Aircraft Stability and Control
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Lecture-57
Tutorial-2

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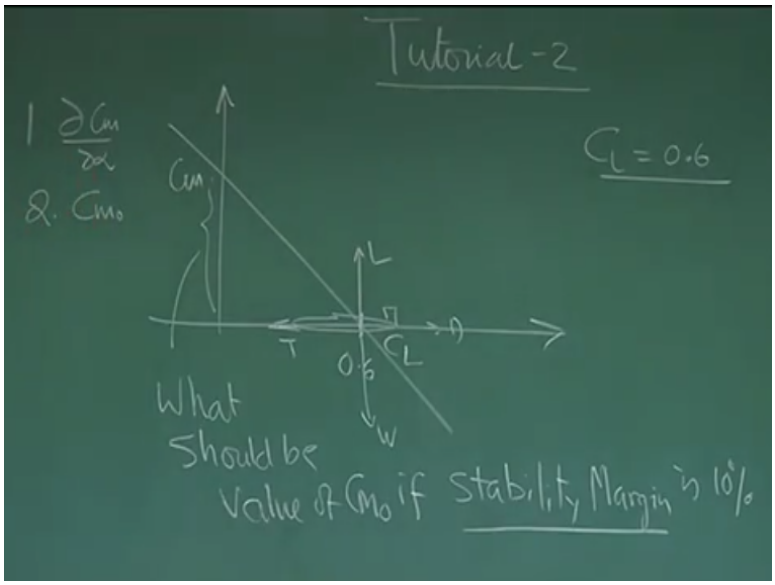

Good morning let us do one problem let us solve one problem I say tutorial -2 and we like to see whatever we understood can you apply this by solving simple, simple problems are not ok. What you understood one thing d c m/d alpha with static stability parameter and that is c m not. So let me create a problem. This is cm let see this is cl and it is let us frame the problem that we want to fly an airplane such that.

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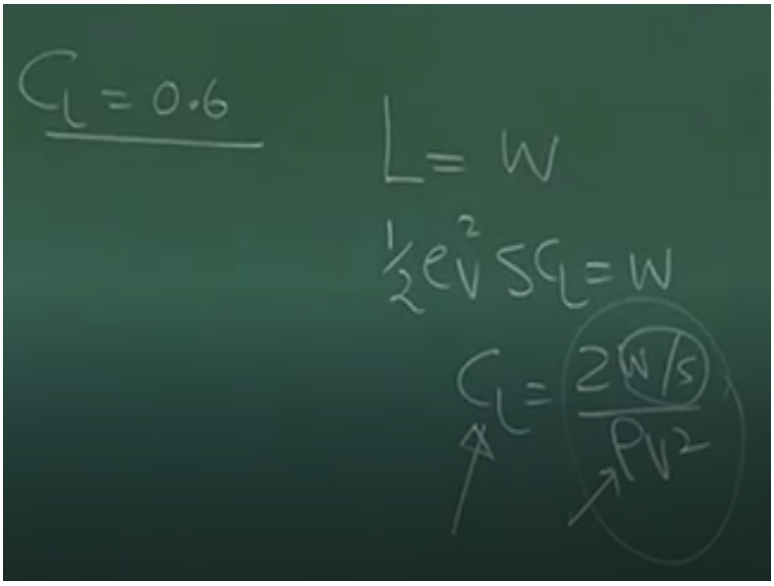

The lift coefficient or cl =0.6 where from you get cl 0.6. After all what is point is cruise that is your equilibrium. So lift=weight I know ½ rho square s cl = weight ,cl=2w/s/rho v square. Suppose I am designing an air plane whose wing loading we have decided, we have decided what is the cruise speed and we have decided what is the altitude at which you are cruising. So if I put those numbers here I will get a value cl and that value cl let's say all those geometrical parameters values 0.6 got it.

So if I try to visualize that this at this point. Lift and weight are equal and of course it is the cruise, thrust and dragger also equal ok. And it goes with the saying am is also equal to zero. Now the question is what should be the value of cm not if stability margin or static stability margin is 10% this is the question what is the question what should be the value of cm not if stability margin is 10%. And I am going fly at cl 0.6. The movement i mention stability margin it should come to your mind that.

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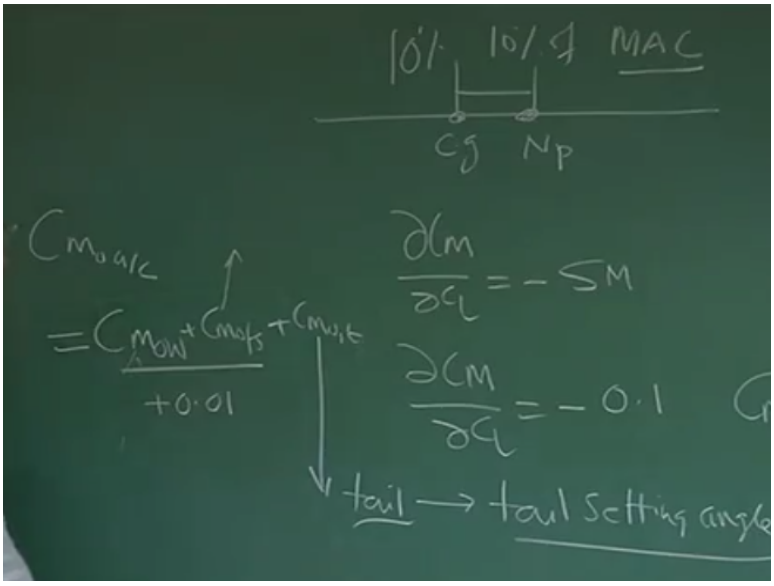

dc m/dcl is nothing but = - static margin we are developed this relationship with some approximately that we will carry on with. So what is the dcm/dcl =-0.1 is 10%. The 10% means what? it actually tells you what is the meaning of the 10% if this is the neutral point and this is the cg location and this distance is 10% of main aero dynamic chord ok. I know dcm/dcl is -0.1

So I know the slope which is here which is linear and same distance -0.1. So what is the cm not then? Cm not=0.1 *0.6 that is 0.06. So I need cm not 0.06. So if I have an air plane I should ensure that cm not of aircraft = cm not wing +cm not few large +cm not tail. What you understand? cm not =0.6 means because contribution from wing fuse large contribution the tail target up to 0.6.

Suppose this two contribution is not giving the 0.6 if is gives 0.6 fine plus .06 which is very difficult you understand mostly the limbered airfoil and that c m fuse large is negative all those small. What is the option? Suppose we have said this is 0.01let us say this. Then remaining point .05 must come from tail and how to get the cm not from tail. You know for tail setting angle.

You can use those formula and find out what is the tails setting angle required ? for making this whole cm not to 0.06 and catch point is here the right now let us further look in this problem.

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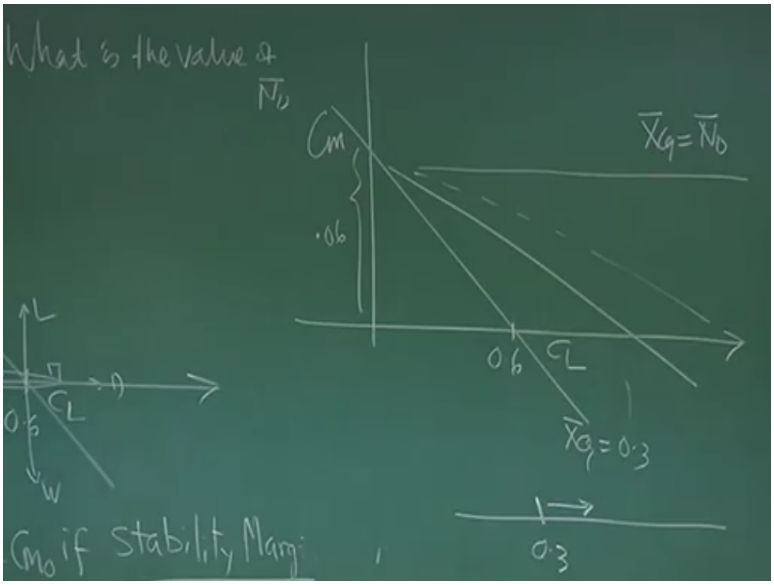

This is point 0.6 this is point.06 and let the x c g bar =0.3 x cg bar means xcg location non -dimensional at which a mini aero dynamic chord. Now if I move the cg backwards if the cg is here 0.3. Now I am moving the cg backwards what will happen as I move the cg backwards the slope will go on the reducing the time will come when the slope will parallel to the x axis and that c g location you all know that neutral point.

So what is the definition of neutral point is that c g location at which is dc m/dcl is zero right of the aircraft is neutrally stable in static sense. Now if I am given this plot x c g and cl and cm not. If I try to find out what is the value of n not I can easily find out that.

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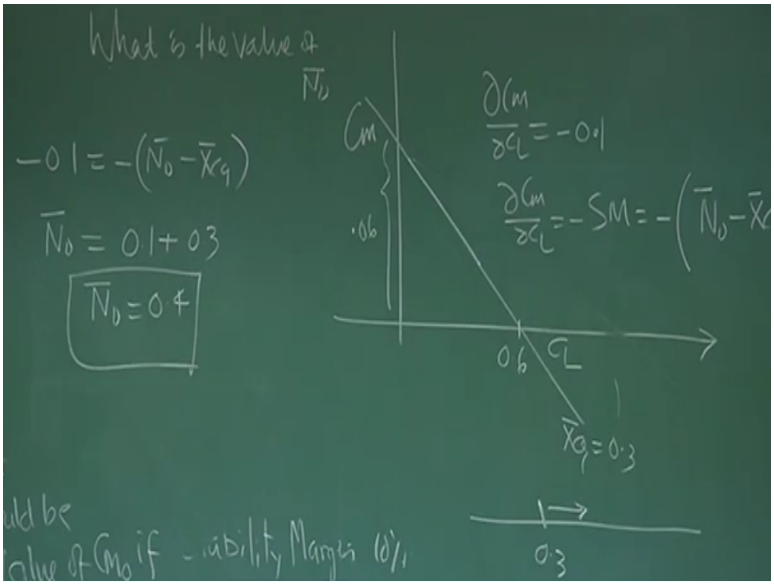

If this is the combination this is given what you know you know dcm/dcl=-0.1 and you know dcm/dcl minus static margin = -(n not – x cg bar) and you know dcm/dcl how much, dcm/dcl is static margin 10% we have mentioned here dcm/dcl is -0.1. So what I can write we can simply write -0.1=-(n not -x cg bar) ; n not bar. And so n not will be, n not bar will be=0.1+0.3 plus x c g. x c g is given .3 so your n not bar neutral point is 0.4 that is simple the reason is given like this cm versus cl x c g is given.

Here this values was given here usually find out what is the neutral point is in the airfoil ok. This is one of the simplest this which you are suppose to do in the sense that automatically come to your mind that this is such reasons. Now let us try to we little more creative.

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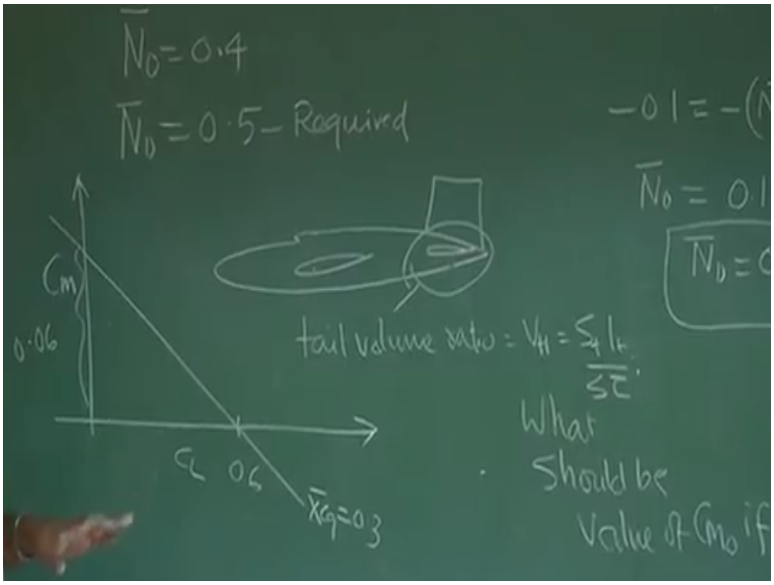

Let say we have got n not=0.4 as given the example but you want the airplane n not is equal to 0.5 that is what is required I am talking about the right what is to be done again visit this diagram. Graph this is cm this is cl this is x cg =0.3 and this is 0.6 and this is .06 most of the student start thinking change the cg and change the n not do is correct. Answer is no because cg nothing to do with n not. N not is decided by whom.

By the tail, tail sides aspect ratio with by wing size aero dynamic properties. Once the configuration is fixed n not is fixe. So by changing c g you cannot change n not. You can change the stability margin that is different issue. But our problem is we want to change n not. So what is to be done. So the best way to do is.This is the airplane this is tail this is the wing I am not changing the n not and the n not whatever number it is.

And the best way to deal this is change tail volume ratio that you remember that is vh st lt /s c bar. What is happening geographically? Please see this I want to change the 0.4 to 0.5. So, what I will do I increase the tail size. The increases the tail size drag also increase right. Better you take the tail little back. If it take further back then the condition of the whole airplane length will be increased even at the right click. The little bit of area little bit of length in effect re effect the volume ratio.

This will immediately change the n not value. So we can find out corresponding change in the vh required to make it from o .4 and .5 best way to handle is, change the tail volume ratio ok. That you should not forget ok. Once this is done. Let's also see another problem so that we can understand all this construction very clearly. Ok.

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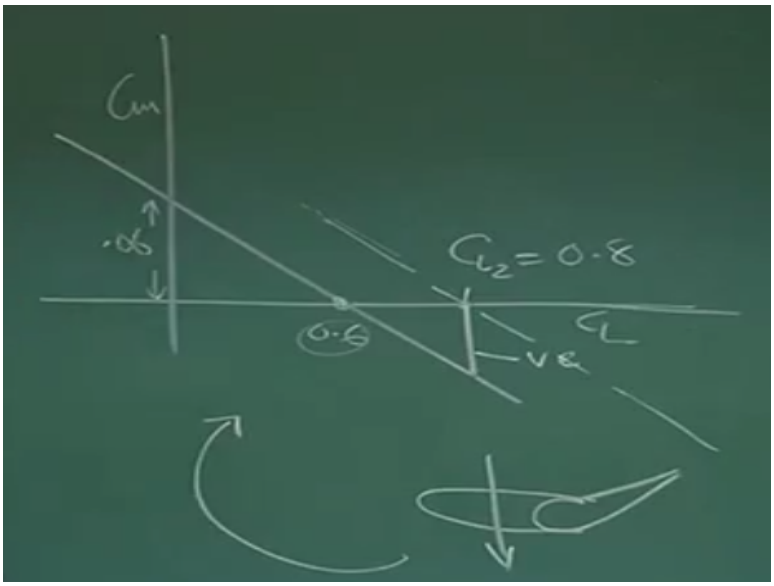

Let us see let this point is .6 let this point is 0.06. We are flying this cl now there could be recall that a now fly in the same altitude at high cl, this c l 2. Let's say the equation as .8. So what will happen the moment you try to fly out the air plane is .8 the air craft will generate the negative pitching mode right? But you want to trim this air plane at this c l without changing the stability all cg remains same. If you want to that mean's this negative pitch movement should be nullified. So who will nullify this?

We will nullify putting the elevator up. Elevator up means it gives force like this moment about cg nose up able to give elevator which will neutralize this negative pitching movement at cl =0.8 and how do I am do that we allow use this relationship delta e =delta e not + d delta e/d cl trim into cl trim.

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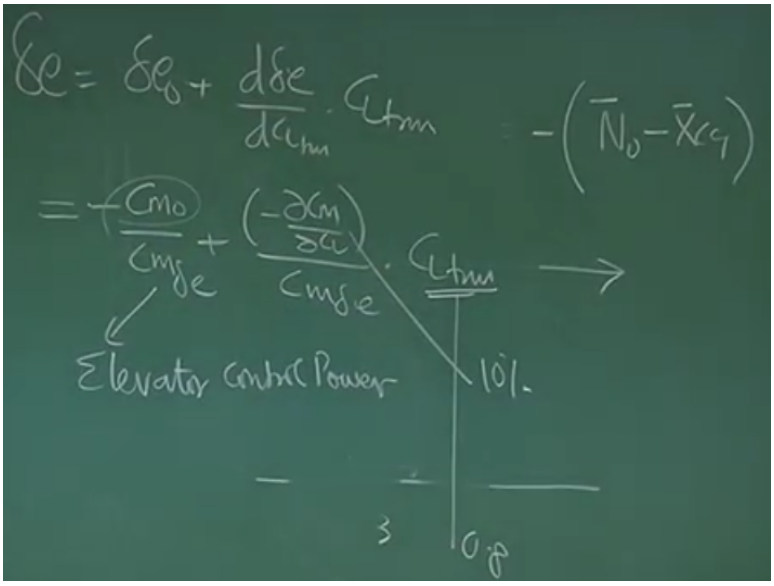

Where cl trim=- cmo/cm delta e+ (-delta cm/ delta cl)/cm delta e.cl trim. And you could see easily for the aircraft the cm not will be available c m delta elevator control power. Control power otherwise you find out the formula cm delta = -cl neeta t c l alpha tail and tau extra. Which you can check your notes d c m/dcl as 10% given here and cl trim is now it becomes point 8.

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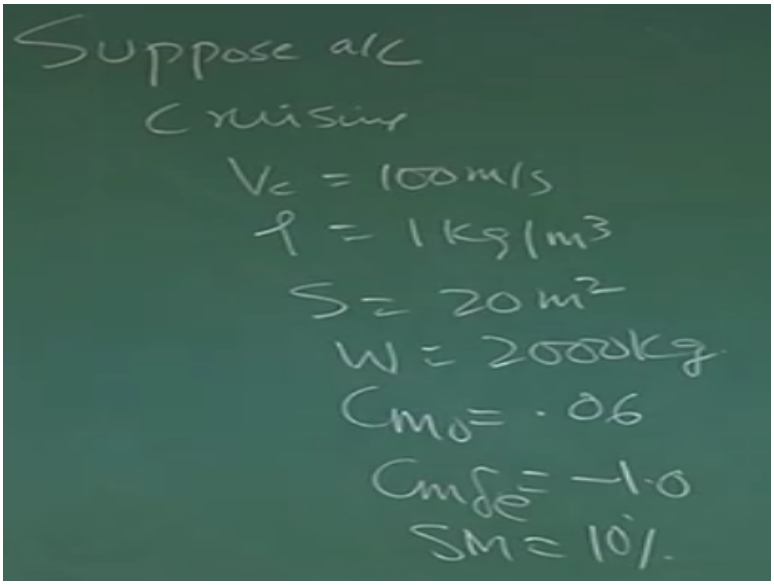

So simple define from the other problem also as discuses suppose aircraft cruising. V cruise is 100 meter per second rho is =1 kg/m cube that is the altitude where the density of mode 1 kg per meter cube. s = 20 m, w= 2000kg, cm not=0.06, cm delta e=-1.0, static margin=10%. What is the

elevator required if this question is asked us what is the elevator required. How can I do that in I come back to that.

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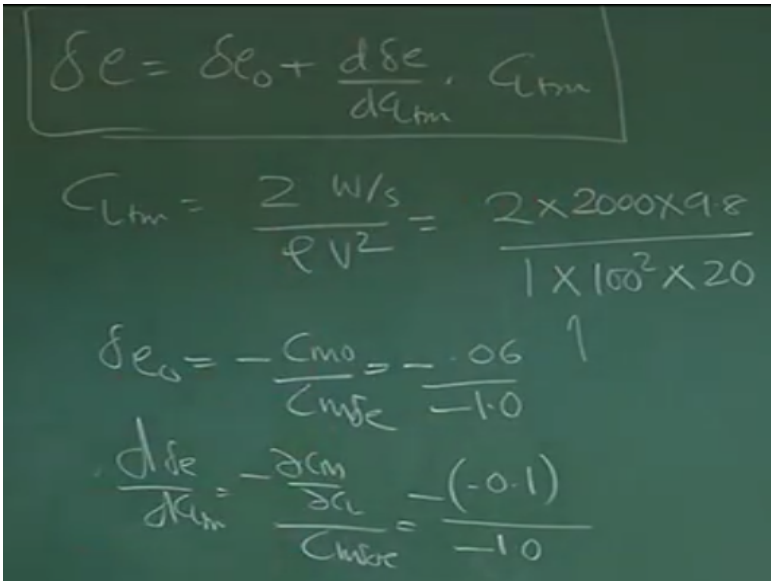

Relationship delta e =delta e not + d delta e/ d cl trim , cl trim what is the cl trim = 2 w/s/rho v square = 2* 2000*98 2 w by rho is 1*100 square * 20. So that will give the c l trim. Delta e not find out the = - cm not/ cm delta e the values are given = - 0.06/-1.0 then the d delta e /dcl trim = - delta cm/ delta cl/ cm delta e again you know that -(0.1)/-1.0. So we get d delta e plus d cl trim. And cl trim we found from here plugging here and get the answer. That is as simple as that ok.

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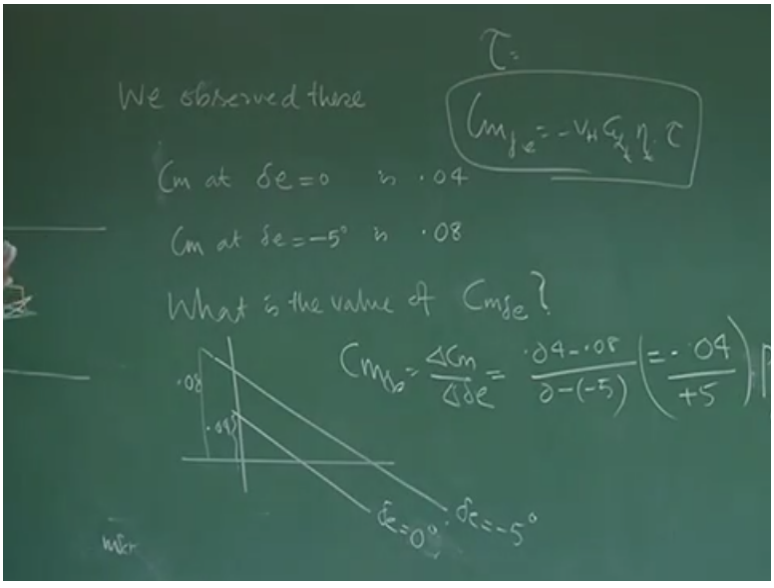

Let us take another problem. Suppose we conduct the wind tunnel test and observe this. In wind tunnel test we found that cm at delta e = 0 is 0.04 cm at delta e = -5 degree is 0.08. So we are contacting a wind tunnel test here put modal this is the elevator here. One test will content the delta is 0 measure the cm string x balance and it is found .04 and second case we will put the elevator let this up – 5 degree and then measure that came out to be .08.

What is the value of cm delta e 1 so the d delta e/ d delta l trim the dcl trim is nothing but de2-delta e 1/delta e1/cl2-cl1 these are the trim value. This is important ok and what is sign of d delta e/ d cl trim you should know sign of d delta e/ d cl trim = - dcm/ dcl / cl delta e and statically stabile at the this sign is negative. This sign is negative sign of d delta e /dcm trim is also negative ok and unit wise we see that this is demarcate is in there per degree then the d delta e/dcl trim degree per radiant d delta e/ d cl trim is a radiant generate will given when degrees ok.

How will find out but the potion what will be happen here this is one delta e one to delta e = 0 degree this value 0.04 is delta e is =- 5 degree from the value is 0.08 ok. What is volume of delta e. the cm delta e= delta cm/delta d e if we take this 0.04 – 0.08/ 0 – (-5) be careful at the sign and that will be give the value -.04/+5 is per radian. To give this much of value per degree not per radian because the angle is in degrees. There is should be very clear right.

So from tunnel test seeing this result immediately find out what is the cm delta e. Once I know is somebody asking Cm delta e what is the corresponding angle of tau? Remember tau is d alpha t by d delta e. rho c m delta e is expressed c l alpha t Neeta t into tau right. Please check the formula so you have to only use this formula here.

If you know the tail volume ratio if you the alpha tail you can manipulate the relationship to find out the value of tau given other change this is way you can go on calculating information based on the other information. We do a on the relationship on the ok right.

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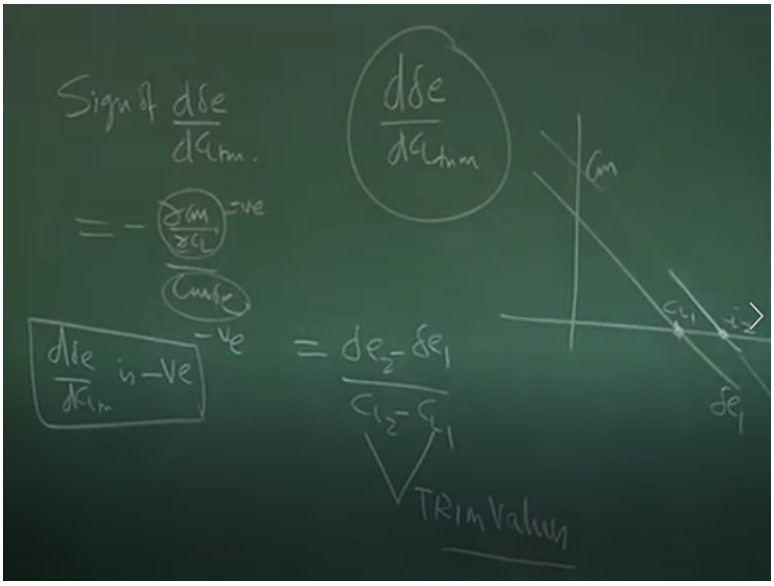

Another important thing we should be known typically d delta e/d cl trim what is the meaning of the cm vs. cl right from one cl under played at cl. This is cl 2 this is cl 1 and the delta e 1 and the delta e2 you want to fly cl2 and I need delta e2 cl is trim for flying at the cl trim 1 it was delta e 1. So d delta e by cl is nothing but delta e 2 minus delta e 1 by cl 2 minus cl 1. Where these are trim values this is important ok. And what is the sign of the delta e by d c l trim also you should know the sign of d delta e by d cl trim.

You know this is nothing but minus dcm by dcl by c m delta e. and for the statically stable airplane this sign is negative, this sign is negative. So total sign of d delta e by d c l trim is also negative. And unit vise you could see that it will be this is dimensional s, this is either per degree If it is per degree then d delta e by dcm is per degree.

With this understand I am sure to able to under the life problem including an example at us I should all the best will we also try to see that will given lectures of the evolution the shapes. How the shapes all is change at the white class as change and could understand what every you studied here how they translated into real product and today you will fly so many high air carft and the understanding this so simple ok. Thank you very much.