Design of Fixed Wing Unmanned Aerial Vehicles

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Lecture – 22
Tutorial 2

Hello everyone. Welcome back to this course called design of fixed wing UAV. Today we will discuss some problems which we made based on the lecture, was based on the last lecture which sir talked to you. So let us start with the problem.

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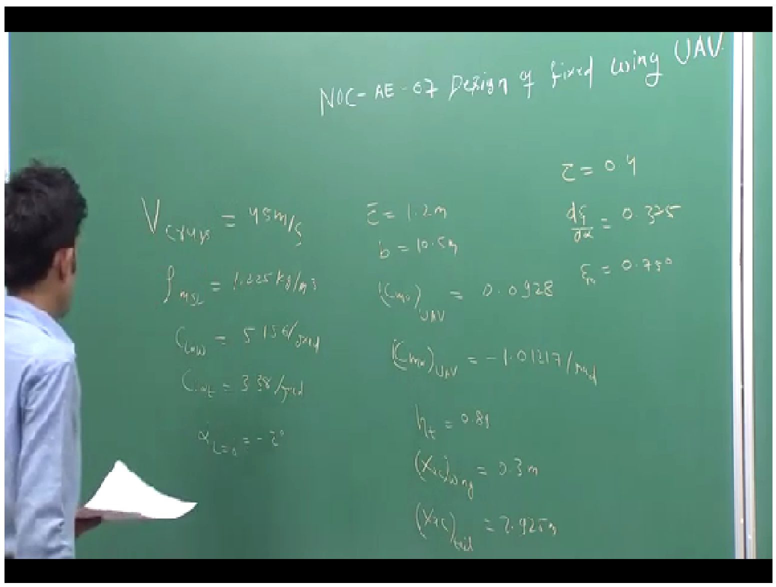

So the data is given cruise speed is 45 m/s and density at mean sea level is 1.225 kg/m cube, CL alpha of the wing is given 5.156/radian and CL alpha of the tail also given 3.38/radian and alpha at lift=0 is -2 degree. We can write some data here. Mean aerodynamic chord is given 1.2 m, span of the wing is given 10.5 m and Cm0 of the whole UAV is given 0.0928 as well as Cm alpha is given of complete UAV is -1.01217/radian.

Tail efficiency factor 0.89, Xac of the wing 0.3 m and Xac of the tail is given 2.925 m. So basically we are measuring this distance with reference to wing leading edge. So this Xac is nothing but the distance from the leading edge and this Xac tail also from wing leading edge. Tau is given 0.5, do epsilon/do alpha is given 0.375, epsilon 0 is given 0.75. With this information, it is given that effect of the fuselage on the stability is negligible and some useful formula also given.

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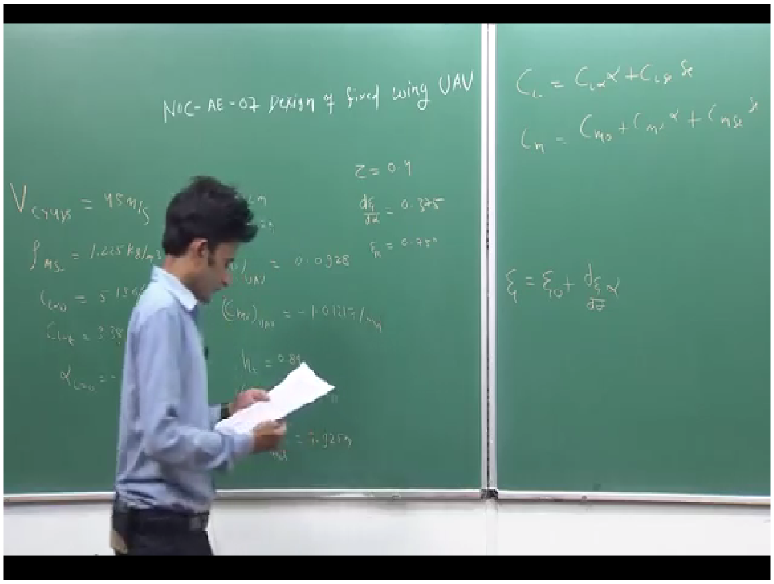

Like CL=CL alpha*alpha+CL delta*delta e and Cm=Cm0+Cm alpha*alpha+Cm delta*delta e. Where delta is elevator deflection. So suppose that if you want to consider only stability, then this part is only there. But if the controls will come, then this part will add. Because when you deflect the elevator, then CL will change and this is Cm=Cm0+Cm alpha*alpha. This is when you consider only stability.

When you consider the control, then Cm delta*delta e will come where Cm delta is the elevator control power and delta is the elevator deflection. So nomenclature has their usual aerodynamic meaning. V is speed, rho is the density, the lift curve slope of wing, lift curve slope of tail, angle of attack when lift=0, mean aerodynamic chord, span, pitching movement coefficient at 0 angle of attack, pitch stability derivative of whole UAV.

Tail efficiency factor, this is nothing but dynamic pressure on the tail/dynamic pressure on the wing. Xac, location of aerodynamic center of wing, location of aerodynamic center of tail. This is downwash angle, do epsilon/do alpha is, you can if you find using Taylor series, downwash angle=downwash angle at 0 angle of attack+do epsilon/do alpha*alpha. So we will solve first problem.

The first problem is find the location of neutral point of the above fixed wing UAV from the winged leading edge. So in first question, you have to find the neutral point. So you have to find the neutral point. See the information is given. Using this information, you can find. First you have to think that in which equation the neutral point will come?

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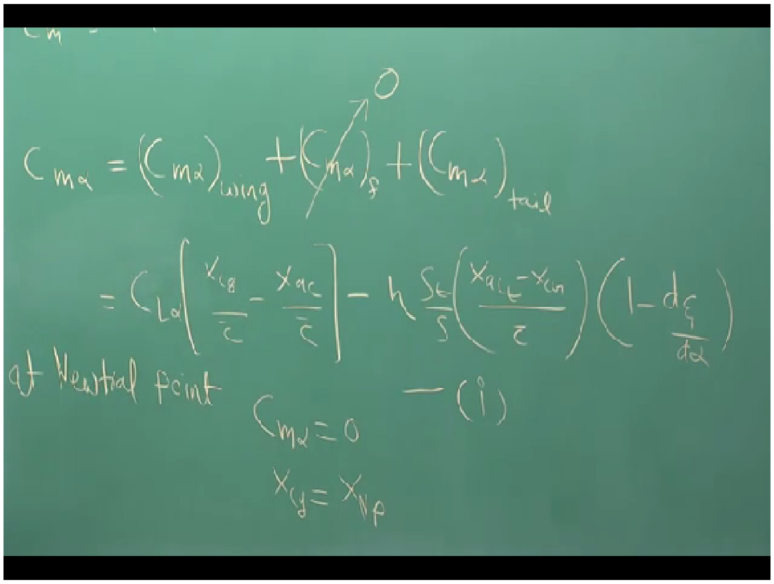

So it is obvious that when you write Cm alpha of the whole aircraft or whole UAV, is nothing but equal to Cm alpha of the wing+Cm alpha of the fuselage+, third part is, Cm alpha of tail. But it is already given that Cm alpha of the fuselage is 0 because in the question it is given that neglect the effect of fuselage on stability. So this will be 0. And what is Cm alpha of the wing? It is nothing but CL alpha Xcg/C-Xac/C.

And this is, Cm alpha of tail is nothing but -eta St/S Xac of tail-Xcg/C bar*1-do epsilon/do alpha. This is the complete expression of Cm alpha. So you have to find the neutral point. So at neutral point, your Cm alpha will become 0. Cm alpha will be 0 and Xcg will reach at neutral point. So put Cm alpha=0 and Xcg=XNP in this equation. Let us make equation 1. So what you will get?

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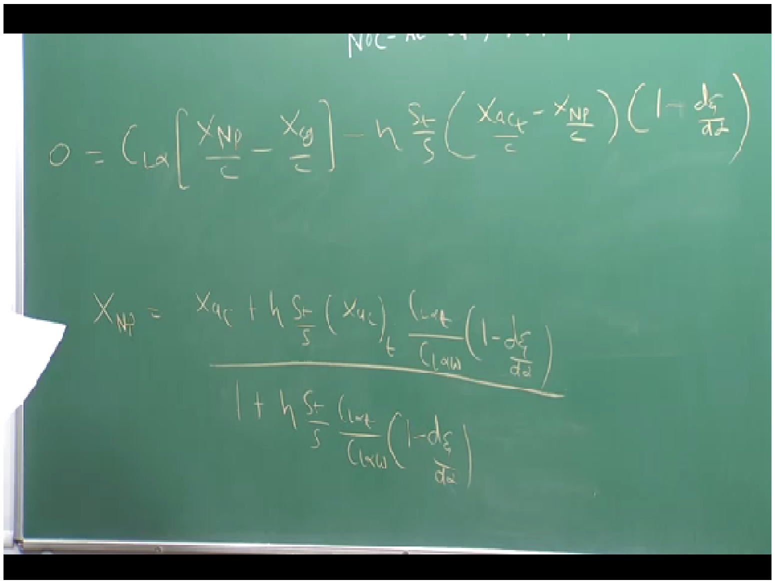

0=CL*XNP/C-Xcg/C-eta St/S Xac of tail-Xcg, Xcg becomes XNP, right and 1-do epsilon/do alpha. So if you rearrange this equation, then what you will get? X neutral point will be Xac+eta St/S Xac of tail*CL alpha of the tail CL alpha of the wing 1-do epsilon/do alpha/1+eta St/S CL alpha of the tail, CL alpha of the wing*1-do epsilon/do alpha, right. You will get this expression. So if you put all these values, what you will get? So just put each and every value from the data which is given.

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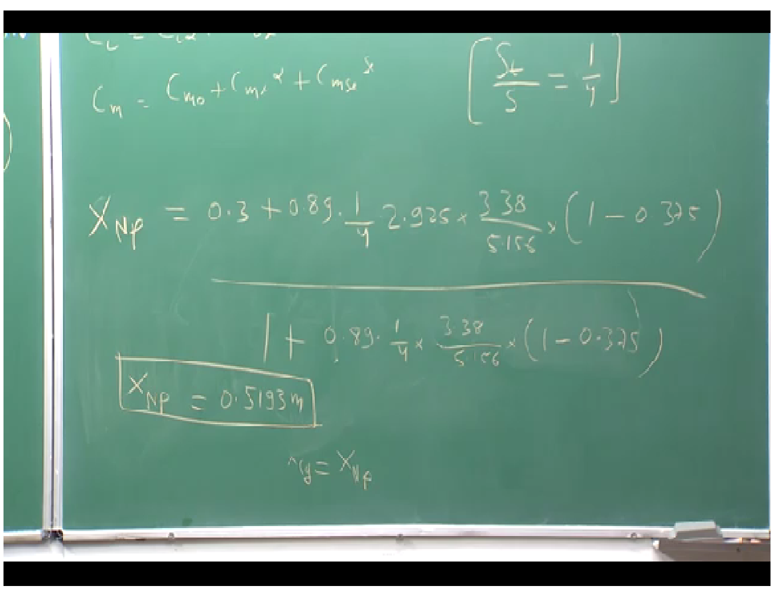

So you Xac is nothing but XNP=, Xac is given 0.3, +eta is given 0.89 and it is also given in the question that the tail area is 1/4th of wing area. So your St/S will be 1/4. Your tail area is 1/4th of wing area. So St/S you can put 1/4. Next Xac of the tail is given 2.925 and CL alpha of the tail is given 3.38.

CL alpha of the wing is given 5.156 and 1-do epsilon/do alpha is given 0.375, okay/1+same thing except CLT 0.89*1/4*3.38/5.156*1-0.375. So when you solve this, you will get, XNP is nothing but 0.5193 m. So this will be the answer of that question. So in second question, you have to find the total lift curve slope of the UAV in per radian. To solve the second question, we will solve here.

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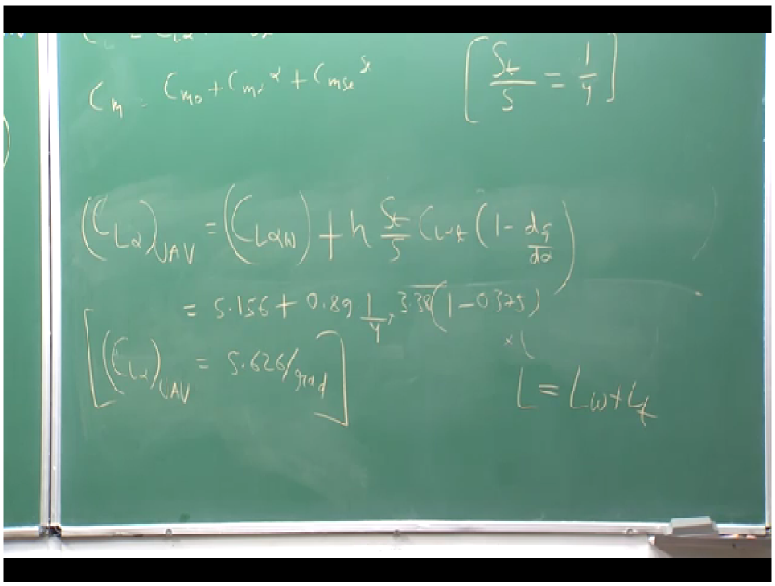

So total lift of curve slope of the CL alpha of the whole UAV is nothing but CL alpha of the wing+eta St/S CL alpha of the tail 1-do epsilon/do alpha. Sir had already given to you the derivation of this CL alpha of the complete aircraft equal to, see how you will get the CL alpha of the complete aircraft? Using CL alpha of the wing and CL alpha of the tail. If you derive, if you use the first principal, that means lift=, in this case it means, total lift of the aircraft will be the total lift of the wing+total lift of the tail.

So if you use this relation, total lift of the UAV=lift of the wing+lift of the tail, you will get this. So CL alpha of the wing is given 5.156+eta is given 0.89 and St/S is given 1.4. CL alpha is given 3.38*1-0.375 which is do epsilon/do alpha. So here you will get 5.626/rad. This is nothing but CL alpha of the complete UAV. So you can see that earlier your CL alpha of the wing was, okay, earlier the lift curve slope was 5.156 when, suppose that there was no tail.

But if you add the tail, then total lift curve slope of UAV will increase. So here is 5.156, here is 5.626/rad. So in the third question, you have to find the static margin. So you know that static margin is the distance between neutral point and Xcg with respect to the mean aerodynamic chord.

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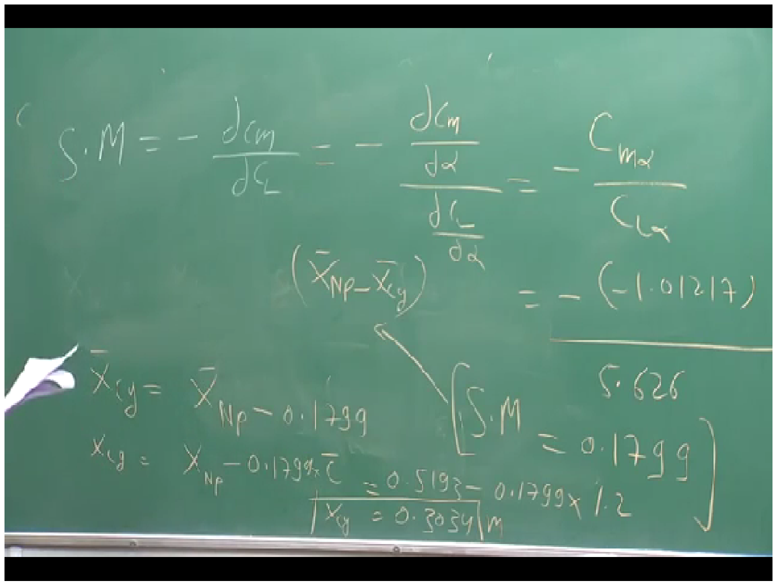

So static margin is nothing but -do Cm/do CL and also write static margin a term of Cm and CL. So this expression you can manipulate, this equation like do Cm/do alpha/do CL/do alpha. This you can further write Cm alpha and CL alpha, right. Cm alpha, it is given in the question -1.01217, right and CL alpha you have found out 5.626/rad. This is also per radian, this is also per radian.

So your answer will come 0.1799, static margin which is also distance between neutral point and Xcg with respect to the mean aerodynamic chord, okay. So this is the answer of third question and in fourth question, you have to find the location of center of gravity. So in this case, you know that okay you have found out the position of neutral point, you have static margin also. Use this relation, you will find the Xcg.

So if you use this, what will be your Xcg? Xcg is nothing but Xcg bar, is nothing but X neutral point bar, -this 0.1799, static margin. This you can write Xcg/C and multiply C in this side. What you will get? You can write X neutral point/C, so C C will cancel. So X neutral point only will be there and 0.1799*C bar, right. So this neutral point you already found out, how much it is? 0.5193, neutral point, -0.1799*, C bar is 1.2, right.

So your answer will come 0.3034. This is the position of center of gravity of UAV from the wing leading edge. This is the answer of your fifth question. So in sixth question, you have to find the, okay this is in meter. So in sixth question, you have to find the tail volume ratio. So basically tail volume ratio tells us how much stability UAV or aircraft has.

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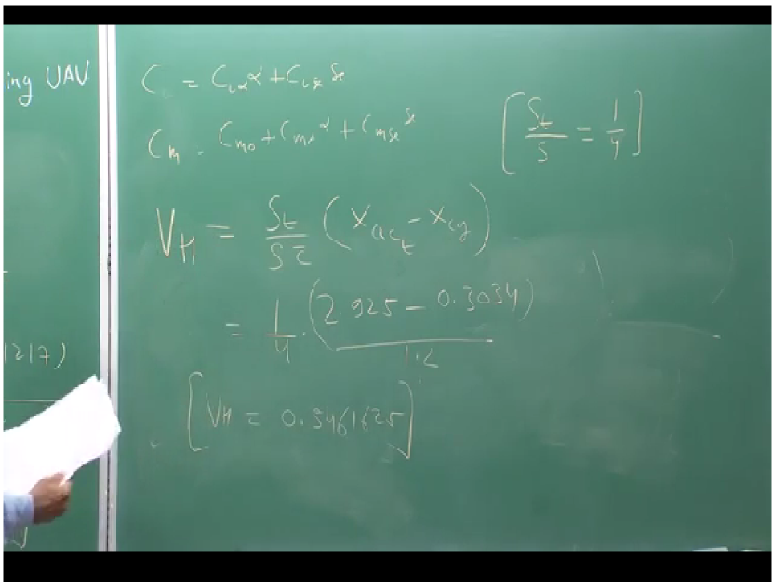

See tail volume ratio, tail volume ratio in this case is horizontal tail volume ratio, is St/S C bar and tail arm which is nothing but Xac of tail-Xcg. So St/S is given 1/4 and Xac of the tail is 2.925, right and Xcg is 0.3034/1.2, C bar, okay. So what you will get? You will get approximately 0.5461625. This is nothing but your horizontal tail volume ratio. So in sixth question you have to find the tail setting angle, how much tail setting angle is required in order to get the desired Cm0 value. So let us solve this also.

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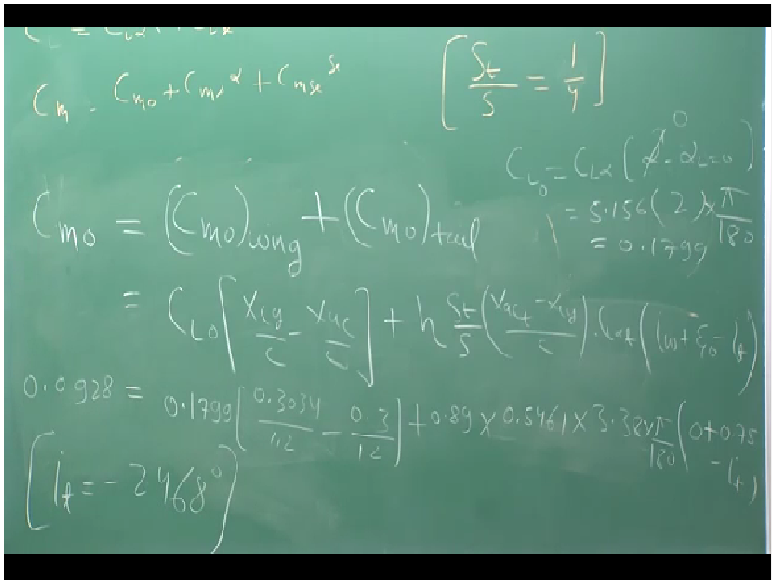

So you know that think about that okay in which equation the It is coming, tail setting angle. So it is obviously Cm0. So Cm0 is nothing but Cm0 of the wing+Cm0 of the tail, okay.

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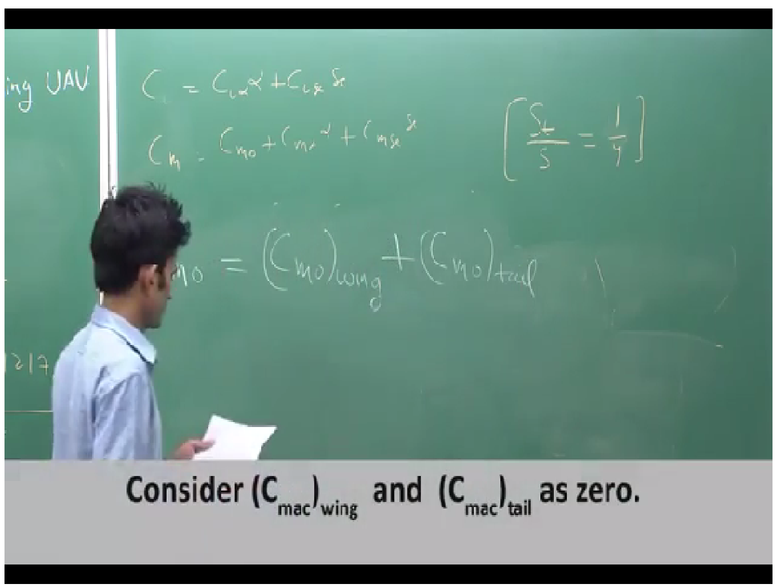

So Cm0 further you can write Cm0 you can further write CL0*Xcg/C-Xac/C and what is the Cm0 of the tail? It is eta St/S*Xac of tail-Xcg/C*CL alpha of the tail, Iw+epsilon 0-It. So this is nothing but your Cm0 of the whole UAV or aircraft. But in this question, it is given that Cm0 of the UAV which is 0.0928. What is CL0? CL0 you can find using this relation. CL=CL alpha of the wing alpha-alpha lift=0.

So if you put alpha=0, what is known? CL alpha. CL alpha of the wing is nothing but 5.156, it is per radian and what is alpha lift=0? It is -2 degree. So minus minus will become +2 degree. You convert this into degree. You will get approximately 0.1799. So this is in per radian. This is in degree that is why we have converted this into per radian. So per radian and radian will cancel. So you answer will be 0.1799, CL0.

If you are putting alpha=0, you have to put 0 here. So you can put this value here, 0.1799. Xcg you already found out, right. How much it is? 0.3034/C bar, 1.2-Xac is nothing but 0.3/C bar, 1.2+0.89, okay. This thing St/S Xac tail-Xcg/C, you can write tail volume ratio, right. This we already defined. So tail volume ratio is nothing but which we got, is how much? 0.5461, just take 4 digits.

CL alpha of the tail is 3.38. Be careful that it is per radian. So you can convert this into per degree then. It will be easy because here you will put in degree always number. So (()) (22:49) angle is 0 and what is epsilon 0? It is given 0.75 degree, -It. Here be careful that CL alpha of the tail is in per radian and epsilon 0 and It is in degree. You want the answer in degree. So you have to convert these things per radian into per degree.

That is why we multiply this. So if you solve this, you will get ideally approximately tail setting angle is nothing but 2.468 degree. Thus you have to solve this. You put in all the numbers, just there in this. So in seventh question, with this given information, what will be the total lift coefficient at 0 degree angle of attack? Means you have to find the CL0 of the complete UAV. See CL0 of the wing we have already found out, 0.1799. So when you add the tail, then how much CL0 the tail will produce, you just have to find out and add this to, you will get the answer. So let us find this also.

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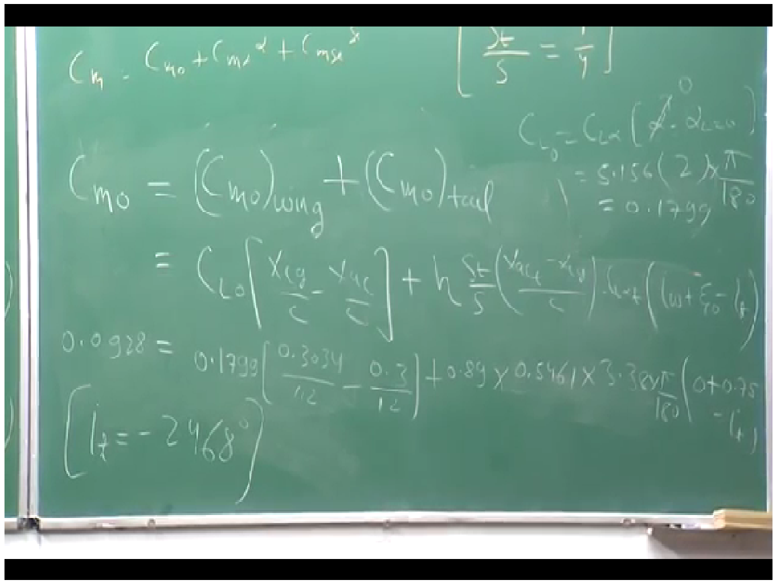

So CL0 of the complete UAV=CL0 of the wing+CL0 of the tail you can say. So what will be the CL0 of the wing? You have calculated. Let us put first formula and then we will further put proceed, right. And CL0 of the tail is nothing but eta St/S CL alpha of the tail It-epsilon 0. So this is nothing but, we have calculated that what is the CL0. It is coming. So CL0 is nothing but, we have found out 0.1799, right+eta is 0.89*St/S is 1.4, CL alpha is 3.38 which is in per radian.

So convert into per degree because here you will put the It and epsilon 0 in degree. So -2.4 something you got, 2.68-0.75. So if you solve this, you will get CL0 of the complete UAV 0.13351. So you can make the observation that okay when you add the tail, then CL alpha of the complete UAV will be more than the CL alpha of the wing alone.

But CL0 of the complete UAV is less than the CL0 of the wing alone. So in eighth question, you have to find the trim lift coefficient of the whole aircraft. So this is very straightforward. When you are flying at trim condition means your forces and moments are balanced. So that means you can write thrust=drag and lift=weight. So we want CL.

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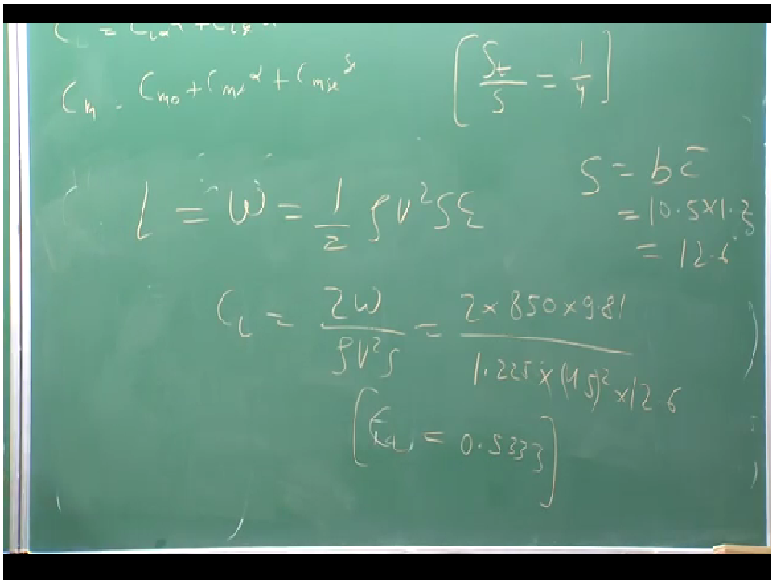

So we will use lift=weight because in that expression, the CL will come. So use this relation 1/2rho V square S*CL. So rearrange this CL=2W/rho V square S. So put this value, 2*, the weight is given 850. So please note down that 850 kg is the total weight because at the beginning of this lecture, I think we missed this weight. So please note down this. So weight will be 850 kg and rho mean sea level 1.225, take 3 digits.

V speed is given already 45, is V square, 45 square and area of the wing is nothing but this is also given that, you have to take the rectangular shape of the wing. So your area will be nothing but b*C bar. So b is given, how much? It is 10.5 and C bar is given 1.2. So your area will come around 12.6. Let me check. Yes 12.6. So 12.6. So if you solve this, you will get 0.533, is your CL value, okay.

This is the eighth question. So in ninth question and tenth question, you have to find the trim angle of attack of the aircraft and to trim this aircraft at that particular angle of attack which we will find in next question, how much the elevator deflection is required in order to trim the aircraft? So we will solve this. See whenever you deflect the elevator, the aircraft will response. If the elevator goes up, the nose part will go up.

If the elevator will go down, then the nose part will go down. So basically you are controlling the plane in longitudinal motion using elevator. So basically when you use the elevator means you are controlling the angle of attack. You are trimming the angle of attack at different different angle of attack with different different velocity. So is there a significance of this question. So let us solve this question.

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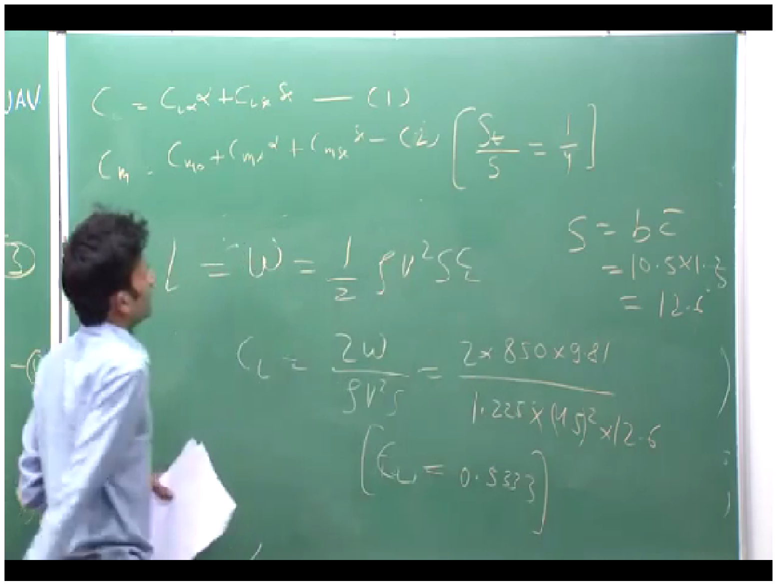

So you know that, okay if you observe this equation, CL=CL alpha*alpha+CL delta*delta e. This CL alpha is nothing but complete CL alpha of the air vehicle which we have already found. This is the angle of attack. This is the, when you change the elevator angle, how much CL will change? So this stability will tell. This stability derivative will tell. And this is the elevator deflection.

In this moment equation, Cm=Cm0+Cm alpha*alpha. This Cm alpha is nothing but Cm alpha of the whole air vehicle. This Cm0 is the Cm0 of the whole air vehicle and plus this Cm delta depends upon, okay, how much air craft will respond, like in terms of moment when you deflect the elevator. This Cm delta is nothing but elevator control power. So whenever you are trimming, means you are balancing the moment, means net moment will be 0.

So the Cm0 will be 0. So use this equation 1 and 2, this make 1 and this 2, you will get the answer of ninth and tenth question.

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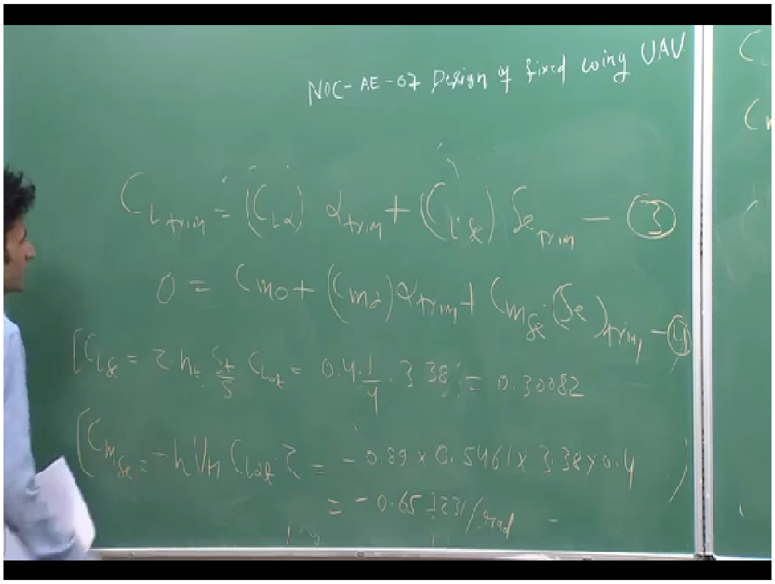

If suppose that aircraft is trim, so then equation 1 becomes CL trim=CL alpha of the whole air vehicle*alpha trim+Cm delta e*delta e trim, right. And your second equation becomes, see aircraft in a trim condition, net moment will be 0, means Cm will be 0. 0=Cm0+Cm alpha*alpha trim+Cm delta e*delta e trim. So you can make equation 2, 1 and 2 you already made it, so let us make 3 and 4, okay.

So seen this equation and you just figure out that okay how much unknown we require? Which variable we do not know? See CL trim you know, you already found out 0.533. CL alpha you know. Cm alpha you know. Cm0 you know. Alpha trim and delta e trim, you have to find. 2 things you do not know, CL delta e and Cm delta e. So let us find the Cm delta e elevator control power and this CL delta e using this equation.

Thus your CL delta is nothing but tau eta St/S*CL alpha of the tail. So tau is nothing but 0.4. St/S is nothing but 1.4. CL alpha of the tail is nothing but 3.38/rad. So your CL delta e will come approximately 0.30082. Similarly, you can find Cm delta e, elevator control power. It is -eta VH, you can St/S*tail arm/C bar. So -eta VH CL alpha of the tail*, you can multiply by tau. So you will get the answer 0.89*, what is your VH? 0.5461 I think.

Your eta VH CL alpha tail is 3.38/rad*tau, okay. So your answer will come -0.657231/rad. So you know CL delta e, you know Cm delta e. So you have 2 equation, 2 unknown, you can solve easily. But if you write equation 3 and 4 in matrix form, then it will be very easy to solve this. So let us write equation 3 and 4 in matrix form. So when you write in matrix form for equation 3 and 4, what you will get?

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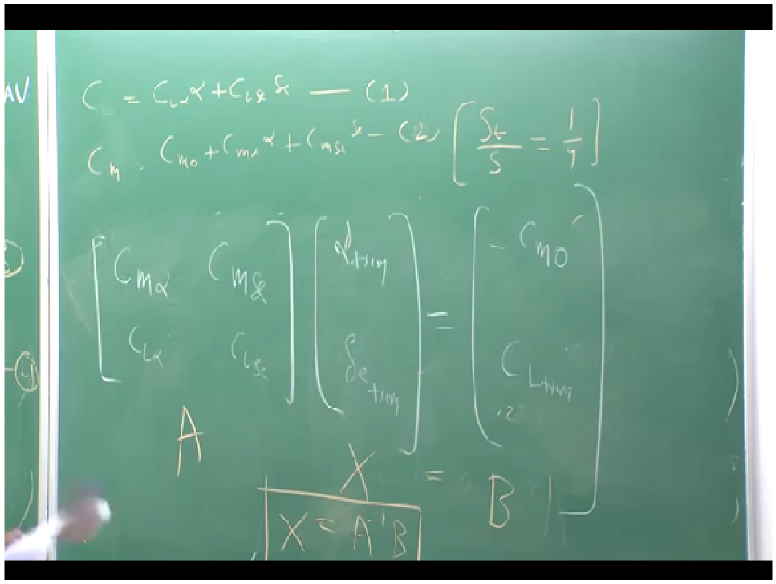

Cm alpha Cm delta e CL alpha CL delta e. That is one matrix. Multiply by the unknown column matrix, alpha trim delta e trim=, if you put Cm0 in that side, this will become -Cm0 and CL trim in 3 equation that side, CL trim will come, yes. So if you multiply by this and this compared with this, you will get equation 4. If you multiply this row to this column, equate it to this, you will get the equation 3.

So basically you can say that okay this is matrix A and this is matrix X, unknown and this is matrix B. So you want X, right. X will be A inverse B. This will be your solution. So in this equation you already know Cm alpha Cm delta e which we found just before. CL alpha you know. CL delta e we found just before. Alpha trim, delta e trim we want. Cm0 we know. CL trim we know. So put all the value in this matrix, take the A inverse, multiply by the B matrix, you will get the answer. So inverse easily you can take using this.

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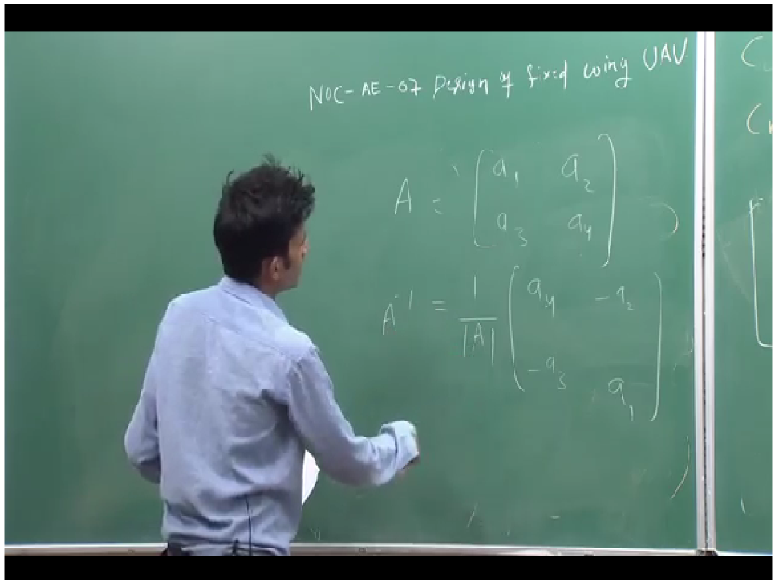

Suppose that you have 2 elements, 4 elements a1 a2 a3 a4 and this will be A and if you want to take the inverse. In this case, you have to take the inverse of this right. So inverse is nothing but this, divided by determinant A, okay. Change this a4, interchange these 2 and put - sign here. -a3 -a4, you will get the A inverse.

So these things you can change. These 2 interchange, put these things with -sign, take the determinant, divided by determinant, you will get the A inverse. This will be very easy to solve, okay. So A inverse is nothing but adjoint of A divided by determinant of A which we have written. So what will be your A inverse will come?

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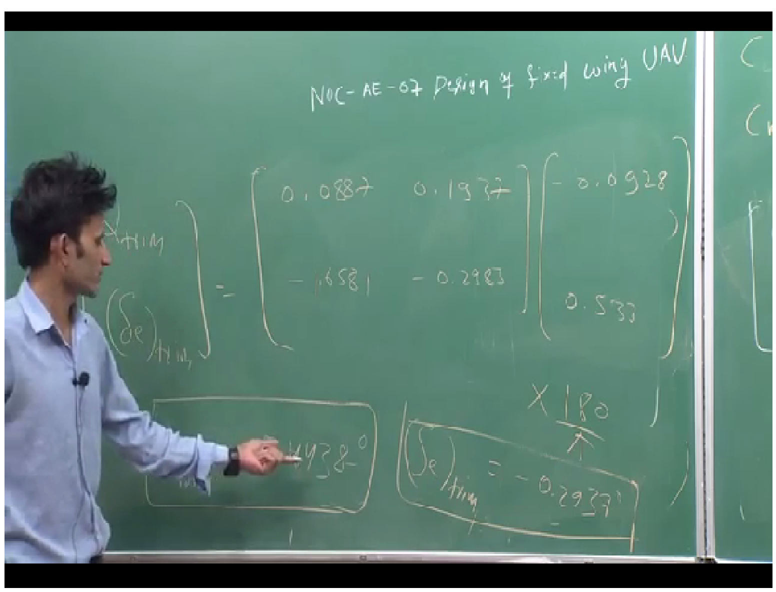

So your X, X is nothing but alpha trim and delta e trim=A inverse, right. A inverse it will come 0.0887 -1.6581 0.1937 -0.2983 0.0928 0.533. See these things, you are calculating in the radian because if you are putting this Cm alpha Cm delta e CL alpha CL delta e per radian and we want the answer in degree.

So you can multiply by 180/pi. So what you will get? Alpha trim, trim angle of attack is nothing but 5.4438 degree and this is your answer of ninth question and delta e trim will be -0.2937 degree. This is your delta e trim. The - sign indicates that you are deflecting the elevator in upward direction in order to trim the aircraft at 5.4438 degree. So this is the answer of tenth question.