Design of Fixed Wing Unmanned Aerial Vehicles
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Lecture – 20
Tutorial 1

Hello everyone. Myself Qazi Salahudden, TA of this course called design of fixed wing UAV. Today we will discuss some assignments problem. I hope you got the assignment 1, 2, 3 and 4. So today we will discuss about the assignments problems. Later we have made some more problems. So in next to 2 days, we will discuss some more problems. So let us start with the assignment 1.

In question 1, you have to answer the following question. The unmanned aircraft can be classified based on, the first option was size of the unmanned aircraft, weight of the unmanned aircraft and third one was the mode of operation and fourth one is the all of these. So by seeing the question, you can directly say that... And second question was the endurance of the aircraft depends upon the...

So first option was payload carrying capacity of the unmanned aircraft. So obviously you that if suppose that payload of the aircraft is in gradient, then total weight of the aircraft will increase. So obviously it will be dependent parameter and aerodynamics of the unmanned aircraft. Already you have seen, sir thought the aerodynamic efficiency L/D ratio. So this one also depends upon the endurance of the unmanned aircraft.

And third one was the atmospheric condition. So you know at lower altitude like sea level, density is 1.2256 kg/m cube. When you go above the sea level, the density will decrease. So the endurance will depend upon the speed and speed is related to the density. And in the third question, the variation of the properties in the standard atmosphere is calculated based on the assumptions.

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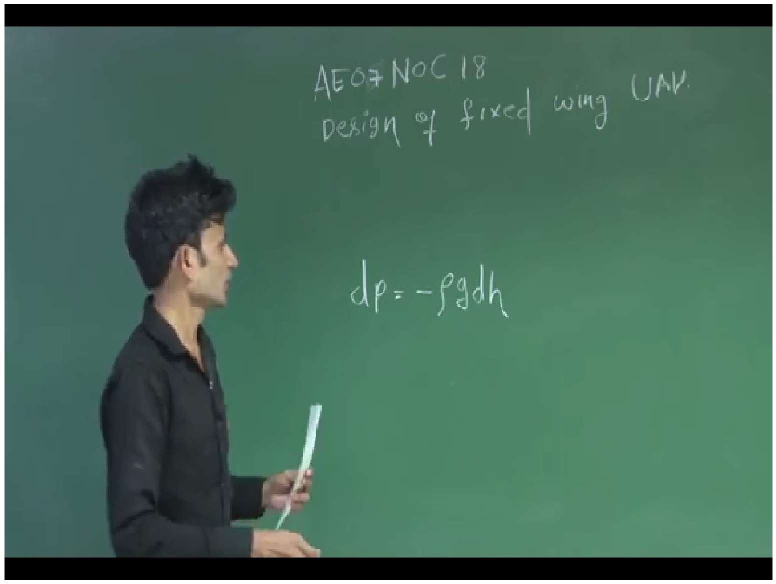

So here is the assumption, is taken G=constant because if you wrote this equation dp=-rho gdh. So this is the hydrostatic equation based on then you derive the p2/p1 ratio t2/t1 ratio and rho 2/rho 1 ratio. In this equation, g assumes to be constant because if you did not assume the g=constant means here will be the 2 variable, g also will be the variable and h also will be the variable.

So this is the multi-integration. So it will be the complicative one. So that is why we assume that g=constant. And we also know that about 0 to 30 km or 32 km, the variation of g is almost negligible. It is 9.81. So only some 2% or 3% change will be there. So based on that you can also say that g is a constant. So if you take g=constant, it will be very easy to derive the pressure ratio temperature ratio and density ratio.

And in fourth question, the question was the high altitude long endurance unmanned air vehicle is flying at an altitude of 17 km of geopotential altitude. When you say that the geopotential altitude, it means you are getting the altitude by keeping the g=constant. So you have to find the density at that particular altitude. So the altitude is given 17 km. So you can use the density ratio formula.

So you know that from 0 to 11 km, the variation will be different and from 11 to 20 km, the variation will be different. So first you have to calculate the density at the 11 km. Because suppose that, the question is find the altitude at the 10 km, then the difference will be the density at the sea level. But here the question is you have to find the density at 17 km. So you cannot take the reference as the sea level. First you have to find the density at the 11 km. You have to take that difference and then you will go to the density at the 17 km altitude.

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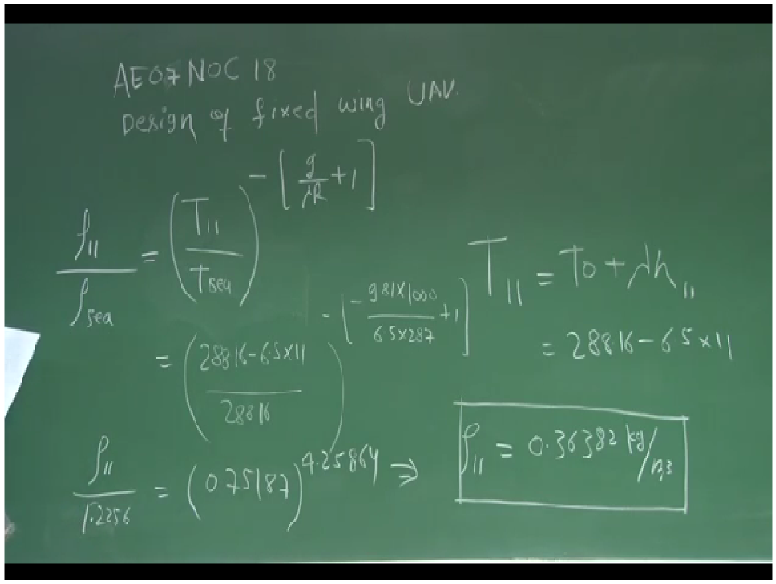

So let us see, we will solve this problem. So you know suppose that this is the density at 11 km and density at sea level. So this will be nothing but temperature at the 11 km, temperature at the sea level. So already derived these things in previous lecture, g/lambda R+1. So R is the gas constant, 287; lambda is the lapse rate which depends upon in which layer you are calculating this density ratio or pressure ratio.

So again this temperature you can find using this relation T11=T0-lambda h. If you want to find the temperature at 11 km, you have to put the altitude 11 km. Lambda is -6.5 and T0 is the sea level temperature 288.16. So if you calculate these things, this will come around 288.16-6.5*11. So I am putting this value directly into this equation. So it will be the easy one-time calculation. So 288.16-6.5*11.

This will be the temperature at 11 km/temperature at sea level 288.16 and the index will be -9.81*1000. 1000 we are multiplying because of, this is in K/km and g unit is m/s square. So you have to convert all these into SI units. 6.5*287+1 and - will be there. So these things will come out 0.75187 and this value will come out to be 4.25864. So this will be the density ratio 11 km/sea level.

So sea level density you know is 1.2256 kg/m cube. So when you solve this, finally you will get the density at 11 km will come out 0.36382 kg/m cube. So this was the very straightforward question but you have to keep in mind that you cannot find the density at 17 km using that, using this formula. So first you have calculated this 11 km. Using this you will go to the 17 km. So generally people do the mistake, directly here they put the T17 and directly they used to find the temperature at 17 km.

If you do this, then it will be the wrong. So that is only you have to take care one thing. You cannot find the density directly; temperature, density or this pressure and density direct at 17 km. First you go to the 11 km, then you will find that. So we have the value rho 11 km. So next we will go to the rho 17 km.

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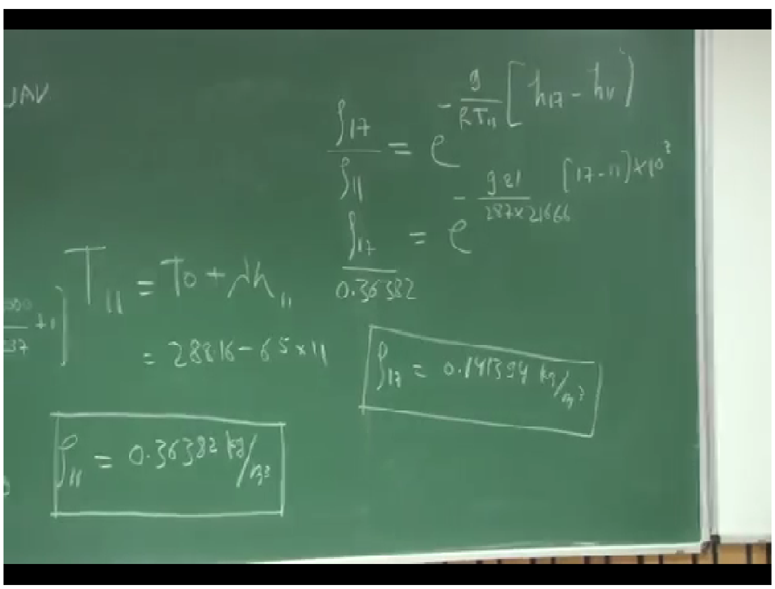

So you know that from 11 to 17 or 11 to 20 or 11 to 22, this formula you have to use. In this region, the temperature will be the constant. So temperature ratio and pressure ratio will be the same and this will be gRT11. You have to take the temperature at 11 km and h 17 km and h 11 km. Now you are going from 11 to 17. So you know the value of rho 11 km, here you have got. This will be 0.36382 and you know this value 9.81, 287 and temperature at 11 km 216.66 and this is 17-11 km.

Then you have to multiply by 10 power 3 to convert it to meter. So rho 17 you can calculate using this. So this will come out 0.141394 kg/m cube. This will be the your final answer. So by solving this, you have to take care of the 2 parts. You cannot calculate the density at 17 km directly. Second one things like the approximately solution. So during the intermediate, you have to take at least 5 digits, so that finally you will get the accurate answer.

If you take 2 digits, then this will give the error also and finally if you put that then the error will come. So your approx solution will not be the approximate. So at least you take 5 digits. And in the fifth question, the high altitude long endurance unmanned air vehicle is flying at an altitude of 15 km of geopotential altitude, the true speed in m/s at that altitude will be nearest 2. So here you have, here the question was you have to find the true speed.

So true speed is basically at what speed the, really the aircraft is flying with respect to the ground. So generally, suppose that you are flying at sea level. So density is different. When you will go above the sea level, then density will be different. So at the same power, you will get more speed.

So that is called true speed. So how this density plays the major important role for the true speed, we will see using this example problem. Generally, in true speed, you can do the mistake like you can put the density at sea level then that will not be the true speed. That will be the equivalent speed. So we will show you the difference between these 2 speeds using the next problem.

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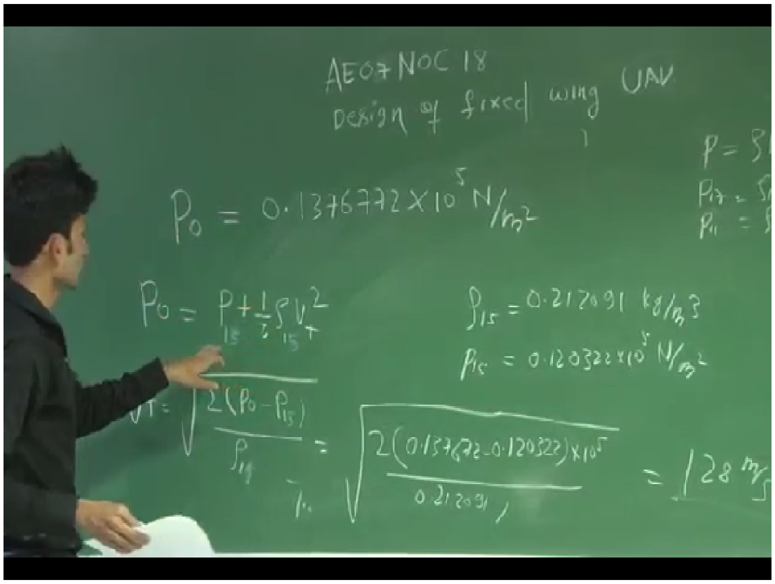

And in fifth question, some additional data is also there which is total pressure sense by pitot tube which is 0.1376772*10 power 5 N/m square. Suppose that aircraft flying is at particular speed. So in pitot tube, if you bring the flow to 0 isentropically, isentropically means you are not adding any heat and you are not removing heat from the system. So if you put the flow isentropically 0, you will get one pressure that is P0.

So if you are not adding any heat and if you are not removing any heat during this process, then you can write this relation P0=P+1/2rho V square. So you have P0 and you have to find the speed at 17 km, so you have to find this true value, pressure and density. So this will be the 17 km and this will be the 17 km. So using standard formula, I have shown 1 example to you how to find the density. So density you know at the 17 km, what will be the density. And density ratio and pressure ratio will be the same, right.

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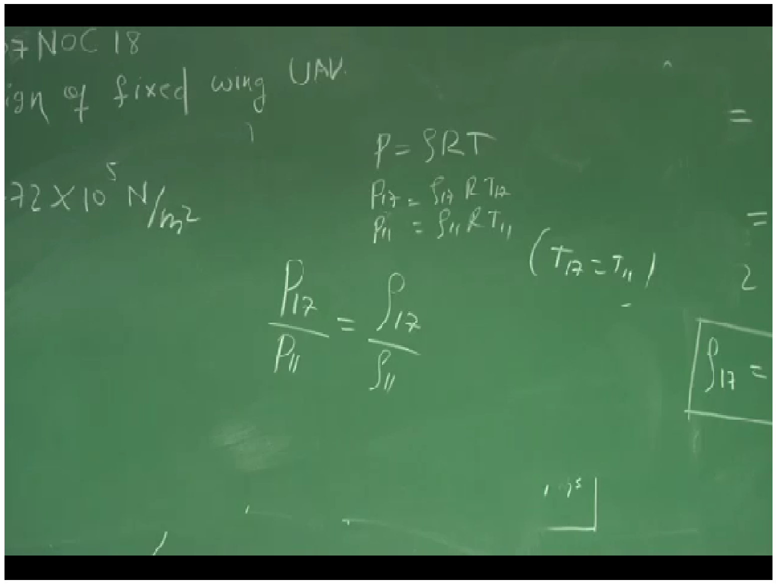

P17/P11=rho 17/rho 11. This will be the same. Because from 11 to 17 km, temperature will be the constant. So if you wrote P=rho RT and if you make 2 equation from this equation like P17=rho 17 RT17 and P11=rho 11 RT11. If you divide these 2 equations, then this T17 and T11 will be the same. So you will get this. So you know this ratio too from the previous question. You know the P11.

You can find the P11. So using this, you can find the static pressure at the 17 km. So from if you manipulate this equation, you will get VT=under root 2 P0-P at 17 km/rho at 17 km. So you have used the Bernoulli equation to find the true speed. But generally people do the mistake, they put the density at sea level. So that will be the equivalent speed. So your answer will be wrong. So you can see that if the density will decrease then speed will increase, means if you are going above the sea level, your true speed automatically will increase.

So this is the significance of this thing. So, yes. So these 2 ratio you know and if you calculate P17, then this question you have to find the true speed at 15 km. So in last question we have solved the density at 15 km, so you can find the density at 15 km also and pressure at 15 km also using the standard formula. So to find the density at 15 km, again you have to use the density at 11 km which you have got in earlier question.

So using this, you can find the value of density at 15 km which will comes 0.212091 kg/m cube and P15 will be 0.120322*10 power 5 N/m square. So in this equation you can put, see total pressure is already given 0.137672-0.120322*10 power 5/0.212091. So this will be sum of kind of get the speed 128, someone get 129, I am getting 128 m/s. So this depends upon the at what digit you are taking the density and the pressure.

So this speed will be the 128 m/s. So the purpose is to know, this formula you know but what is the significance of this where you can do the mistake and also you have to just remember these things, you have to calculate the pressure at 15 km, density at 15 km, then only you can find the true speed at 15 km. So this is the, and if you decrease the density, then the true speed will increase and this equation is valid for the (()) (20:20) less than 3 in compressible flow.

So these things you have to keep in mind. And the sixth question was the temperature ratio at 17 km and 13 km. So you know that from 11 km to approximately 20 km, this is the isothermal layer. So in that the temperature remains constant. So if you take the ratio like 13 km to 17 or 17 to 15 or 15 to 16, if the temperature is same, then the ratio will be 1. So without solving this you can mark this, you can solve this problem.

So the answer will be 1. And similarly the density ratio at 20 km and 11 km. So first you have to find the density at 11 km using the standard formula because you know that from 0 to 11 km, the formula will be different to calculate the pressure ratio, temperature ratio and density ratio. And from 11 km to 20 km again the formula will be different. So you have to keep in mind that how this, where we have to use the correct formula.

So using the standard formula, you can find the temperature, density ratio at 20 km and 11 km. This is question 7. So I am not solving this. You can solve this using, first you have to find the density at 11 km that you know already. So you can see that the data which you solved for the earlier question like same data you are using for the next question.

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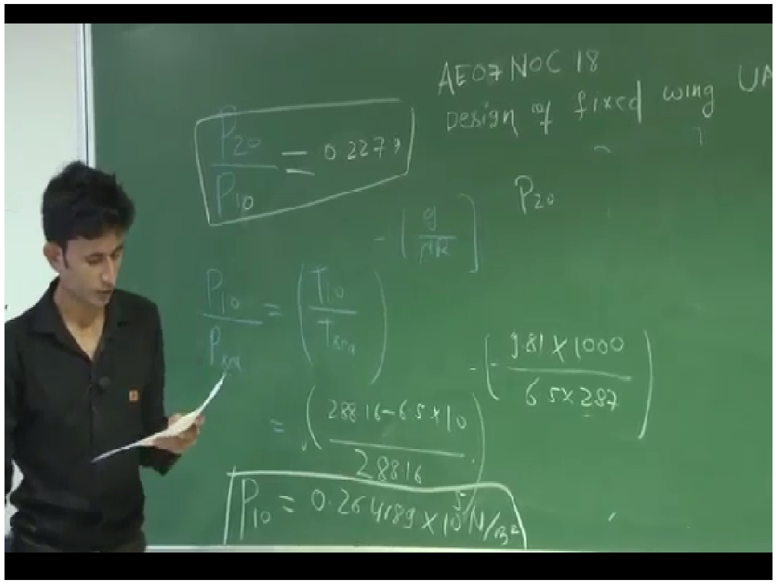

So you have to find the density ratio at 11 km to 20 km, density ratio at 20 km and 11 km. So you know the density at 11 km is 0.36382 kg/m cube. This is your one density. You can find the density at 20 km. So you have to find the pressure ratio, means P20/P11 you have to find. So you can find the, P10 sorry. So you can find the pressure at 20 km, first pressure at 10 km and then using this, you can find the pressure at 20 km and you can divide.

So first you have to find them individually P10 and P20. So first we will go the P10. So P10, for this you have to use the reference sea level, P sea level. So P10/P sea level is T10/T at sea level. This will be the -g/lambda R. So 288.16-6.5 and you have to multiply by the altitude 10/288.16 and this index will be the 9.81. This is K/km, then it will go to the, 1000 will go to the meter if you convert into the, from km to m, 6.5 with - sign, will come and 287.

And at sea level you know 1.01325*10 power 5 N/m square. Then put this value and you can calculate the P10. So your P10 will be the 0.264189*10 power 5 N/m square. So this will be your P at 10 km. So you got P at 10 km. So our next step is to calculate the pressure at 20 km. Then finally we will divide this too, we will get, we will reach our answer. So here you can use some tricks or you can find the individual at P at 11 km. So you have to find the value of P at 20 km.

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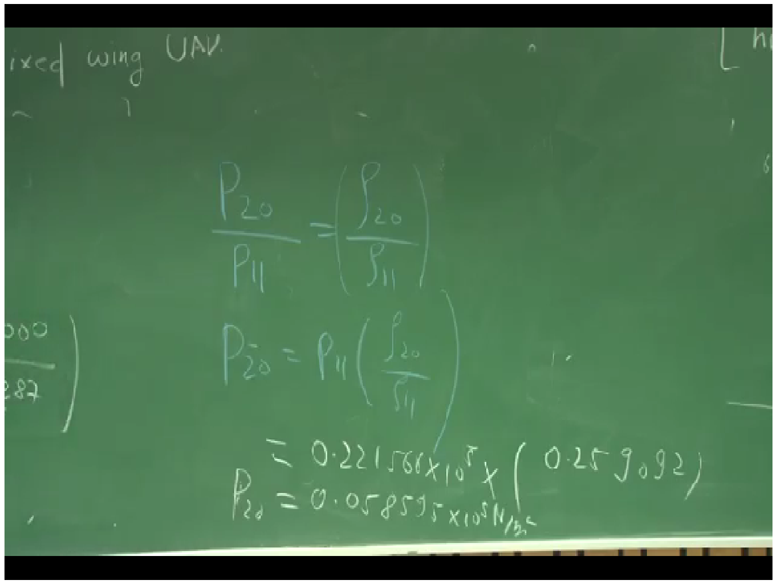

So P at 20 km you can find using this. P20/P11=rho 20/rho 11. I am not using P10. Because this 2 ratio will be the same. So you can directly use this data from the previous equation. So what P20 will be. P11/rho 20/rho 11. So you no need to calculate again, no need to calculate the P20 individual like using the formula P20/P11, P10=-T2 the power -gRT, h2-h1 or h20-h10. So using this trick you can find the value of P20 easily using the previous data because you know ratio will be the same.

So this value you know that rho 20/rho 11 you got from question number 7 and from question number 5, you got the P11, 0.221566*10 power 5 and your density ratio will be 0.259092. So your P20 will be the 0.058595*10 power 5 N/m square. So you have 2 impression P20 and P10 here. So you can divide these 2, you will get the answer. And your answer will be P20/P11, here you can write, P20/P10 is 0.2279.

So you can save some of your time using this trick. And the tenth question, the ninth question was. At high altitude long endurance unmanned air vehicle is flying at an altitude of 15 km of geopotential altitude. The pressure at that altitude will be nearest to. So you have to find the pressure at 15 km. So you can find the pressure at 11 km, then from 11 km you can go to the 15 km. So at the 15 km, the pressure will be, I am not solving this problem because in many of questions we have used the standard this formula. So you have to put just the value and you get the answer.

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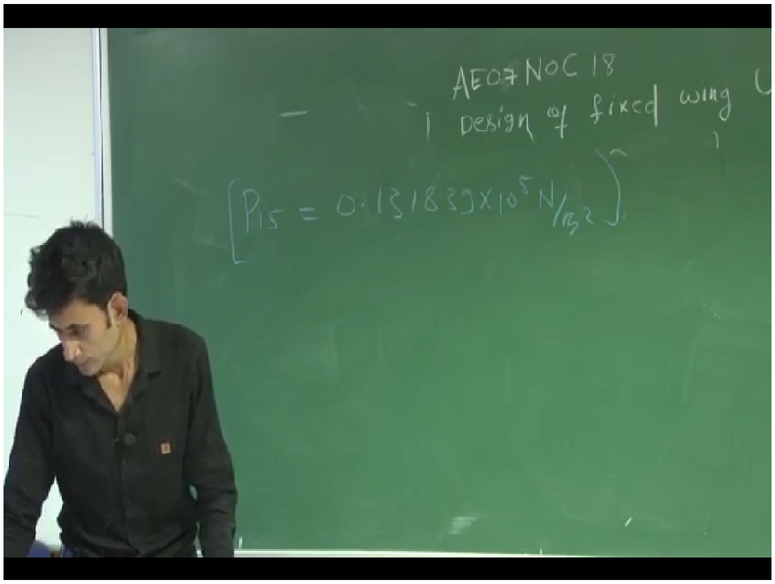

So if you find the pressure at 15 km, it will come out 0.131839*10 power 5 N/m square. So you can see the solution of 5, you will get the idea of how to get directly the pressure at 15 km. So no need to solve again and the last question was the value of geometric altitude in km at which the difference between the geometric altitude and geopotential altitude is equal to the 1%. So basically in this question, the question was just to give the idea of the difference between the geometric altitude and geopotential altitude.

Okay if you are going above the sea level, then how much it will affect, means is there same for every altitude, means geopotential and geometric or it will be different. So the tenth question was the value of geometric altitude in km at which the different between the geometric altitude and geopotential altitude is equal to the 1% of geopotential altitude.

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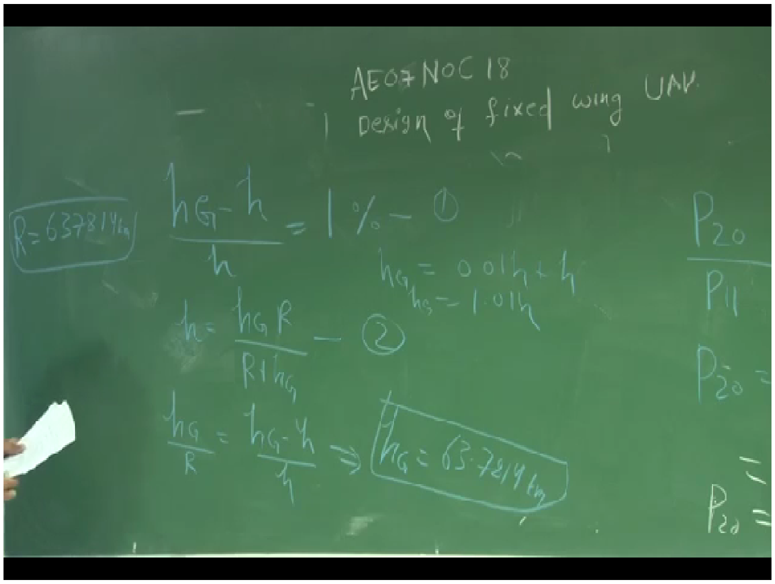

So in this, you know from the statement you can write that, you have said that h difference or geometric altitude and geometric altitude is 1% of geopotential altitude. So this is equation 1. And you know that the relation between geometric altitude and geopotential altitude is h=hGR/R+hG. R is the radius of earth 6400 km. So if you rearrange this things, then you will get, if you manipulate this equation, equation number 2, you will get hG/R=hG-h/h because you have to bring this equation in this form so that you can put this value into directly this and you calculate this.

So if you put this value 0.01 and R is 6400, we have taken the correct value of R which is 6378.14 km. So your geopotential altitude will be hG=6378 and 14. I have already explained about these things, about the significance and physical meaning of this geopotential altitude and geopotential altitude. So this is the one numerical where you can see that the geopotential altitude is coming 63.78.

So you can say that okay which altitude will be the greater, yes. For each and every altitude, hG will be greater or h will be the greater. So you can see that hG is, if you put this thing you can say 0.01h+h. 1.01h, hG. So if you put hG value here 63.4, the h value will be less than that of hG value.

So hG will be always greater than the h. At the sea level, it will be the same but when you go above the sea level, the difference will come. So using this you can say that. So this was about the assignment 1. So we will go to the assignment 2 also. So if you have any doubt regarding this, any steps or things you can post directly into the forum. We will give the answer.

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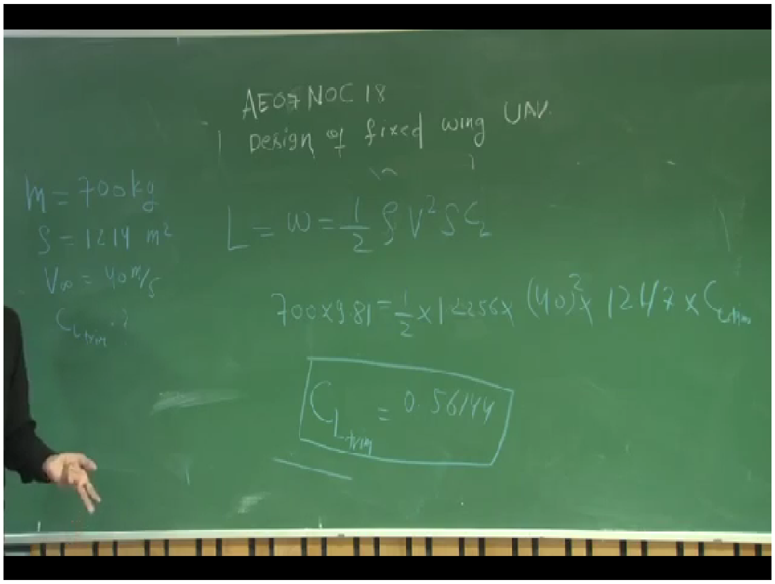

So in assignment 2, first question, an unmanned air vehicle with mass of 700 kg and wing area of 12.47 is flying in cruise at speed of 40 m/s. The air vehicle trim lift coefficient at mean sea level. So you have to find the CL value. CL trim is nothing but when the forces and movements are balanced, then the aircraft is said to be the trim condition. Means there is no rotation about the cg of the aircraft.

So you can call it trim. So basically the given information is mass of the aircraft which is 700 kg and wing area is 12.47 m square and freestream velocity is 40 m/s and you have to find the CL trim. This is at the mean sea level. So you know that okay the aircraft is cruising, then the lift is, the weight of the aircraft is balanced by exactly lift. So using lift=weight which is 1/2rho V square S*CL.

So 700, this is the weight, so you have to convert into Newton. So 700*9.81=1/2*1.2256*, speed is given 40 m/s, 40 square*area is 12.47 and CL trim. So if you solve this you will get, CL trim=0.56144. In this case basically the aircraft is flying without any rotation about cg at cruise condition. So just you have to find the value of CL trim. So based on data, you can see that which equation we have used.

See in cruise condition, if suppose that aircraft is cruising, thrust is balanced by drag and lift is balanced by weight, so which equation you will use. So based on data you can use that okay we have to use this equation. Because weight is given, freestream velocity is given, area is given. So all the things are coming in this equation. So only 1 parameter you have to find, then you can use this relation and you can find this.

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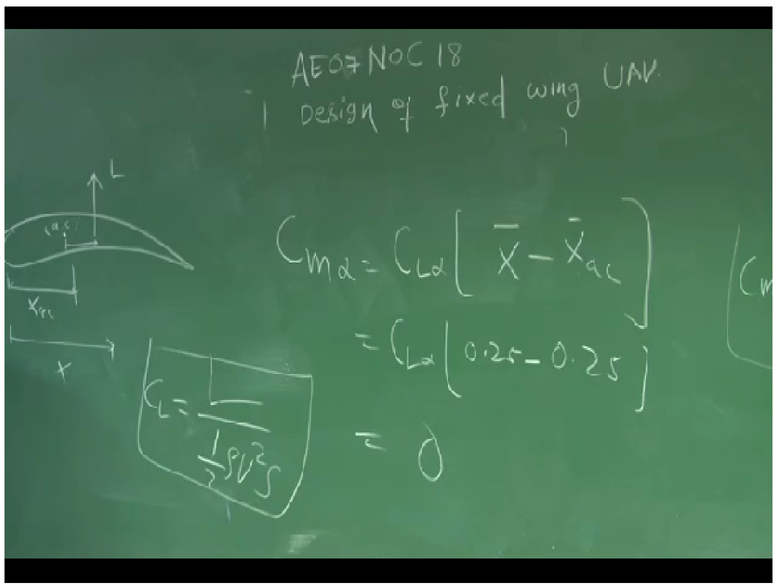

And second portion is, so in second question, the question was, you have to find the value of Cm alpha in per radian at quarter chord point of thin airfoil. And note also you have given that Cm is the first derivative of the pitching movement coefficient with respect to the angle of attack. So this is just the introductory part of this like you can say stability and sir already told about that I think in last lecture that what is Cm alpha, what is Cm0, what is aircraft stability and how the Cm alpha and Cm0 plays the important role in longitudinal aircraft stability.

So basically the question which is given to solve this just to introduce what is Cm alpha. In this question, you have to just balance the movement and you will find the answer. So this is little bit tricky. So you have to find the value of at quarter-chord point. So you know that you have given the airfoil and also in this question, it is mentioned that the airfoil is thin. Then you can consider as a, you cannot think about this camber airfoil or symmetrical airfoil because in this question, you not need to go about that deeply well.

You just draw the airfoil and you just try to balance the movement and at what point you have to balance the movement. Then you will find the answer. So in this question you have to find the, suppose that this is the airfoil and 1 lift force is acting in this direction and you have to balance the weight, sorry you have to balance the movement. So movement you can write Cm, force*distance and if you take the first derivative of that, then Cm alpha will become CL alpha X bar-Xac.

So this is basically, this CL, first this CL is coming from the lift. If you divide lift/1/2rho V square S, this will be your CL. So first you take that this lift and what will be the distance between these 2 points, X bar-X, Xac, these 2 points which I have written here. Divide by this coefficient you will get CL. Take the derivative, you will get CL alpha and here Cm is what?

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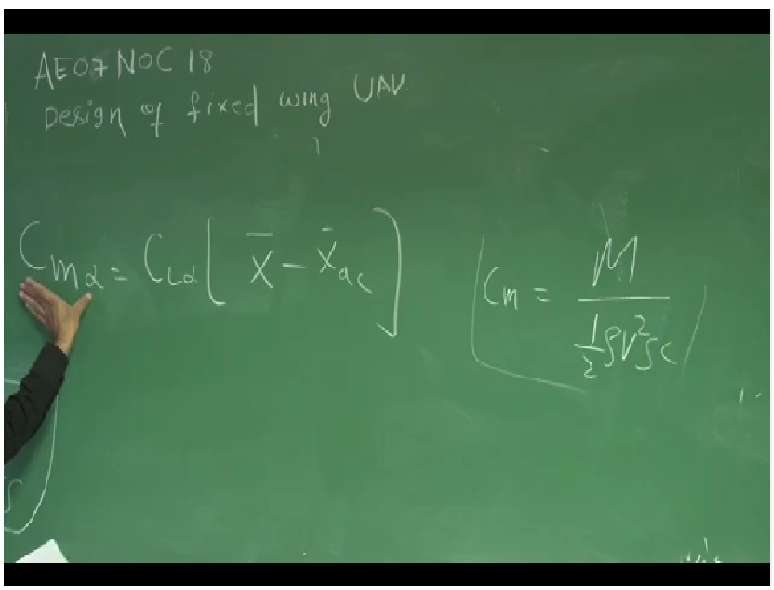

M/1/2rho V square S*C. And C bar you can take here, this is X bar, means X/C and X bar ac means X ac/C. So C will come here and if you take the derivative, this will become Cm alpha. So basically this is the movement and movement=force*distance and you have asked to find the value of Cm alpha at quarter chord. Means 1/4th. 1/4th of this chord length. So you know that this is also given quarter, means 0.25 of chord.

X bar is already given, X/, X bar is X/ac, which is given quarter chord point. So 0.25 and you know that for a location of X ac is approximately at quarter chord point of the airfoil, means if you have the chord length, if you divide it by 4, that will give the location of the aerodynamic center. So this will be also 2.5. So this is 2.5, sorry this is 0.25 and this is also 0.25. So this will be 0.

So at the quarter chord point, the Cm alpha will be 0. So later sir will tell you about this where the Cm alpha will be 0. At the neutral point, the Cm alpha will be 0 and neutral point is the aerodynamic center of the whole aircraft. Sir will explain you about this Cm alpha and neutral point and with that derivation also, then you will get the feed for that. Okay, so basically here you are doing, you are trying to find the Cm alpha at aerodynamic center.

In the whole aircraft, you will find the neutral point. So neutral point will be the aerodynamic center of the whole craft. So at neutral point, the Cm alpha will be 0. So you will get, you can relate this when sir will explain about you the neutral point, then you can relate this to this, then you will get the feel of okay the aerodynamic center of the wing is equivalent to the neutral point of the whole aircraft.

So in upcoming lecture, sir will explain you very clearly this is... And in third question, the maximum distance in percentage of maximum thickness of the airfoil between the mean camber line and chord line. So in a symmetrical airfoil, the mean camber line and chord line will coincide.

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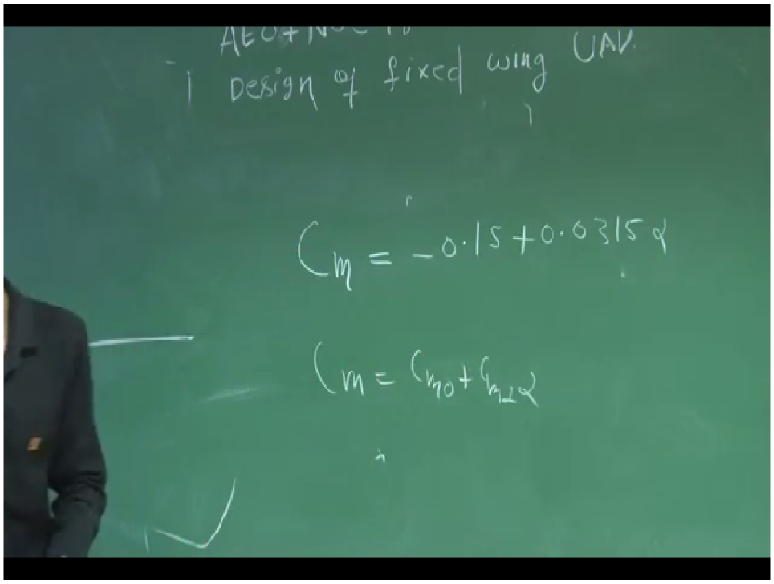

So if you are taking this chord, so mean camber line will be the same as the chord line. So by seeing this question itself, you can say that the maximum distance or distance will be 0 or in whatever you can with respect to any percentage or anything you can take. So the straightforward answer will be 0 because mean camber line and chord line will coincide each other.

And fourth question is which one correct about the aerodynamic center of the airfoil? So basically at the, you can see that at aerodynamic center, the Cm alpha is 0. Means it is independent of angle of attack and to get the Cm alpha, you have to feed the velocity. Means okay you are getting this Cm at this particular velocity or you are getting the movement at particular, with fixed velocity.

If you change the velocity, the movement will change. So basically in this question, the first option was the aerodynamic center and pitching movement coefficient is independent off angle of attack. So we have seen that the Cm alpha=0 at aerodynamic center. So by seeing this, you can say that okay it is independent of angle of attack. And second and fourth option you can eliminate because third option was aerodynamic center varies with variation of angle of attack and the location of aerodynamic center depends upon the camber of the airfoil.

So by solving this, we did not consider anything. We did not think about that the camber and symmetrical things, so this option also you can eliminate and the location of the center depends upon the thickness of the airfoil. Thickness we did not also consider that, okay no need to worry about the thickness and all. So the answer was the aerodynamic center, at aerodynamic center the pitching movement coefficient is independent of the angle of attack because Cm alpha at aerodynamic center will be 0.

So in the fifth question, the pitching movement coefficient about the leading edge of the airfoil is given by this expression. So Cm is given -0.15+0.0315 alpha, where alpha is the angle of attack of airfoil in degrees. The trim of angle of attack of the airfoil in degrees will be. So basically for the trim, the word trim means that there is no rotation about the cg of the aircraft. Means forces and movement are balanced. So movement are balanced means you can put Cm=0.

If Cm=0, means there is not rotation about the cg, means it is called trim condition. So if you put Cm=0, you will get the alpha. So basically in this equation, it is given, sir will explain about this significance of this equation that you can write Cm=Cm0+Cm alpha*alpha. So this is basically Cm alpha and this is basically Cm0. So for Cm0 is negative, unstable. Cm alpha is positive unstable. Sir will explain these things. Sir already explained these things, the importance of Cm0 and Cm alpha in aircraft stability, the aircraft longitudinal stability.

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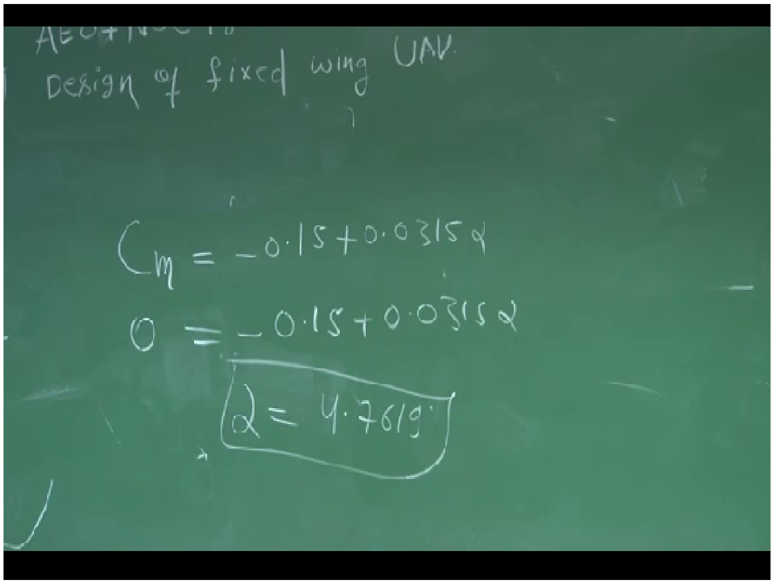

So here you can put Cm=0 because it is trim condition. So you will get alpha value, will be

4.7619. So in this question, just understand what is trim. If you know what is trim, then you can say that okay movement is balanced, we have to put Cm=0, you will get alpha. And sixth question you can easily solve by using the, because we have given the (()) (49:43). So in this question by increasing or decreasing the freestream velocity, the aerodynamic center will not change.

In this case, the difference between the location of aerodynamic center in both cases will be 0. So whatever you increase the speed, suppose that aircraft is flying 20 m/s, if you increase 40 m/s, the aerodynamic center is a fixed point. It is not dependent upon the speed and all. So in both the cases, the aerodynamic center will be the same. So if you take the difference between both the cases, then obviously it will be 0. Like cg will vary when you change the weight.

Cg will not vary when you increase or decrease the speed. Like aerodynamic center also, aerodynamic parameter which does not depend upon the speed and all.

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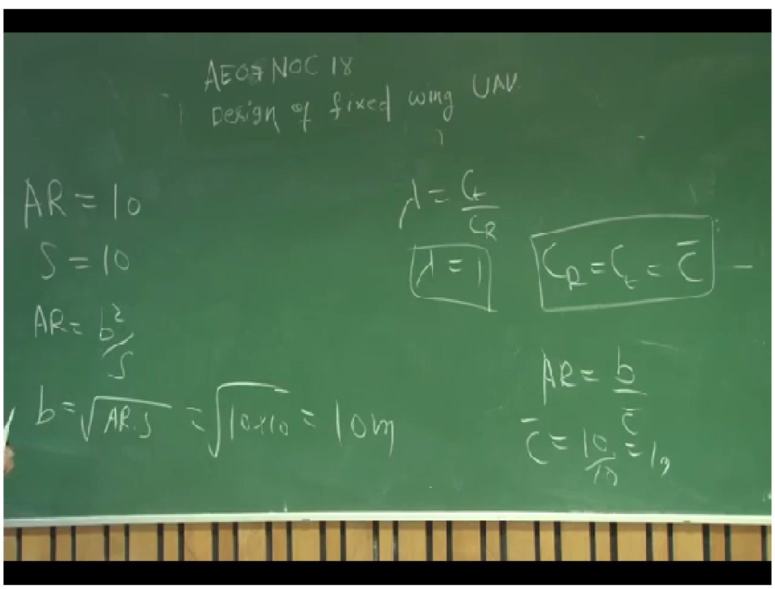

So in seventh question, the aspect ratio is given 10 and wing area is given also 10. So using this aspect ratio=b square/S. So b will be under root aspect ratio*S. So aspect ratio is 10, area is 10. This will be the 10 m. So you have to find the mean aerodynamic chord for the rectangular wing. So actually in rectangular wing, what will be the taper ratio. Ct/CR. So Ct and CR will be the same for rectangular, so lambda will be 1.

So in this case, thus root chord, tip chord and mean aerodynamic chord will be the same. So basically you can use this formula aspect ratio which is equal to b/C bar. This is called rectangular wing only and this is for the generalized formula. So people can do the mistake. They can use this formula for all the cases but this is not true. This is valid for the rectangular wing only. If you put S=b*C bar for rectangular wing, this will come out to the b/C bar.

So aspect ratio is 10, b is also 10. So from this, you can see that 10/10 is 1 m. That only you have to find these things.

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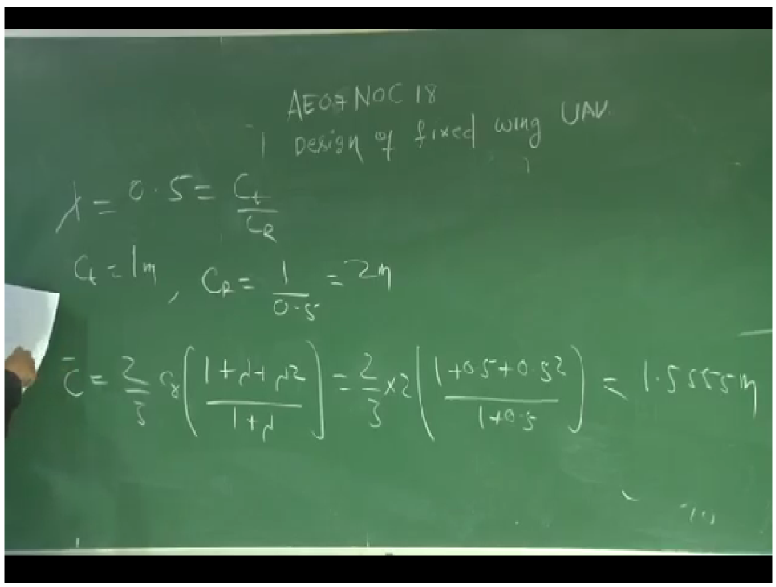

And for the eighth question, again the lambda is given which is 0.5 which is Ct/CR, tip chord/root chord and Ct is given 1 m. So CR you can find using this. CR will be the 1/, this 0.5 which it will be 2 m. So first you have to find the, after that you just find the value of mean aerodynamic chord which will be 2/3CR, root chord, 1+lambda+lambda square/1+lambda. So this will be the 2/3*2, 1+0.5+0.5 square, 1+0.5. This will be the, C bar will be 1.5555 m.

So in this question you have to find the distance between aerodynamic center of this wing with leading edge, or nose. So to get the answer of this question, you have to draw this figure so that you can get, feel this.

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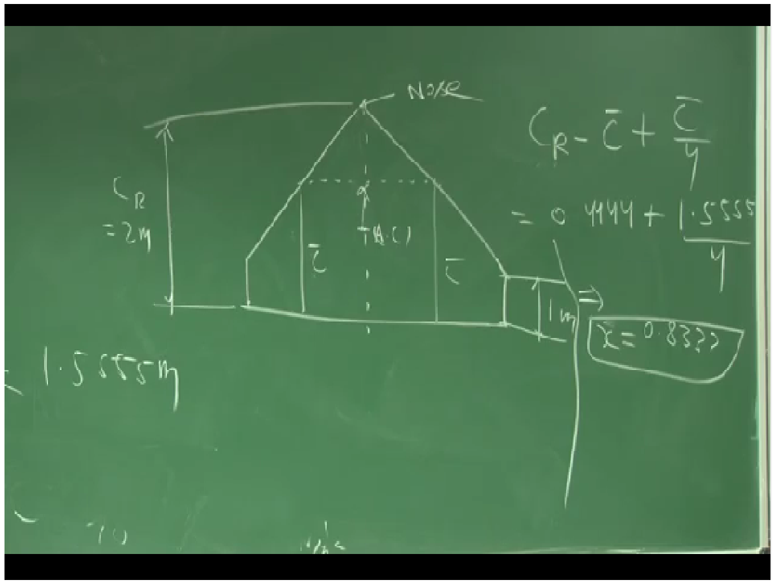

So what is given? This was given 1 m and using the data, you found that the root chord will be, this chord, root chord, CR is 2 m. So this thing you know. You draw this and you also find the mean aerodynamic chord somewhere located here, which is 1.55, this is C bar. So basically you have to find the aerodynamic center of this wing from the nose. You can say this is nose or leading edge, this nose.

So you know that the aerodynamic center for the subsonic wing, if your flight is subsonic, then you can approximate, the aerodynamic location will be 0.25 of the mean aerodynamic chord. But this distance will be from here. Not from here and it is asking that you have to find the distance between nose and aerodynamic center. So you have to add the distance also. So to get this distance, what you will do?

You can subtract CR-C bar. So CR-C bar if you subtract, CR=C bar and then I use capital or small r, it will be the same. So CR-C bar will be this distance. This distance you have to add this much distance because this is the location of aerodynamic center. You can get by dividing C bar/4. So this distance will be the C/4. So this distance+this distance will be your answer. So CR-C will be this distance, C/4 will be this distance.

So CR-C will come out 0.4444+, you know mean aerodynamic chord, 1.555/4. So the distance between the nose and aerodynamic center, this is your aerodynamic center, will be, you can consider as X=0.8333. This will be your answer. This is all about assignment 2. So we will go for the assignment 3 also. In assignment 3, the question was an UAV has a weight 7500 N and lift to drag ratio is 13, wing span is 15 m and mean aerodynamic chord of 1 m. It is flying at speed of 35 m/s at mean sea level. The lift coefficient at cruise will be. So this is very straightforward answer.

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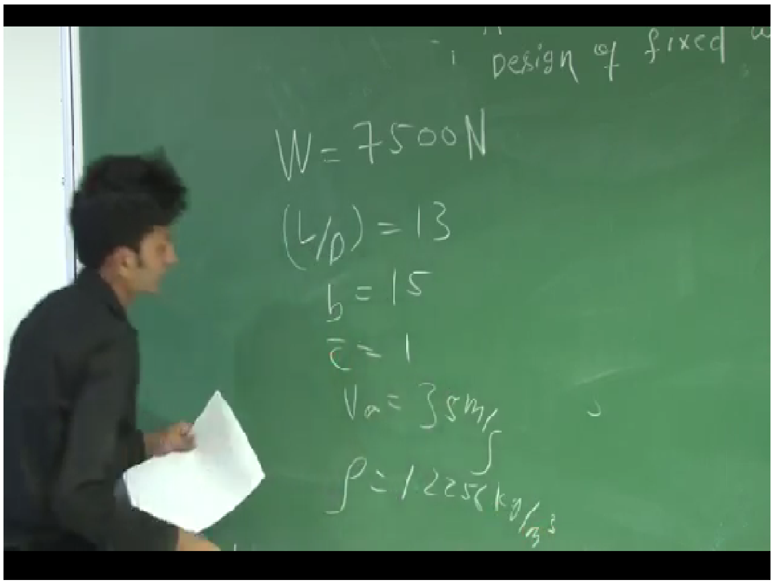

The weight is given already in Newton, so no need to calculate, no need to convert. L/D is given 13. b is given 15. C bar is given 1 and V infinity is given 35 m/s. So mean sea level, rho will be the 1.2256 and wing you can take rectangular.

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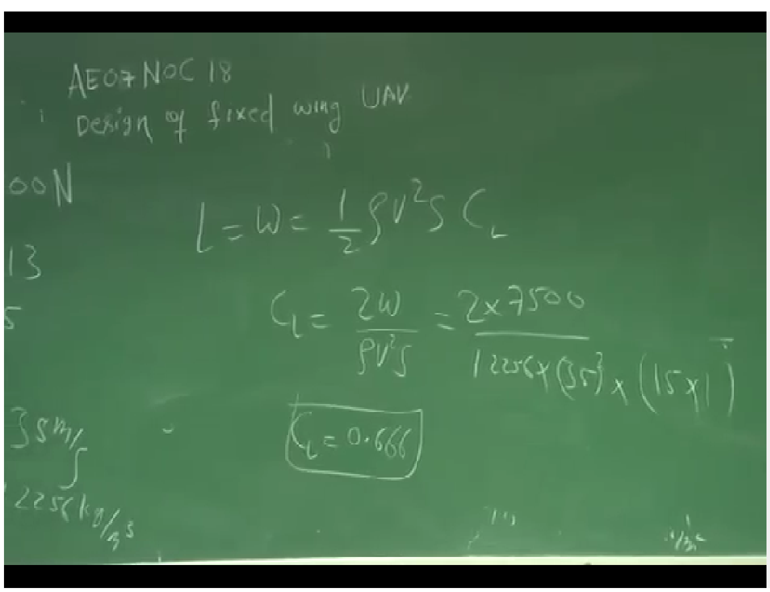

So using this lift=weight, which is 1/2rho V square S*CL. So CL will be 2W/rho V square S. So you can put this value, 7500/1.2256*, V is 35 whole square. What will be the area, b*C bar. B*C bar is 15*1. So your CL will be 0.666. This will be the CL. And in second question, you have to find the value of CD.

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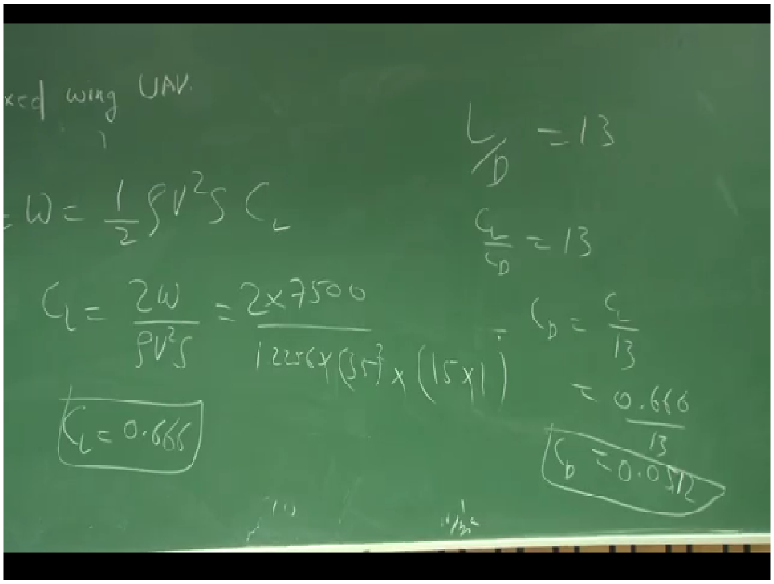

So you found out that CL is 0.666 and given the lift to drag ratio, so lift to drag ratio is given that 13. This is nothing but CL/CD which is 13. So CD will be CL/13. So what is CL? 0.66/13. So CD will come out 0.512. This will be the CD value, 0.0512. This will be the CD value. And the third question, you have to find the value of thrust required. So you know that the aircraft is cruising.

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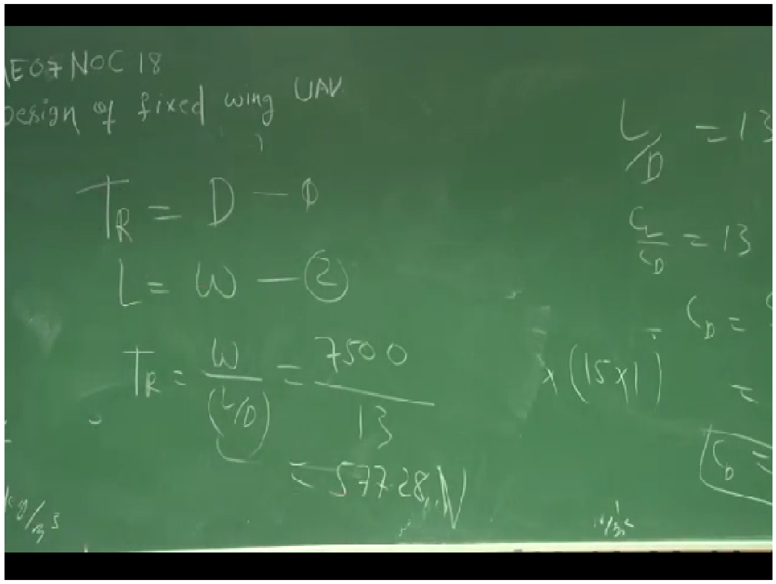

So if the aircraft is cruising, so the thrust is balanced by the drag. At that time, what will be the thrust required and lift=weight. This is the question 1 and question 2. If you manipulate question 1 and 2, you will get, thrust required will be the W/L/D. So W is given, 7500. L/D is 13. So your answer will be 577.28 which is in Newton. So this will be the your thrust required. Again you have to, in fourth question, you have to find the power required. So power required is nothing but thrust required*velocity. So thrust required already you know.

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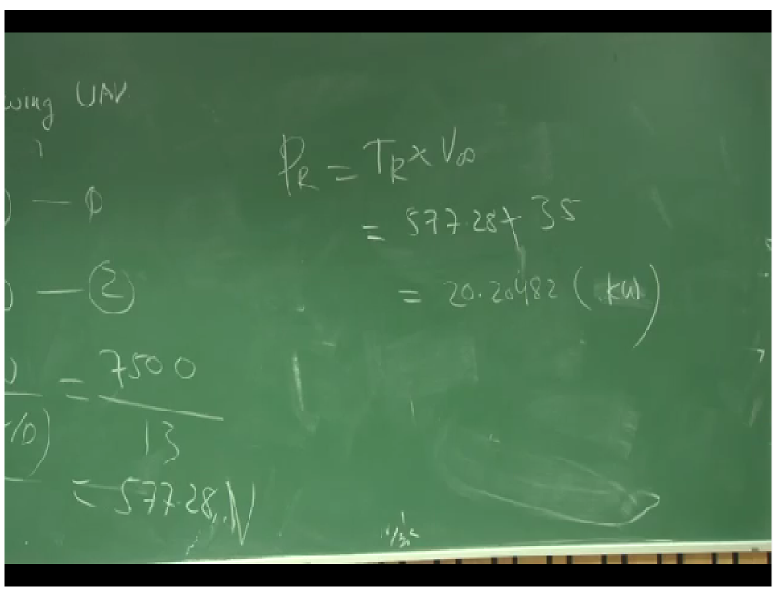

Power required you can write. Thrust required*V infinity. So thrust required you got 577.28, velocity, we already gave you 35 m/s which will be the 20.20482, which will be in kilowatt. So this is your answer of the fourth question. And in fifth question, you have to use some trick. If you clear the meaning, if you know the actual meaning of aspect ratio and definition of aspect ratio, then you will get the correct answer of this question.

So fifth question was keeping all parameter constant, how the wing span b is related due to the induced drag. So suppose that if you fix all the parameter, speed is fixed, weight is fixed. You are flying at a constant altitude, means rho is fixed. Your mean aerodynamic chord is not changing at all. Then what will be the effect of the induced drag when you increase the span. Like glider has a larger span compared to this trainer type of aircraft like CANSA, Cessna. So this will give the significance of that, this question.

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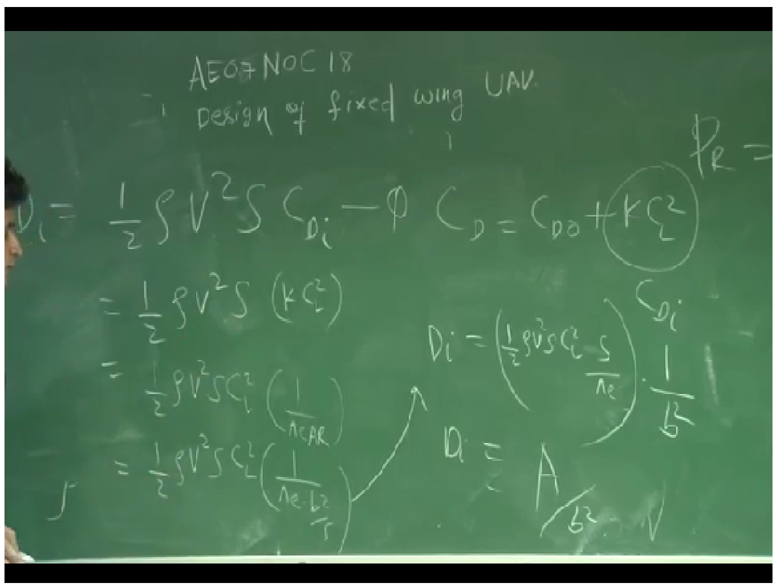

So basically you can write drag is what? 1/2rho V square S*CD. If it is induced, you put I, this is induced drag. And from the drag polar, you know that CD=CD0+KCL square. So this is parasite drag coefficient and this is the induced drag coefficient. So induced drag coefficient is the lift dependent drag and this is parasite drag coefficient, means it will not depend upon the lift. So this will be the induced drag coefficient.

So basically this is CDi. So you can put this CDi in a question 1, you will get 1/2rho V square S and CDi will be KCL square. This will be the things. So 1/2rho V square SCL square and K is what? 1/pi*e*aspect ratio. 1/2rho V square SCL square 1/pi e, aspect ratio is b square/S. This equation you can write, this equation. Di=1/2rho V square SCL square, S will go up, S/pi e, so this again we mentioned that all parameters are constant.

Just to bring this equation in span, how the induced drag is related to the span. So this parameter is constant. You can consider this as A only, A/b square you can consider, A/b square. Here you can do one mistake that you can aspect ratio, if you put b/C bar, then your answer will be different. So this aspect ratio b/C bar is valid for the rectangular wing, not the all case. So Di is 1/b square. So you can see that if you increase 3 times, means if your weight is increasing 3 times, so basically if you increase the weight of that 3 times, then what will be the CL?

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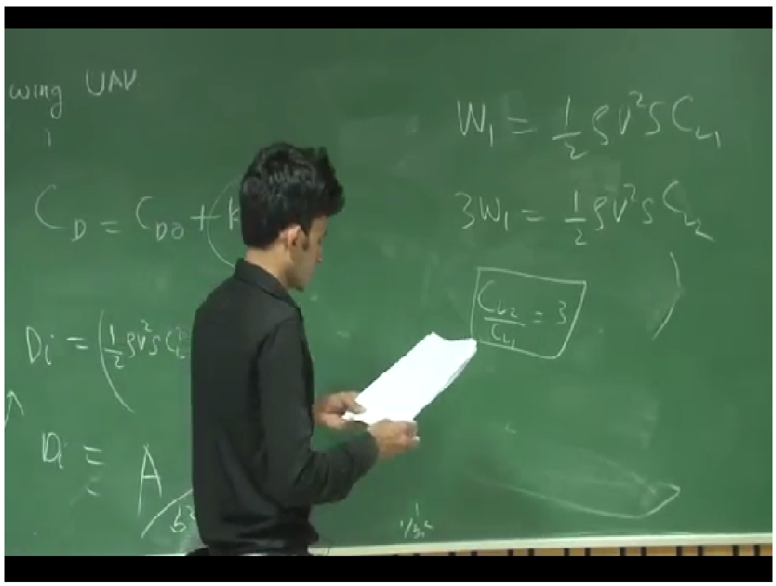

So first one, the weight is W, you can say 1/2rho V square S*CL1. So this is lift coefficient when the weight was W. You increase 3 times. 3 times means, 3W1, 1/2rho V square S*CL2. So if you divide these 2 equations, you will get CL2/CL1=3, "theek hai." This is the answer for the sixth question. Fifth question already I have given the answer. Induced drag is inversely proportional to the b square, span of the things, okay.

This is the answer for the sixth question. So in the sixth question, if suppose that your weight was W1 and you increase the weight 3 times, so 3W1 will be the 1/2rho V square S*CL2. So if you divide this equation, then CL2/CL1 will be 3, okay. So you know that drag formula is here. Induced drag is 1/2rho V square S*CDi. CDi you can put KCL square. If you divide these 2 equations in terms of CDi1 when you put 1, you have to put 1 here. If you Ci2, then CL2.

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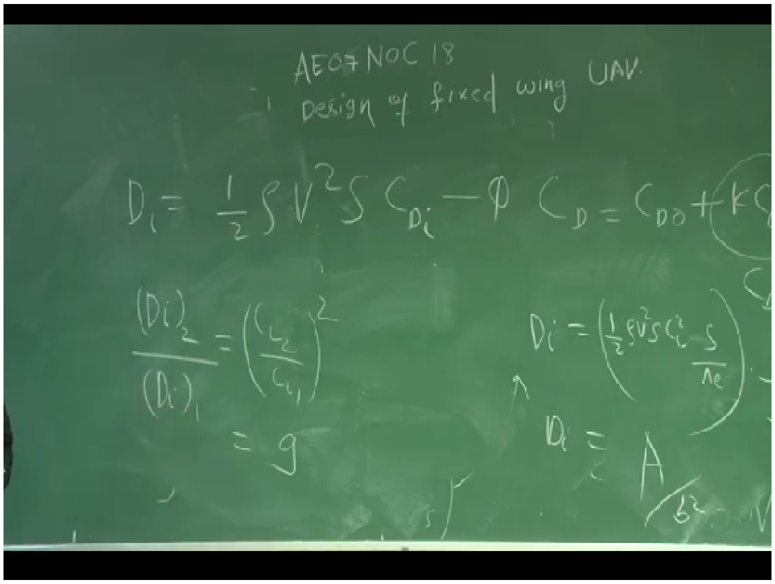

So basically if you divide this equation, you will get Di2/Di1=CL2/CL1 square. Because CDi KCL square and all the parameters are constant. So if you divide them, you will get CL2/CL1 whole square. And we have given that weight is increased by 3 times, then you can get the CL2/CL1 using this. You can put these things into this, you will get 3 square will be 9. So your induced drag will be 9 times of the previous one if you increase the weight 3 times.

So basically this is the effect of the weight when you are not changing any other parameter. So this is the answer of sixth question. The seventh question was at the steady and straight level flight, what will be the flight path angle. You have 3 things here. You have chord line, you have velocity vector and you have horizon.

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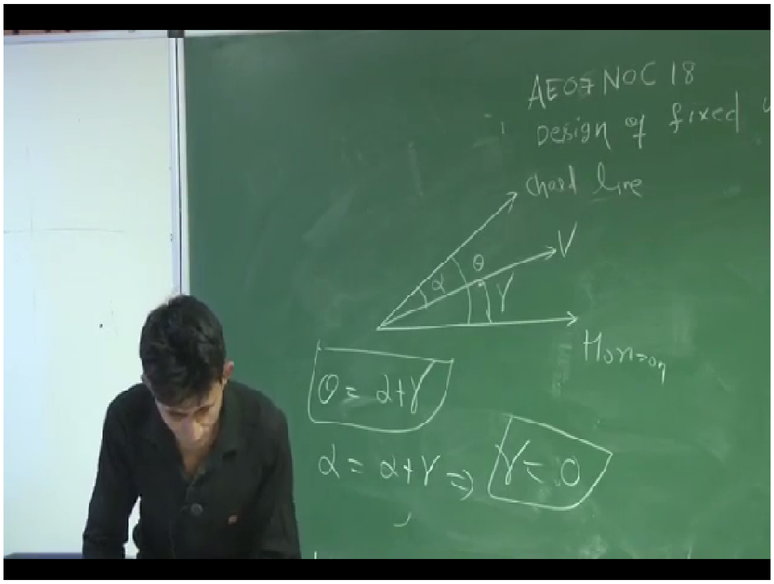

Suppose that this is horizon and this is your velocity vector. This is your V vector, this is surface which is parallel to the earth and this is chord line. So already I told you the angle of attack between chord line and freestream velocity is the angle of attack, alpha and angle between chord line and horizon will be the pitch angle and angle of attack between the velocity vector and horizon will be the flight path angle.

So using this you can say that okay if you have alpha+theta, you will get theta, alpha+gamma, you will get theta. So theta will be the alpha+gamma. So if you are cruising at a steady and straight level flight, this means these 2 lines, velocity vector line and horizon vector line will coincide. This means that your theta will be the angle of attack. So alpha=alpha+gamma, then gamma=0.

And for eight question, the question was a rectangular wing with elliptical lift distribution span of 10 m and chord length of 1 m is constructed with the airfoil having the lift curve slope of 0.1 per degree. What will be the total lift curve slope of the wing? So basically you have to convert this thing, from 2D to 3D.

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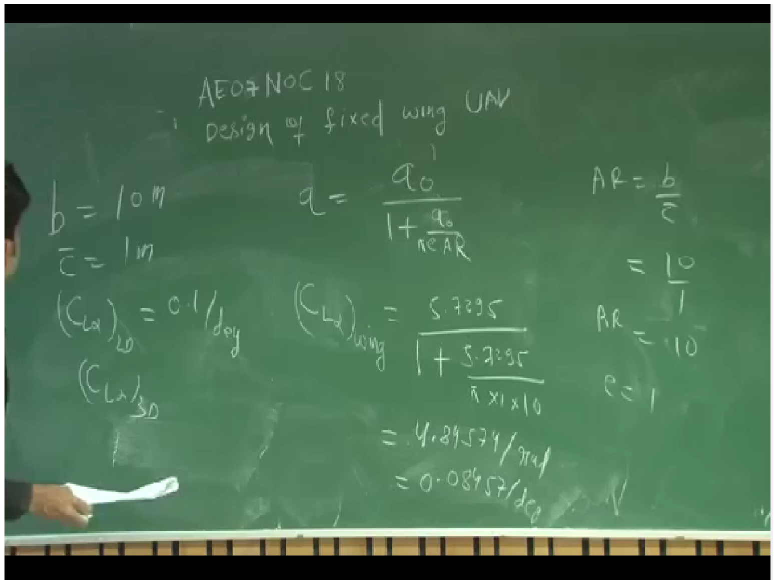

So we have the information span is 10 m, chord length is 1 m, CL alpha lift curve slope of 2D is 0.1/deg. You have to find the CL alpha of 3D? So you can use this formula a=a0 1+pi*e*aspect ratio. This means a0, so pi will be here and a0 will be here. So a=a0 1+a0/pi*e*aspect ratio. a0 is the lift curve slope of the airfoil, a is the lift curve slope of the wing. So you can write CL alpha wing, so with the given information, the wing is given rectangular, so you can find the aspect ratio b/C, aspect ratio is b/C bar.

So 10/1 will be 10. Given that electrical distribution, e will be 1 and it is given that the lift curve slope of the airfoil is 0.1/deg. So when you are calculating each and everything, you have to first convert into the radian. Because this pi is given, pi will be in the radian, so in each and every I would like you to put the value means you have to convert into the radian. So CL alpha is airfoil is given 0.11/deg.

Then if you convert into radian, if you multiply by 180/pi, you will get in per radian. So this will come out 5.7295, 1+5.7295/pi, because here you will put the value of 3.144 something as radian basically. 1*10, aspect ratio is 10. So you will get CL alpha of the wing is 0.08457/radian. First we can write directly into per radian, then in next step we will convert per degree. So 4.84574/radian.

Multiply by pi/180, you will get 0.08457/deg. So you can see that earlier lift curve slope was 0.1

and now it has decreased. So when we increase the affect, when you increase the aspect ratio or if you decrease the aspect ratio, then what will be the affect of the lift curve slope you can see from this answer. That is why sir asked me to design this type of question. So if the aspect ratio is, if you put infinity, then this will become 0 and your lift curve slope of the wing and lift curve slope of the airfoil will be the same.

So when the aspect ratio is infinite, means your wing lift curve slope and wing airfoil lift curve slope will be the same. So for an infinite aspect ratio, it is called airfoil and when you have some finite value, then it will be less than that airfoil. You can see this. So in the ninth question, the drag polar is given and you have to find the drag coefficient and maximum L/D ratio.

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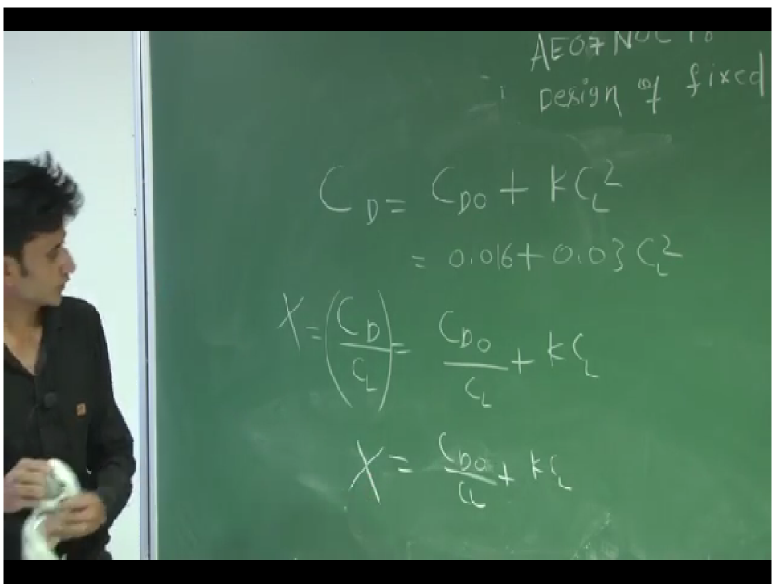

So the drag is CD0+KCL square is given. The value of CD0 is 0.016, the value of K is 0.03 CL square. So you have to find the drag coefficient at maximum L/D. So basically you are maximizing the CL. So if you have one equation, then if you want to find the maximum and minimum value, you can easily find. You have to find the CL/CD max, this means if you reverse this, CD/CL, it will become minimum.

So you can divide this equation by CL. So CD/CL=CD0/CL or +KCL. This is your one function, let us call this X. So X is your CD/CL. So you want to maximize the CL/CD. This means if you reverse this CD/CL, it will become minimum. So see you can write capital X=CD0/CL or +KCL.

So if want to get the value of maximum and minimum value, you have to take the first derivative of this and equate it to 0. With the second derivative, if it is coming negative, means function is maximum and if it is positive, second derivative is positive, then function is minimum. So based on that you can check.

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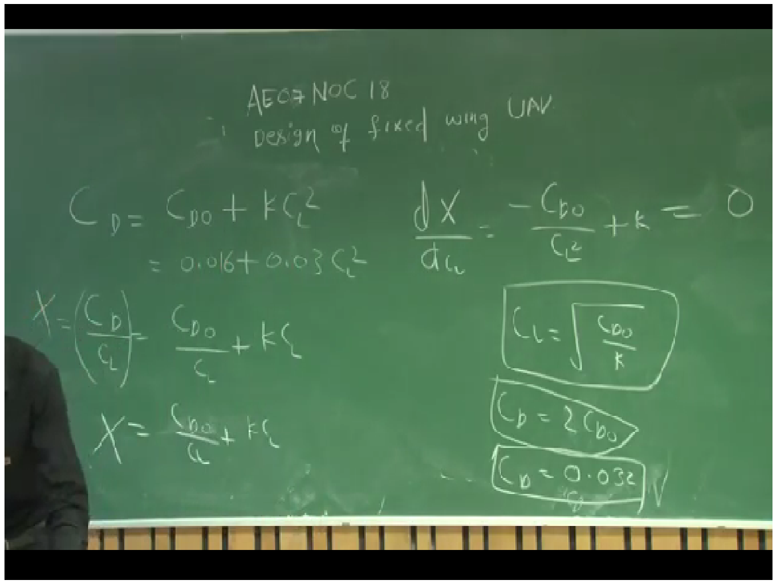

So if you differentiate this with respect to the CL, so dX/dCL=-CD0/CL square+K, 1/X differentiation is -1/X square, thus we have written. You have to equate this thing to the 0, then you will get CL=under root CD0/K. So if you take the second derivative, this will become the positive. Positive means minimum. For any value of CL and CD, it will be the minimum. So second derivative is minimum, means CD/CL is minimum.

So CD/CL is minimum means L/D is maximum. So this is CL=under rootCD0/K. If you put this CL equal to here, you will get CD. CD=2CD0. So you will get the relation CL=under root CD0/K and CD=2CD0. So the value is given CD0=0.016. If you multiply it to X, you will get CD. So CD will be 0.032. This will be your answer. This is the answer of ninth question and in tenth question you have to find the value of CL.

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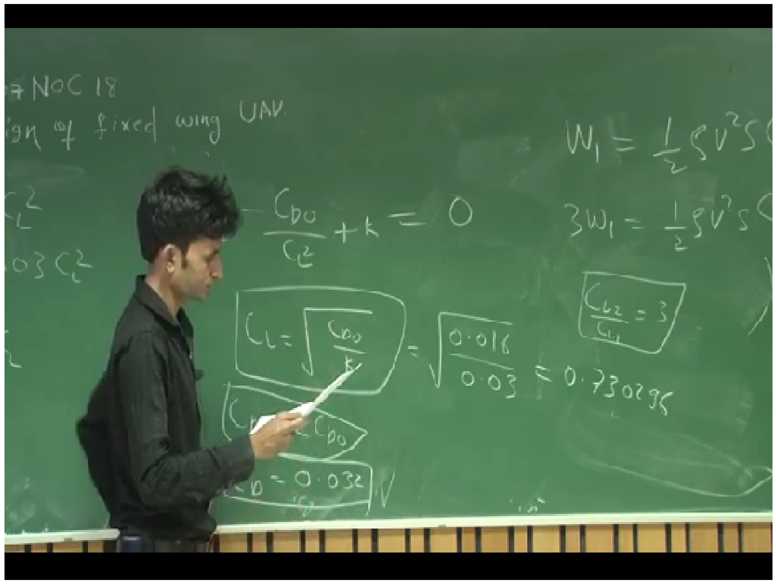

So you know CD0 under root 0.016 and K is 0.03. You will get 0.730296. So that CL will come out to be 0.730296, that will be the answer of your tenth question. And ninth question you have to find the value of CD which is 2CD0 which is 0.032, that is all about assignment third. So last we will discuss about the assignment fourth. So in assignment fourth, the data is given for the UAV and some additional information is also given.

First we have to go through the information which is given. For an UAV, the wing is rectangular shape. In addition, following data is applicable. The weight of the aircraft is 250 kg and cruise speed is 35 m/s and CL alpha is 0.1/deg. CL alpha of the UAV is 0.11/deg and alpha at lift=0 is -2 degrees. C bar is 0.8 and b is 5.5. So in first question you have to calculate the 0 lift coefficient at 0 degree trim angle of attack of UAV.

So you have to notice in this question that the 2 lift curve slope is given. One is given for the CL alpha of the wing and second one is given for the CL alpha of the UAV. So you can see that the CL alpha of the UAV is greater than CL alpha of the wing. So how this is coming sir will explain you, when you go for the tail contribution, you will be get that okay lift curve slope of the aircraft is greater than the lift curve slope of the wing alone. Sir will explain about this.

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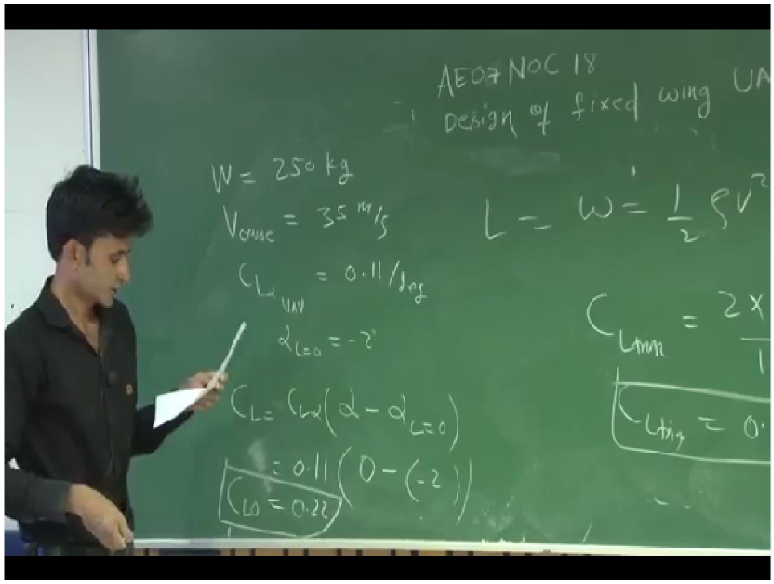

So the information is given weight 250 kg, V cruise is 35 m/s, CL alpha of UAV is given 0.11/deg and alpha when lift=0 is -2 degree. Here the alpha at lift=0 is same for alpha at wing and aircraft. So in first question, with this given information first you can find the CL trim.

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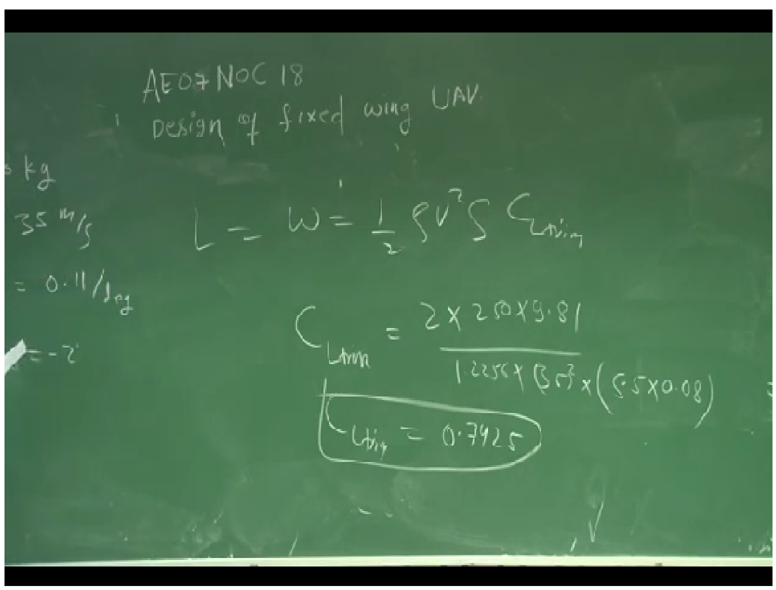

So what will be the CL trim you can just equate lift=weight=1/2rho V square S*CL trim. CL trim will be 2W, 2*250. It is in kg, you have to convert into Newton, divided by 1.225 sea level density and velocity square, 35 square and b is 5.5*C, it will give us area because the wing is given rectangular. So the CL trim will come out 0.425, that will be the CL trim. So in first question, you have to find the lift coefficient at 0 degree trim angle of attack of UAV.

So you can use these things. Use this relation CL=CL alpha alpha-alpha lift=0. So first question is about the UAV. So here you can, you have to use the lift curve slope of UAV. So 0.11, this is 0 because you have to find at 0 degree trim angle of attack. This is -2. So if you multiply this thing, you will get 0.22.

CL we have found; this will use in for other question but this is a straightforward. You can call it as CL0 because you are finding CL at 0 degree angle of attack, you may find CL0. Similarly, for second question, you have to find the CL at 0 degree trim angle of attack for wing. If it is asked for wing, then you have to use the lift curve slope of the wing.

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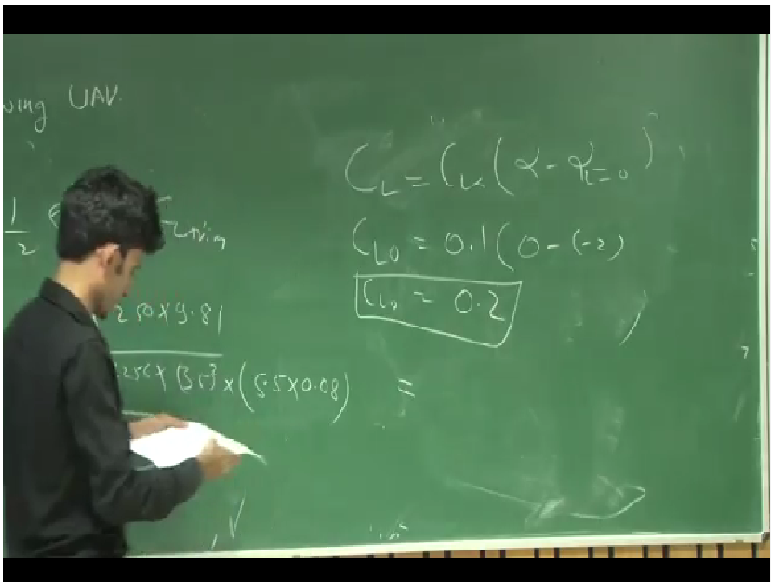

So use the same equation which we used here. CL alpha alpha-alpha lift=0, okay. So this thing you have to find, CL0. Because you have to put 0 here, that is why we are putting 0 here, - -2, 0.1, because already we told you that okay lift curve slope, alpha and lift=0 is same for UAV and wing. So this will come to 0.2. You can also say that CL0 of the UAV is greater than CL0 of the wing alone.

So if you combine this wing and this, if you talk about the whole aircraft, then this CL trim is based on the UAV data. This means CL trim is the whole aircraft. This CL trim is not for the wing. This, CL trim is for whole aircraft. So what will be the, at what trim angle of attack, UAV is flying actually. You can calculate in third question that we have asked. So in third question, again use the same relation.

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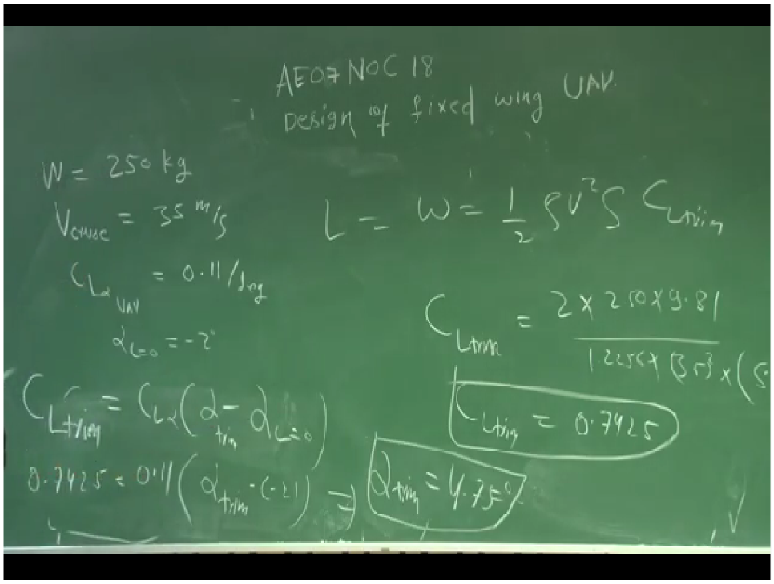

CL trim=CL alpha alpha-alpha lift=0, you can notice that in first 2 questions, we have asked the CL0 by putting alpha=0 for both the cases, for wing alone and with complete UAV. But in this question, we have asked to find the value of trim angle of attack at this condition. So actually to achieve this speed, what will be the trim angle you have to maintain. So using this data and these things, which comes out using this data, you can find the alpha trim.

So if you are putting CL trim, you have to put alpha trim here. So you can put CL trim value here. What is the value? 0.7425 and you are talking about whole UAV. So you can put the lift curve slope of the UAV, 0.11, alpha trim you have to find and you know these things will be -2. So using this, alpha trim will come out approximately 4.75 degree. And in fourth question, we have asked find the value of CL trim, so this is the answer of the fourth question.

And in fifth question, to maintain the same trim lift coefficient obtained in question 4 for UAV at 15 km geopotential altitude, what will be the new close velocity? So basically if you ask for the 5 km, right, so this means you are going above from the sea level. This means at that particular altitude, density will be less than as compared to the sea level density. So suppose that your density is decreasing, so you know that lift=weight, 1/2rho V square S*CL.

You cannot change S and this is automatically changing, where you cannot change. So somehow you have to balance the same weight. So your speed will be such that the product of V square and CL will be such that the weight will be, you can maintain the same weight by arranging, by changing this V. So V automatically will change. If density will change and you are maintaining the same weight, this means the velocity will change.

So how this, you can check this using this equation. So if you are not changing anything, means you are not changing CL, you are not changing S, you are not changing W and you want to make the same weight. Density will decrees. So if you use the initial velocity 35 m/s and if you calculate this thing, this will be not exactly equal to the weight. So if you increase the velocity, this will take care part of the decrease in density. So if you want to maintain the same lift at 5 km, the velocity will increase.

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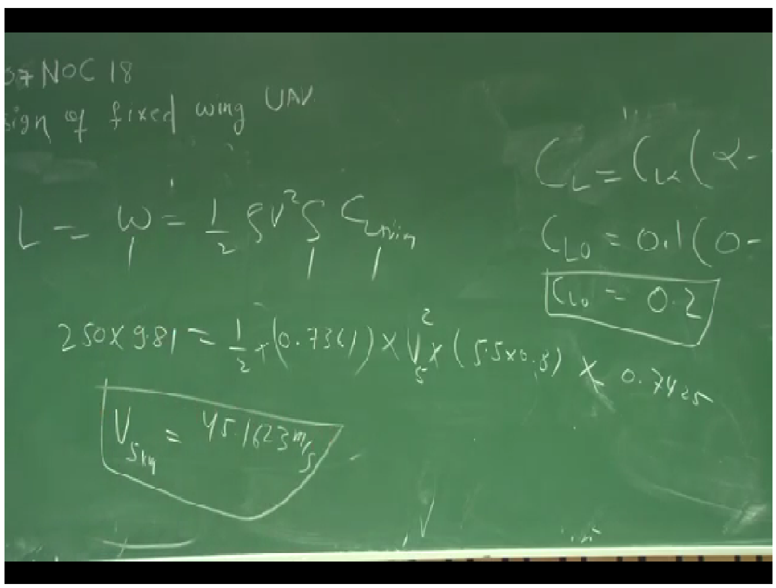

So you can see by using this, weight we have put here, 1/2, density you can use standard atmospheric relation, can find out, it will come out 0.7361. V you have to find at 5 km. Area is 5.5*0.8 and you have to multiply by the CL trim which comes out 0.7425. So this V at 5 km will be 45.1623 m/s.

So you can see approximate 10 m difference will be there. If you fly the UAV at sea level and UAV at 5 km and if you want to maintain the same weight. And in sixth question, the drag polar is given and you have to find the parasite drag coefficient. So by seeing the equation itself, you can directly say that this will be the parasite drag coefficient.

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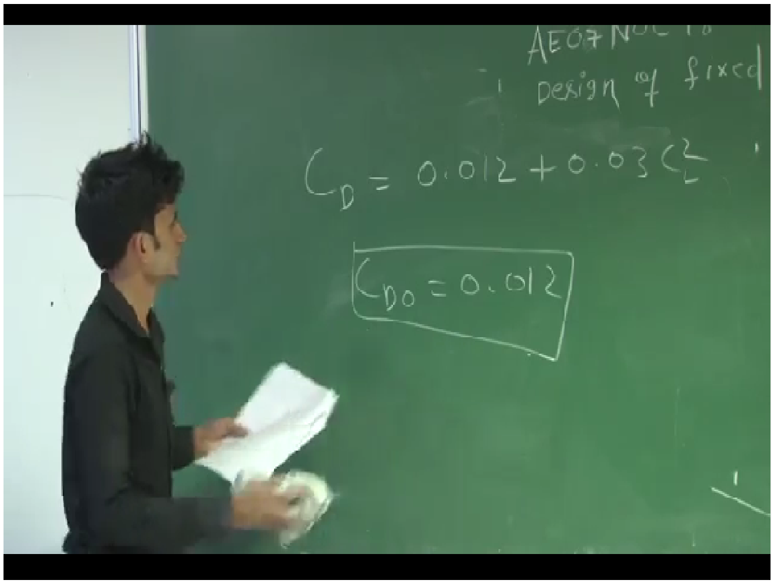

Basically this is CD=, what is given? 0.012, in sixth question I am talking about. 0.03 CL square, right. So this is basically parasite drag coefficient, CD0 and this is K and this is CL square. This is lift dependent; this is parasite drag coefficient. This value, so what will be the CD0? CD0 will be 0.012. So during the flight, this will change when you deploy the landing gear flap and these are the lift dependent things.

So if suppose that aircraft is flying and CD0 is gain, suddenly you deploy the landing gear or CD0, then this value CD0 will change and sir already explained you about that things. So the sixth question was very straightforward, you can by seeing that question itself, you can answer this.

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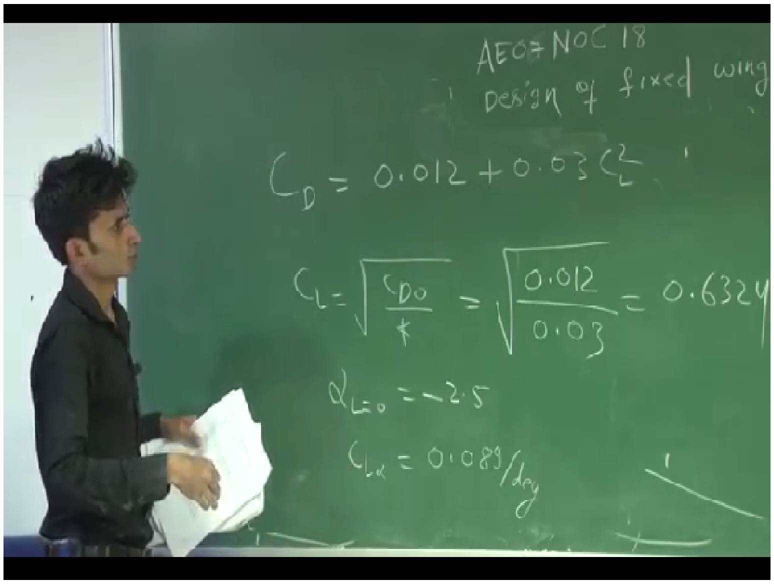

And you have to find the CL for the same maximum condition, so maximum L/D. So CL you can find using under root CD0/K which we have derived earlier. So this will be the 0.012/0.03. This will come out 0.6324. This will be your answer. Not answer, you have to find the value of trim angle of attack, I think. Seven, right, yes. You have to find the trim angle of attack or angle of attack at maximum range condition.

So you know that at maximum range condition, L/D will be the maximum. You have to use same formula to get the CL. So you got the CL and some other information is also given, like alpha lift=0 is also given -2.5. Lift curve slope also given, CL alpha is given 0.089/deg. So alpha lift=0, CL alpha is given, CL you have found out, so you can use this relation. CL is given, CL alpha is given, alpha lift=0 is given, you have to find alpha. Then I can use same relation here.

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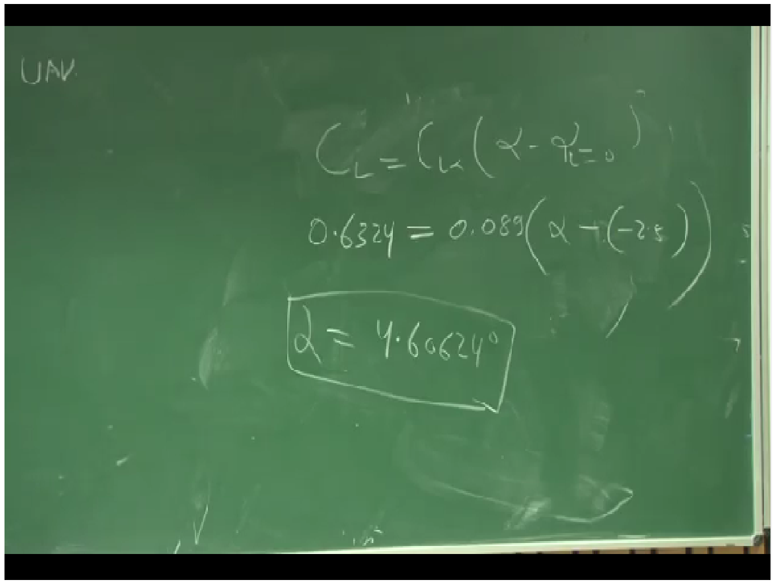

So CL comes 0.6324. CL alpha is given 0.089, you have to find this value alpha and this is also given -2.5. So your alpha will be alpha or alpha trim will be 4.60624 degree. This will be the angle of attack of the aircraft. And yes, and in eight question, the same lift curve slope is given, same alpha at lift=0 is given and the angle of attack at minimum power condition. In assignment we have missed the word maximum.

We have given the maximum power mentioned but it is minimum power condition. So like at minimum, like when L/D is maximum, means this is the maximum range condition and for the minimum power condition, I will explain about that how to derive the CL and CD for the minimum power conditions, then you can use or you can use directly formula.

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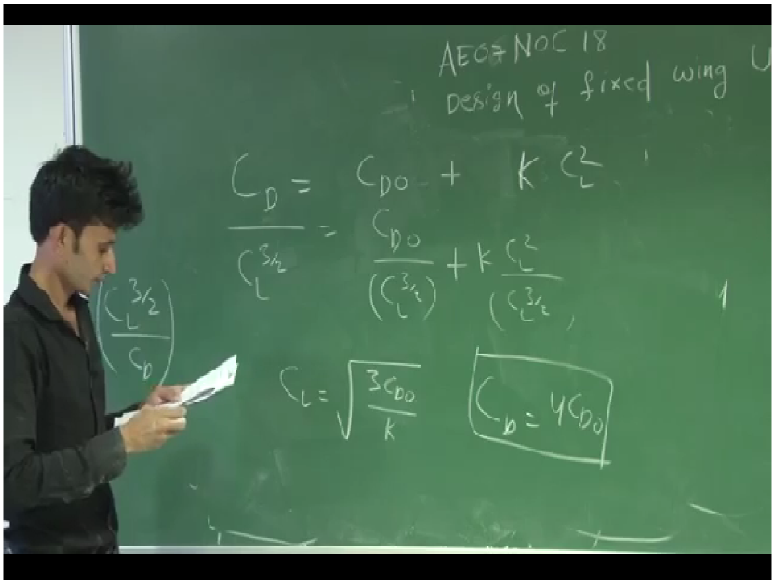

So just I am giving you the hint that, okay this CD=CD0, you can write generalized form. This is CD0 and this is K. So sir already told you that CL3/2/CD is the minimum power condition. So to get CL3/2/CD is the minimum power condition, then CD/CL3/2 will be the maximum power condition. So you can rearrange these things. You have to put this thing in denominator, sorry in right hand side you have to manipulate this equation like this, like we put that before L/D, we divided this thing by CL and we got one ratio here CD/CL.

And we assumed it is X and we differentiated with respect to the CL and we got that CD=2CD0 and so here you can multiply by CL3/2 actually. If you multiply by CL3/2, +KCL2/CL3/2, differentiate this, equate to 0, you will get CL=under root 3CD0/K and CL, if you put this CL into this equation, you will get CD=4CD0. So this is the minimum power condition. So in question seven, you have to find the angle of attack, same angle of attack.

So these things will change. CL will change. Because this comes out using under root CD0/K and for this question, the CL will be under root 3CD0/K. So in this equation, you can multiply under root 3, you will get the CL and use the same equations. CL will be under root 3CD0/K, means root 3*these times.

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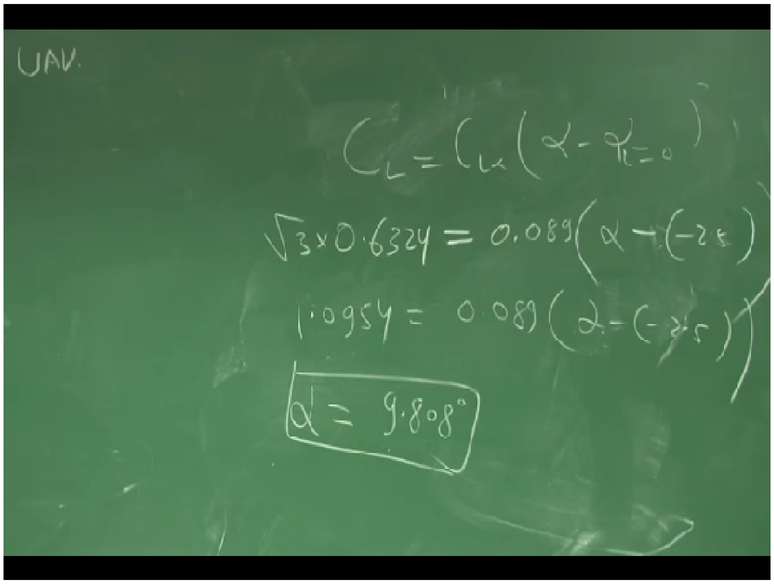

This will be 1.0954=0.089 alpha- -2.5. So your alpha will be 9.808. You can see that angle of attack at minimum power condition is more than the angle of attack at maximum range condition. So in ninth question, the true aircraft speed is 50 m/s at minimum drag condition at altitude where the value of atmospheric density, density is given directly 0.364724. If the weight of the aircraft is 7500 Newton and lift to drag ratio of the aircraft is 13 and wing span of 13 and mean aerodynamic chord of 1 m, the span efficiency factor, e you have to find.

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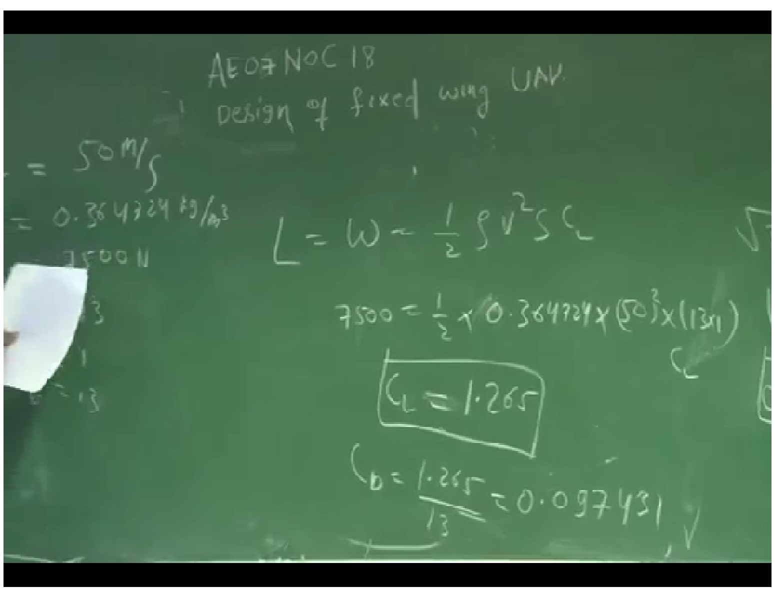

So true speed I am writing VT, 50 m/s is given and rho is directly given, no need to calculate, 0.364724 kg/m cube. W is also given 7500 Newton. L/D is given lift to drag ratio 13, C bar is given 1, b is given 13. So the wing is rectangular. So S will be b*C bar. Use the lift=weight and find the value of CL. I will tell you why we have to go first CL. 1.2 rho V square S*CL, then this is 7500, 1/2, density is given 0.364724, 50 square*13*1, this will be the area and this will be the CL.

So CL will come out, lift coefficient will be 1.265. This will be the CL. What will be the CD? CD will be CL/13. CL you got, so you can, 1/265/13. So you will get 0.097431. This is your value of CL and CD using this given data. So at minimum drag condition, CD will be 2CD0 and CL will be 3CD0/K. So you can find the value of CD0. I will tell you why we are doing these things to find the span efficiency factor.

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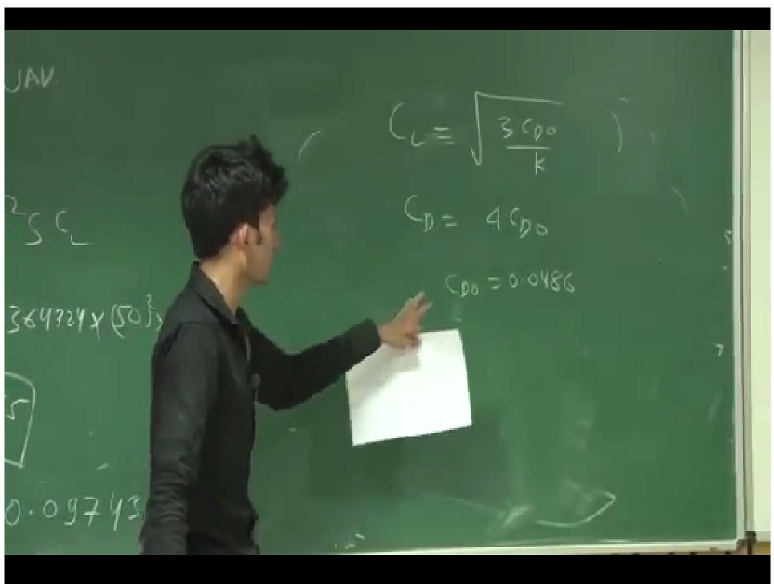

So if you use CL=under root 3CD0/K and CD=4CD0, you can put CD here. Use this CD to find CD0 and your CD0 will be 0.0486. So you have to find the span efficiency factor at minimum drag condition.

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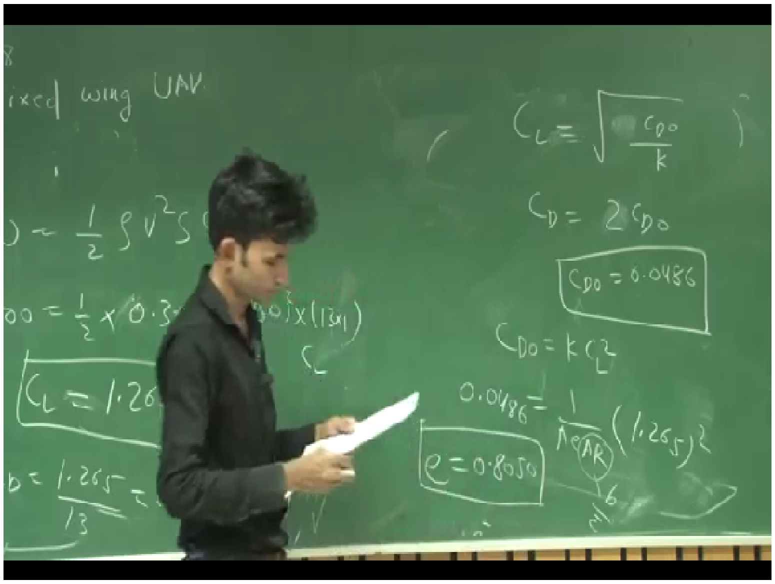

So at minimum drag condition, this 3 will not be there and this 4 will not be there. It will be 2. So I thought it is the minimum power condition but in question, it is minimum drag condition. So minimum drag condition, this is the minimum drag condition, CL=under root 3CD0/K and CD=2CD0. So This comes out CD0 equal to this much, okay. And you know that CD0=KCL square.

Square this equation, you will get this, okay. What is K? K is 1/pi*e*aspect ratio, right. And CL is what? You can find the value of CL already, 1.265 square, aspect ratio. Aspect ratio, this you can write say b/C bar or you can finally, so 13/1 will be the 13. So 13 you can put directly, pi you know. CD0, we have calculated that, 0.0486. CL square, CL is given. So you can take e this side and you will get e as 0.8050.

So basically you have to think that where the e is coming. So if you think that where e is coming then you can get K is 1/pi*e*aspect ratio. Where the K is coming? K is coming in the CL equation. Then you have to find the CL. That is why we have found that CL is 1.265. Then CD is 0.5 and use this, you can get. We got in this equation, CD0 also is given, is required to solve this. So CD0 we know and CD=2CD0, CD also we have to find.

Then you will get the CD0 and CL also we have to find to get the K. So if you know the CL, if you know the CD0, you can square this equation. This equation becomes CD0=KCL square. Put this CL using this, the data which we got here and CD0 using this and put the aspect ratio value, you will get the e value. This is all about the assignments. We have made some more problems, some more realistic problems. Sir told me to make this type of problem.

So that you can feel some more realistic problem. So that we will discuss in the next class, next today. Thank you.