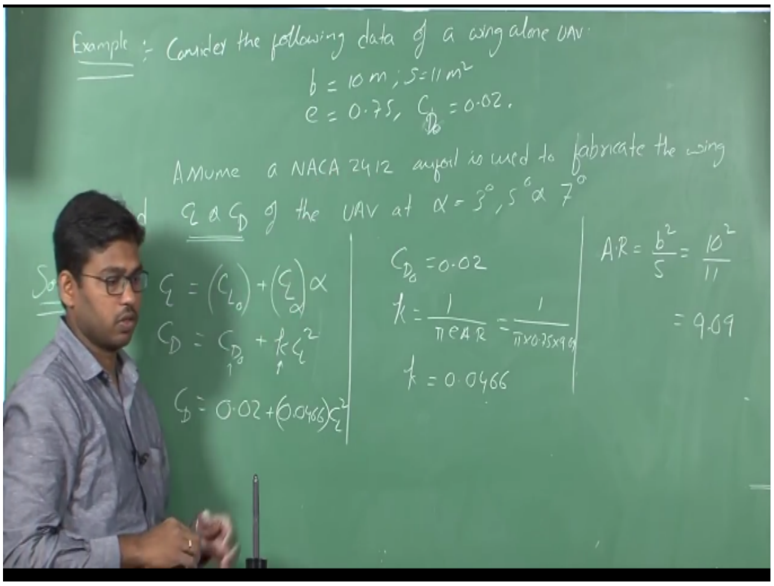

Let us take the first example. Consider the following data of a wing alone UAV. So there is some dimensional data is given here which is b is 10 meters, planform area is 11 meter square and the Oswald's efficiency is 0.75 and CD0 of the wing total UAV is given as 0.02 right. Assume a NACA 2412 airfoil is used to develop this UAV or since it is a wing alone we can say develop this wing.

Find CL, CD of the UAV at alpha is=3 degrees, 5 degrees and 7 degrees. When you say CL, CD here is for the entire UAV right and now how to solve this question? So CL is=CL0+CL alpha*alpha of the wing right and what is CD, once you have CL if you CD0+k CL square right, you can just drag polar of the wing and you can find the corresponding CD at this particular CL right.

So what is this CD0 is given right, so CD0 is given as 0.02 and k, can we find k here? Induced drag coefficient factor 1/pi e AR right. So we know we have span and planform area, we need to find what is the corresponding aspect ratio. What is aspect ratio here? AR=b square/S=10 square/11 that is 9.09 right. So 1/pi e AR is so 1/pi*0.75*9.09 that is 0.0466 right is the value of k.

Now you know what is k, you know what is CD0. What is CD0 here? 0.02+0.0466*CL square. Now for different values of CL you will get the corresponding CD, so when do you get different values of CL when you have different angle of attacks say 3 degrees, 5 degrees, and 7 degrees here but you need to find out what is the CL alpha of this wing and CL0 of this wing right. For that the information given is this is made out of 2412 NACA 2412 airfoil.

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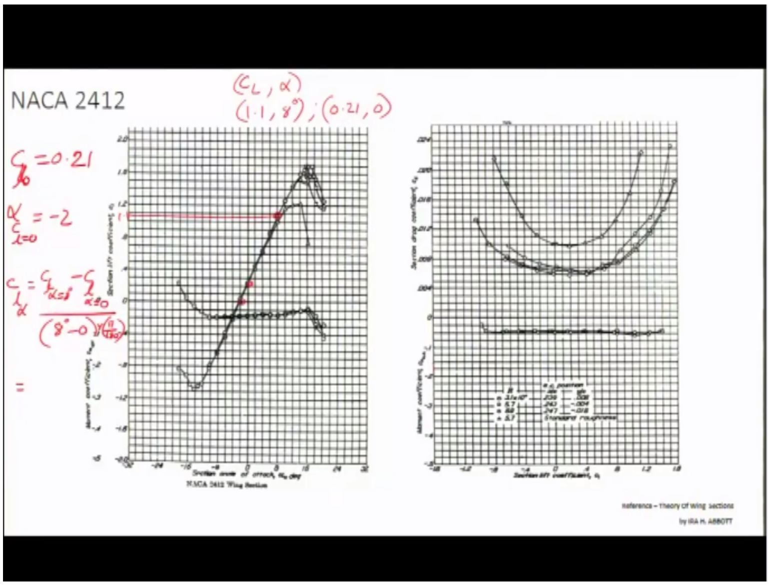

So if you go back to NACA 2412 airfoil here, what is CL0 of this airfoil which is 0.21, so Cl0 I can write down here as Cl0 of airfoil is 0.21 right and alpha at which Cl=0 is -2 degrees which is this point this particular point, this point we have Cl0 and what is CL alpha of this plot say this plot we can safely assume a linear till 8 degrees angle of attack right. So Cl alpha 2D is=Cl at alpha is=8 degrees-Cl at alpha is=0/8 degrees-0*pi/180, you will get it in radians right.

So what is Cl at 8 degrees? Cl at 8 degrees is 1.1 approximately, so this particular point is 1.1, so CL, alpha you have at alpha is=8 degrees what you have is 1.1 and at alpha is=0 degrees what you have is 0.21, 0 so by using this since it is a straight linear equation you can find the slope of this curve right, y2-y1/x2-x1.

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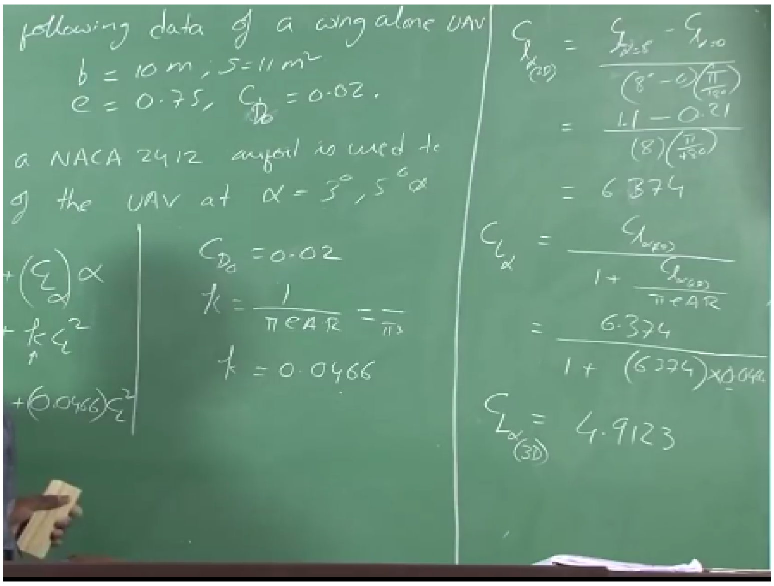

Cl alpha 2D of this airfoil is=Cl value at alpha is=8 degrees-Cl value at alpha is=0/8 degrees, so per degrees I am converting into per radians. This is alpha is=8 degrees you have 1.1 approximately -0.21/8*pi/180. So this turns out to be 6.374 right. This is your Cl alpha 2D and other important yes once you have Cl alpha 2D, you can find out CL alpha 3D. CL alpha of this wing is=Cl alpha 2D/1+Cl alpha 2D/pi e AR=6.374/1+6.374*1/pi e AR is 0.0466 = 6.374 is 4.9123.

So CL alpha per radian of this wing entire wing is 4.9123 you have CL alpha here 4.9123 per radian right. Now you need to find out what is CL0 to find out CL at different alpha as well as CD at different alpha right. So how to find CL0? So the assumption that we make here is since we are making this wing out of this airfoil right.

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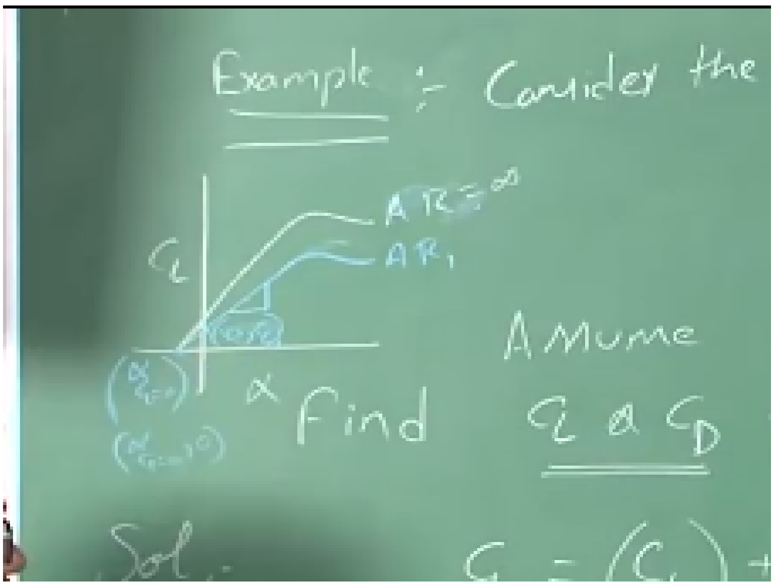

So we have Cl versus alpha plot right this is for say airfoil okay and we are making this wing out of this airfoil, the assumption here is the alpha at which Cl is 0 for airfoil will remain same for the wing as well right. So this is the point at which alpha Cl=0 will remains same for wing as well as airfoil, this is the assumption.

So this is the alpha at which Cl=0 so this remains same for infinite wing and finite aspect ratio wing as well as infinite aspect ratio wing right okay. Now with this assumption, we know what is CL alpha of this curve so we got CL alpha 3D and you know this point we need to find this point, so this particular point is CL alpha at which CL=0, 0 and this particular point is CL0, alpha is 0, CL0.

So you have two points, you know the slope like you do not know one of the point here, you can straight away find this CL0 right.

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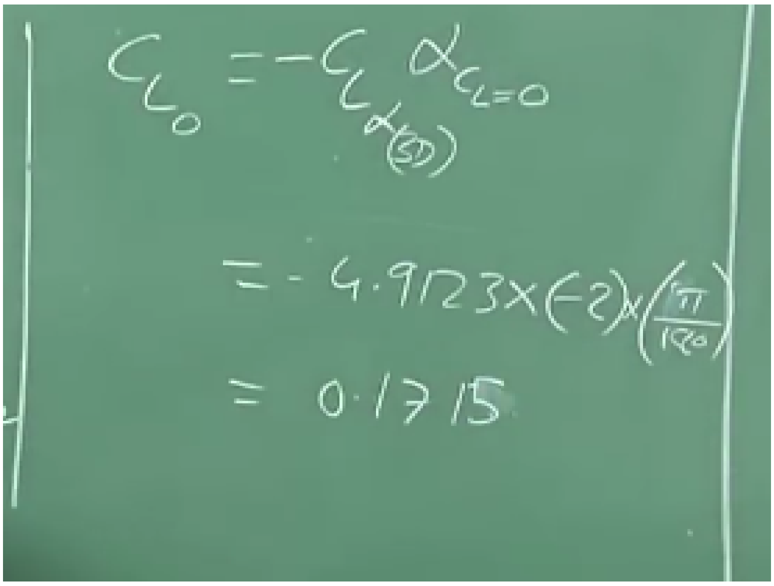

So this CL0 is=CL0 3D is=CL alpha-CL alpha 3D*alpha at which CL becomes 0 right that is what you will end up solving that equation right. So what is CL alpha 3D here is 4.9123*alpha at which Cl is=0. So from the airfoil data we can find alpha at which Cl is=0 is -2 degrees right. So -2 is it right? We considered CL alpha as 4.9123 per radian so you need to convert this alpha from degrees to radians. So this is 2*pi/180=0.1715.

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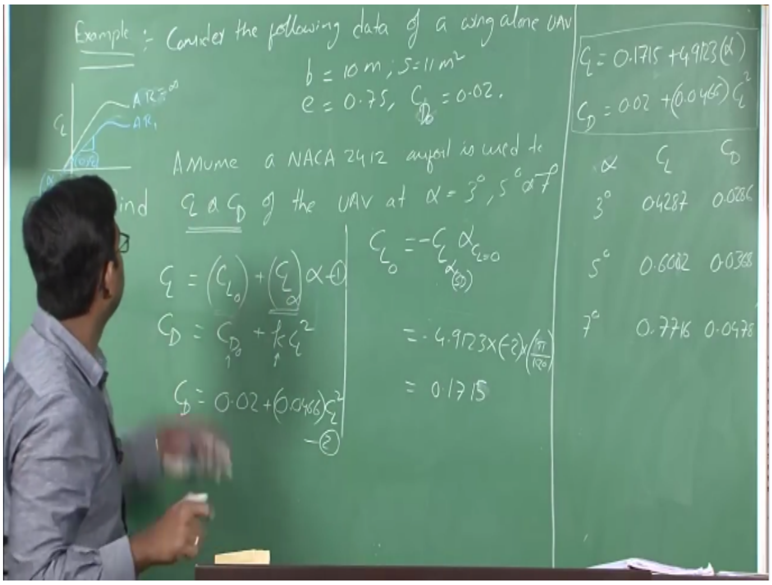

So now substitute this in this equation 1 say this is your 1 and say this is your 2 right, so from equation 1 what we have CL0 is 0.1715+CL alpha is 4.9123*alpha right. So alpha should be in radians here and CD is=0.02+0.0466*corresponding CL square. Now we need to find this at 3 different trim conditions that is alpha for 3 degrees, alpha for 5 degrees, alpha for 7 degrees.

Now if you substitute alpha 3 degrees what you end, so say this is alpha, this is CL and the corresponding CD, say write alpha 3 degrees what you have CL is 0.4287 is the CL and the corresponding CD is 0.0286 right. So at 5 degrees the CL value is 0.6002 and CD value is 0.0368 and alpha at 7 degrees CL is obtained as 0.7716 and CD is 0.0478. So all these values are obtained by using these two equations right.

So what we have done, given a wing planform geometry and Oswald’s efficiency factor as well as profile drag coefficient, so we figured out what are the CL alpha of this wing and CD of the I mean CL0 of this wing and then we calculated what is the corresponding lift coefficient at different angles of attack as well as the drag coefficient at different angles of attack. So it is given that this wing is made out of NACA 2412 airfoil right.

So from this airfoil data we have figured out what is the Cl alpha of this airfoil and alpha at which Cl=0 for this. So using those two, we found out CL0 of the wing, initially we figured out what is CL alpha of this wing and then the CL0 of this wing again right. So substitute those two parameters in this equation, we will be able to find out the corresponding CL and CD at different angles of attack.

Now why we need to find CL and CD in the first case? Now we will appreciate as we progress, once we start discussing about the what should be the lift or design CL right to sustain at particular weight at a given altitude right, how to design the wing that means how to select the planform geometry, during that time you need to have a design CL, what should be your design CL right.

So when we discuss those concepts, you can appreciate this particular exercise and say which power plant you need to select that requires the information of CD right. So during various phases, we need to move at different velocities and different flight conditions. So which power plant to be used, how do you find? You first gather the information about what is the requirement of the system.

The requirement of the system is drag, so to calculate drag you need to find what is the corresponding CD.

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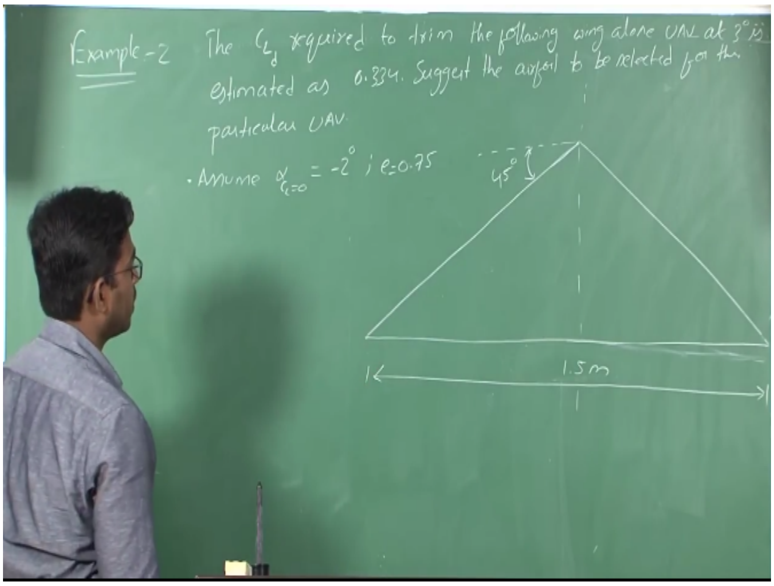

Now let us consider this example 2. The CL design required to trim the following wing alone UAV at 3 degrees is estimated as 0.334. Suggest the airfoil to be selected for this particular UAV right and it is also given alpha at which Cl is=0 is -2 degrees and Oswald's efficiency is 0.75. Now this is the planform that has been given, the UAV is with the triangular planform here right.

So we know what is the span of this and leading edge sweep of 45 degrees is given right, with this information we need to find out, we need to suggest an airfoil which means how do you suggest an airfoil?

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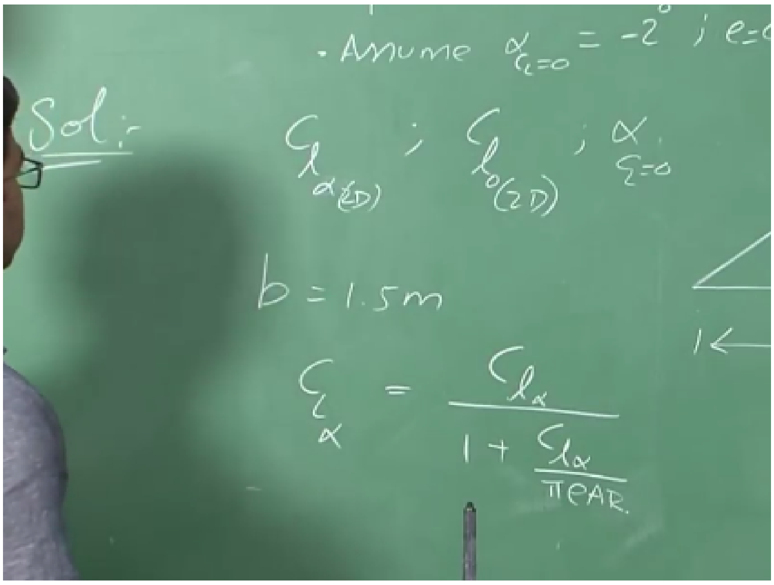

In terms of lift and drag characteristics right, say lift characteristics here. We need to give what is Cl alpha 2D of that airfoil should be and say Cl0 of that airfoil right, Cl0 2D and say alpha at which Cl=0 right. This is already given because for wing as well as airfoil it will remain same here that is the assumption that we have okay. Now first figure out what is given here right. The span is 1.5 meters right.

What is the relationship between CL alpha 3D and Cl alpha 2D? CL alpha of wing is=Cl alpha of airfoil/1+Cl alpha of airfoil/pi e AR. So what you need is Cl alpha 2D right. How to find Cl alpha 2D?

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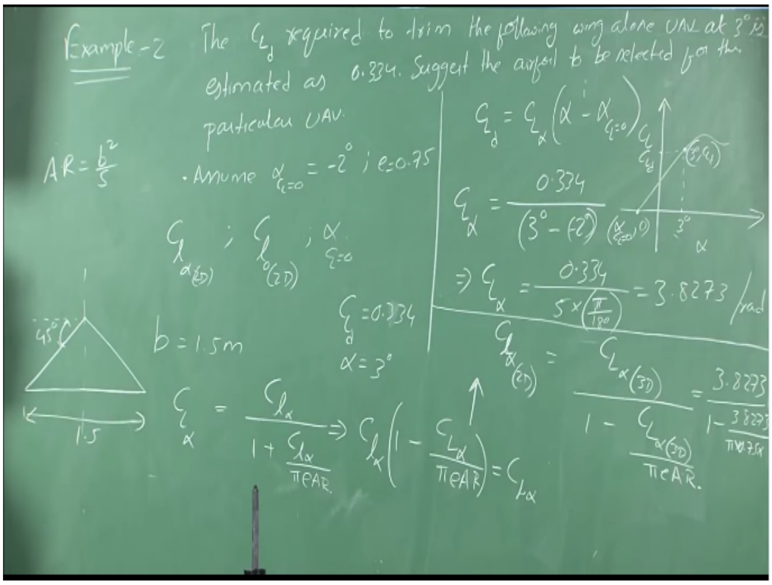

Planform right with the span of 1.5 meters and leading edge sweep of 45 degrees and with the leading edge sweep of 45 degrees. Now I need to find out what is Cl alpha 2D right of the airfoil from the wing data. So how to get wing data here in the first case? So CL design=CL alpha*alpha trim-alpha at which CL=0 can we do this? What does this equation mean? Assume that this is the Cl versus alpha of your aircraft or the wing right.

This is your Cl versus alpha. The information that is given is CL design, CL design is given as 0.334 and this design is achieved at angle of 3 degrees at an angle of attack of 3 degrees, say at 3 degrees this is your CL design. That means you know this point right and you have the information about this alpha at which CL=0 right this point corresponds to alpha at CL=0, 0 you know this point that is 3 degrees, CL design right.

From here you can find out what is CL alpha slope of this curve right. So CL alpha of the wing is=CL design is 0.334/3- -2 degrees right this implies CL alpha of the wing is 0.334/5*pi/180 which is in radians. So here it is in degrees, it is per degrees here it is per radians which is=3.8273 per radian right. Now you have CL alpha 3D. Now can we find out Cl alpha 2D from here. So how to do this? Can we manipulate this equation?

This implies Cl alpha*1-CL alpha/pi e AR is=CL alpha 3D right. If you take this to your left hand side or multiply this equation by this denominator right and then taking this second term to the left side, to the right hand side so what you have here is so this implies Cl alpha 2D is=CL alpha 3D/1-CL alpha 3D/pi e AR. Now substituting these values of CL alpha 3D is

3.8273 and 1-3.8273/pi * what is Oswald's efficiency factor 0.75 * we need to find what is the aspect ratio of this wing.

So how to find aspect ratio of this wing? We know aspect ratio AR =b square/S right. So we have the span, we need to find the area of this particular planform.

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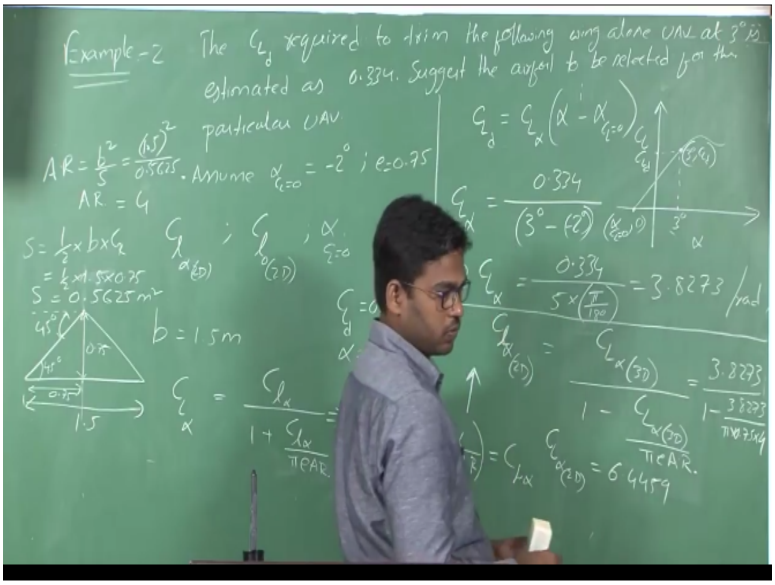

So how to find the area? 1/2 base*height 1/2 span*say CR root chord here, so what is the root chord here, so when say this angle is 45 degrees this is also 45 degrees, you know this span, you know this, you know this semi span which is 0.75 right. So tan 45 is opposite by adjacent, so this particular height will also become 0.75, tan 45 is 1 right. This particular height of this or the root chord of a triangular wing is 0.75.

So you have 1/2 base * height, you can simply substitute 1.5*0.75. So S is 0.5625-meter square right. So the corresponding aspect ratio is 1.5-meter square/0.5625. This is 4, the aspect ratio turns out to be 4. Now substituting this aspect ratio in this equation, you can find out what is Cl alpha 2D. Cl alpha 2D is 6.4459. Now we are able to find what is the Cl alpha 2D of this airfoil. Now we need to find what is the Cl alpha, Cl0 of this airfoil right.

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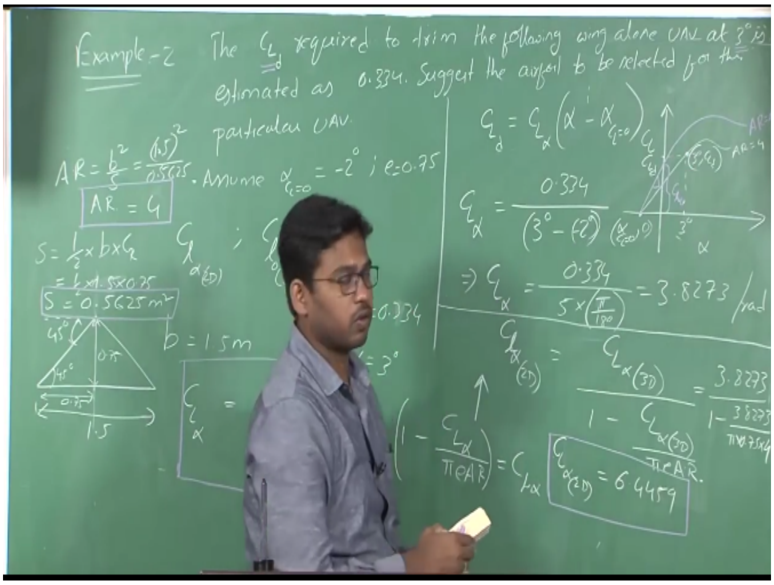

Say this is my airfoil data right. Aspect ratio is infinite, here we have aspect ratio is=4 so this one. So now we need to find out this y intercept of this airfoil. We know the slope of this airfoil, just now we have figured it out as 6.4459. So we need aspect ratio to calculate that CL alpha, so we have figured out area from the given geometry is the relationship we have started with 2D Cl alpha and 3D CL alpha.

So we figured out 3D CL alpha based upon the design CL as well as the angle of attack which this aircraft has to be trimmed or that CL design has to be achieved. Now we have 2D Cl alpha right, you know the slope, you have this point alpha at which Cl is=0 right which remains same for airfoil as well as wing that is the assumption. Now using the slope and this we can find out what is the Cl0 of that.

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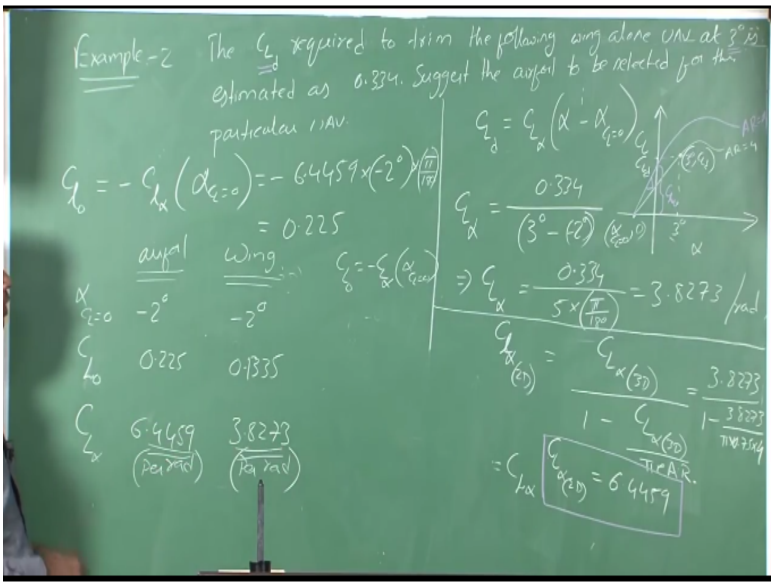

So Cl0 of airfoil is=-Cl alpha of airfoil*alpha at which Cl is=0 which is -6.4459*- -2 degrees, -2 degrees is given, is given in the question. So this is in degrees that the slope is in per radian so I need to convert it as I need to multiply this by pi/180=0.225 right. Cl0 is 0.225 for an airfoil and let us see what is alpha at CL=0 and Cl0 of airfoil or say Cl0 and Cl alpha for airfoil and wing.

So for airfoil it is -2 degrees and yeah here it will be same. What is Cl0 of airfoil is 0.225. What is the Cl0 of the wing? So we can find out right that CL0 is=-CL alpha*alpha at which Cl=0, you substitute the wing CL alpha you will get the corresponding CL as 0.1335 right and Cl alpha of the airfoil is 6.4459, of the wing is 3.8273 these are per radian, per radian. See lower the aspect ratio lower is the CL alpha of your wing, you can observe that right here.

The same airfoil is used to build a wing which is of lesser aspect ratio wing, so the aspect ratio is about 4 here. So the CL alpha drastically reduced right. Now say if I increase the aspect ratio.

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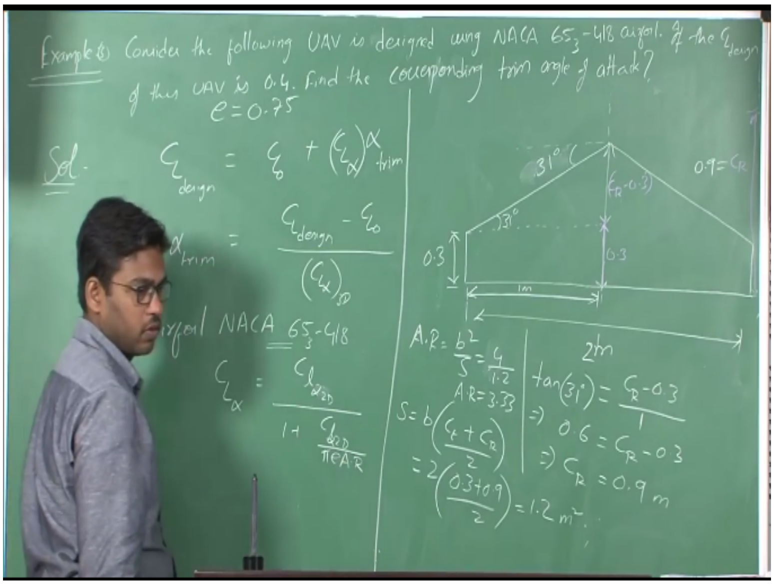

Now let us consider example 3 here. Consider the following UAV is designed using NACA 653-418 airfoil if the CL design of this UAV is 0.4 find the corresponding trim angle of attack? So the planform of this UAV is given which is of 2-meter span and 0.3 meters tip chord and the leading edge sweep of 31 degrees right so we need to find the trim angle of attack right first.

See if the CL design is known then we can express the CL of this wing as Cl0+CL alpha of the wing*alpha trim. So this alpha trim is=CL design-CL0/CL alpha which is 3D of wing and this CL0 is also for wing right. So what are unknown here? So CL design is given which is 0.4 here and we need to find out what is CL alpha 3D and CL0 of this wing.

So another information given is airfoil right. Regarding airfoil which is used here is NACA 653-418 right. Now we need to find out in order to find out CL alpha 3D, we need to know what is the airfoil Cl alpha okay so as well as the aspect ratio and Oswald's efficiency. Say the Oswald's efficiency is about 0.75 right. So we need to find the aspect ratio of this wing. First how to find the aspect ratio?

We know AR is=b square/S, so we need to find what is area here. So to find out area I need the root chord right, I need to know what is this root chord. So how to find this root chord? So say this is say up to this it is 0.3 right, say this entire thing is CR, the entire chord at the root is CR, so this is like CR-0.3 right. Now I know leading edge sweep of 31 degrees, I know the span here which is 1 meter right.

So we know aspect ratio=b square/S right, so to find this aspect ratio I have to know the area here, to find this area I need CR here. So tan 31 degrees is=CR-0.3/1, so this implies tan 31 is approximately 0.6 is CR-0.3. This implies root chord is 0.9 meters. So CR here is 0.9 right. Once I have CR, I can find out what is the area of this wing. What is area of this wing? b*Ct+CR/2 which is=otherwise b/2*CR*1+lambda.

So b is 2-meter span*0.3+0.9/2 which is 1.2-meter square right. So substituting this span 2 meter square and 1.2-meter square area will get to know what is the corresponding aspect ratio. So aspect ratio here is 4/1.2=3.3, so aspect ratio again is a low aspect ratio wing here, it is a low aspect ratio wing UAV, so you know what is aspect ratio. Now what you need to know is Cl alpha 2D. This is Cl alpha 2D or say let us go back to this airfoil data NACA 653-418 right.

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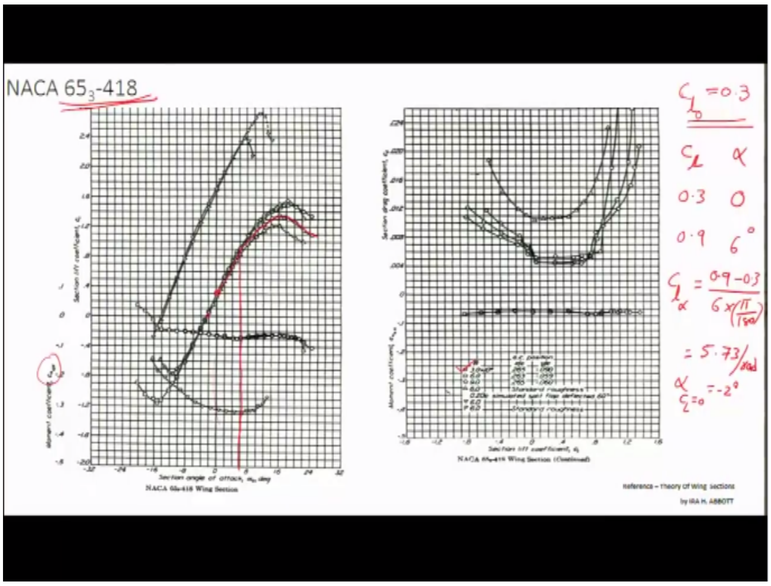

So this is the airfoil, so we can see here the variation of lift coefficient with angle of attack as well as pitching moment coefficient with angle of attack and this particular plot represents the drag polar and this is Cm versus Cl about c/4 or the aerodynamic center here. This is about c/4 okay. Now what we need to find out is CL alpha of this wing first. So let us first consider what is Cl at alpha is=0 that is Cl0 is 0.3. See this is Cl at which alpha is=0 right.

So this particular point CL and alpha for 0 degrees, this is 0.3. Now how far we can consider this linear here? So let us say if we look at this curve, so we are talking about this curve right, 3 million Reynolds number right. So it is with circular marker and see if you consider that curve here how far this is linear? So here about this point so maybe up to this point you may consider this as a linear curve right, straight line.

So the corresponding angle of attack is approximately 6 degrees because beyond which there is a change in slope right, it becomes nonlinear. Now at this 6 degrees angle of attack, the corresponding Cl value is 0.9. So how to find out the CL alpha of this curve? y2-y1/x2-x1 that is Cl at alpha is=6 degrees is 0.9-0.3/6*pi/180. So what is the answer for this? 5.73 per radian right.

Now you got to know what is CL alpha of this plot and let us also look at alpha at which Cl is=0. So this particular point is the alpha at which the CL is=0 which is approximately 2 degrees right, 2, 4, 6, 8, got it. It is approximately -2, alpha at which CL is=0=-2 degrees. Let us take this information and find out what is the CL alpha 3D which is the CL alpha of wing made out of this airfoil.

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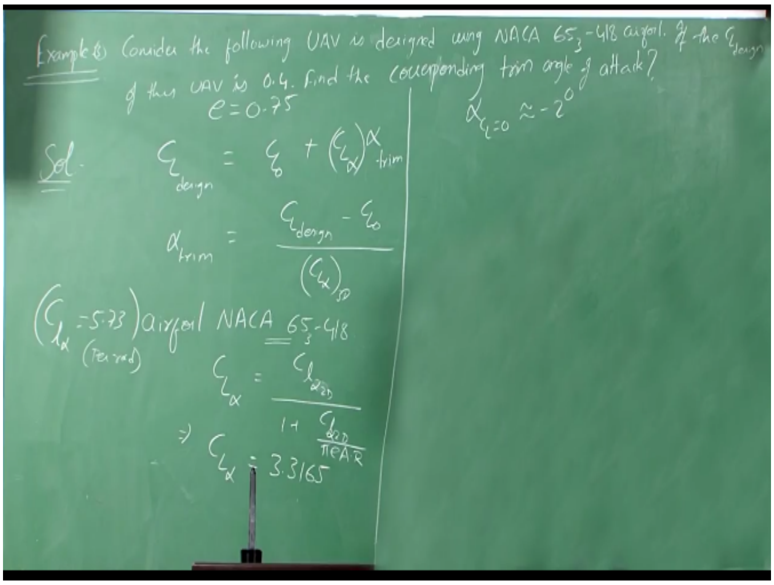

Now from here we got to know using this NACA 653-418 Cl alpha of this airfoil is 5.73 per radian, this is the Cl alpha of this airfoil. Now substituting this in this equation what you will get CL alpha 3D is approximately 3.3165 right. So to calculate alpha trim, we need another variable here CL0 right or another parameter CL0 that we need to find out. So CL alpha is 3.3165. Now how to find out CL0 of this wing?

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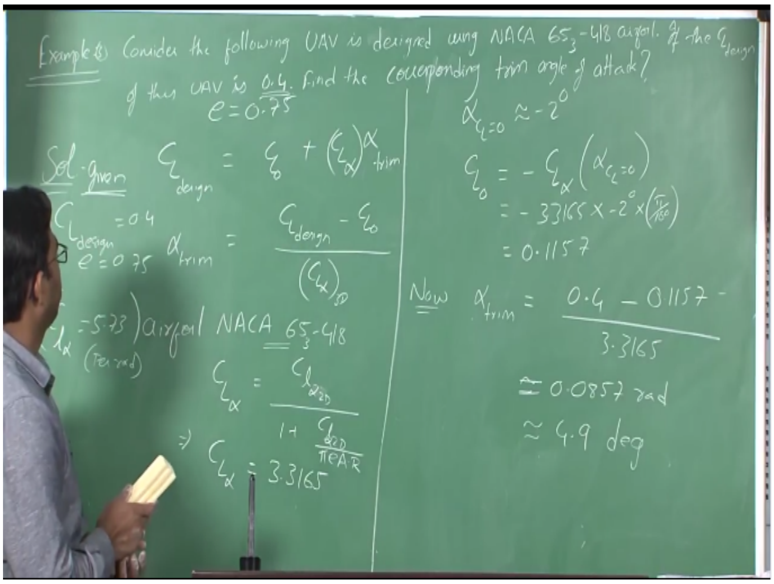

Again from the airfoil data, we have alpha at which CL is=0 is approximately -2 degrees, alpha at which CL is=0 is approximately -2 degrees which will remain same for wing as well as the airfoil okay. Now CL0 of the wing is –CL alpha of wing*alpha at which CL=0, this=-3.3165*-2 degrees*pi/180 because this slope is wing per radian right. So the corresponding value of CL0 is 0.1157.

Now alpha trim can be estimated from this equation as CL design is CL design given is 0.4 that is CL design given is 0.4 and e is 0.75 here, so this is like 0.4-CL0 which is 0.1157/CL alpha of this wing which is 3.3165=0.0857 radians, so in degrees it is approximately 4.9 degrees right. So this is the angle of attack with which the aircraft will be trimmed to attain this particular CL right.