Aircraft stability and control
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Lecture- 04
Stability: Wing Contribution

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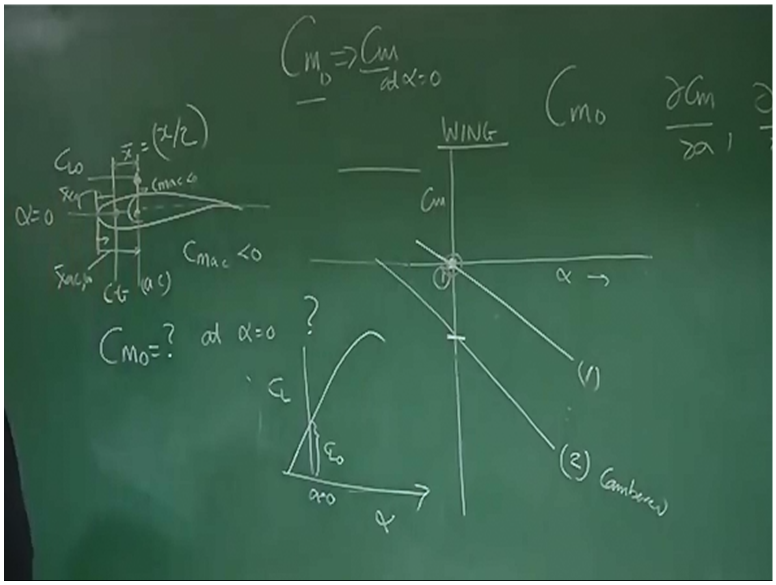

We have been talking about CM0 and DCM by D Alpha, or DCM by DCL. Let us cross check whether we are ready to generate or develop a field for this numbers through some examples, okay. Let's say, we have a symmetric aerofoil. I will take only wing as a medium to explain this and we want to draw CM versus Alpha. If it is symmetric wing and let's say AC is here, and CG is somewhere here, First case.

So how this CM vs Alpha will look like qualitatively we want to do that, this will tell us where they really have got the final points clear or not. when I try to do for wing literacy, similar thing can be extended for tail, if I am trying to draw variation of CM versus Alpha for a symmetric wing and I have to draw CM versus Alpha that means I need to know what is CM at Alpha = 0.

Because I know, I am looking for something CM = CM0 + CM Alpha into Alpha, okay? So it is very simple you go back to a school level, it is Y = C+ MX, where M is nothing but CM Alpha

X is nothing but Alpha C is nothing but CM0, Y=MX + C to be more precise its is a linear fit right.

Straight line. Now let us see what is C here, so C means what? CM0. CM0 means what? What is CM0? CM0 in this case is, CM at Alpha = 0. Correct, because of I am writing CM = CM0 +, CM Alpha into Alpha if I have written CM = CM0 + DCM by DCL into CL then I would have define CM0, CMX CL =0.We are not doing that, we are doing CM = CM0 + CM Alpha into Alpha, right. So what is CM 0? CM0 is CM, CM is what?

Pitching moment coefficient about which point about CG, right. At what condition at Alpha =0. So See Alpha = 0, because symmetric aerofoil will there be any CM about CG. No because they already no net force. So this will be this point to be here that is, at Alpha = 0, CM = 0, that means CM0 = 0, first point we have got, to draw Y = MX+C we need two information's, one is intercept that already have got C =0, second is the slope, that is DCM by D Alpha so CM Alpha will be what DCM by D Alpha since we are doing it qualitatively.

So we will just put the sign, that's all right we will not be bother about actual magnitude, but from here you could see that as I give some Alpha, because CG is ahead of C it will give nose down moment about CG right? So that will be pitching moment coefficient negative as per signature. So DCM by D Alpha will be negative so for this CM versus Alpha will look like this, clear.

For let's say case 1 is it clear. CM at Alpha = 0, Which is CM0 which is 0, now DCM by D Alpha that is the slope of this line and you know since AC is behind CG for any all positive Alpha it will give a nose down moment, nose down is negative, so the sign will be negative so the slope at equilibrium this is the equilibrium point, at this slope has to be negative. So I have drawn a negative slope here correct. Now let us see what happens if I keep everything same but I just convert this into a cambered aerofoil, okay.

So again, AC is here and CG is a head off, and let's say this distance is X bar, X bar is what about absolute distance none dimensionalize with mini aerodynamic chord. Again we want to plot

CM=CM0+ CM Alpha into Alpha, what will happen? Once I draw a cambered aerofoil, I know that at AC there will be CM, AC which is magnitude by that is less than 0. That is like a concentrated moment it is there that you know already good transferring of forces. Now, if I give some sort of an Alpha as a disturbance I see because AC is behind CG, so it will have stabilizing effect. It will give pitching moment negative for a positive Alpha.

So one thing I know DCM by D Alpha will be negative, so statically stable. Our next question comes what is CM0? CM0 we know, CM0 at Alpha =0, what happens that Alpha =0, for a cambered aerofoil. at Alpha = 0 let us see you go back to this diagram, CL and Alpha, remember this graph looks like this, and this is CL0 this is that Alpha =0.

So at Alpha=0 there is a positive lift, so that will happen somewhere here, okay? This is CL0 now what this CL0 will do. I am trying to find out what is CM at Alpha =0, CM0, so at Alpha =0 we realized there will be a CL0, which is will be acting at CMAC at AC we also know CMAC is less than zero which is typically -0.02 to -0.1.

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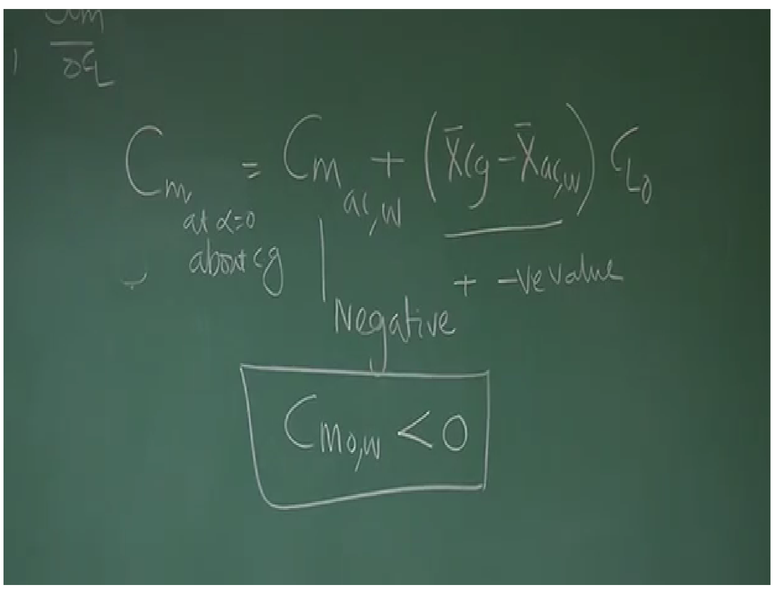

So now, we want to calculate what is the CM, at Alpha = 0 about CG, This will be CM, AC of the wing, which is a concentrated moment, and for a cambered aerofoil, for a cambered aerofoil it is negative so this + XCG bar - XAC wing bar into CL0, right. You know that at Alpha =0 there will be CL0. So what is happening for a cambered aerofoil. This man is negative, what about XCG – XAC.

Since XAC is behind CG so X if I measure it from here you see XCG is this distance XCG and then XAC is this distance XAC, So XCG - XAC of the wing will be negative right. Please understand CG is here AC is here so XCG this distance - this distance will be negative so this gentleman also will give a negative value. So what will happen CM0 of the wing will become negative, for a cambered aerofoil?

If AC of the wing is behind CG, although it will be statically stable, so what we learned? That slope will be negative, but its intercept will be also negative. So the intercept just to somewhere here which is negative, and the slope also will be negative, so I draw it like this, so this is for configuration two cambered. What is the message from this wing? It says, it is statically stable but you cannot this value is at negative angle of attack.

So message is you are not able to trim this airplane or trim this flying wing, at a positive angle of attack right. That is you cannot trim a cambered aerofoil at a positive angle of attack, correct. That is why, what is done you know we put a reflects aerofoil on reflects aerofoil you put this becomes positive or slightly positive correct. So this part is cleared, as well CM and Alpha variation, is extremely important. Right. now let us see for the tail.

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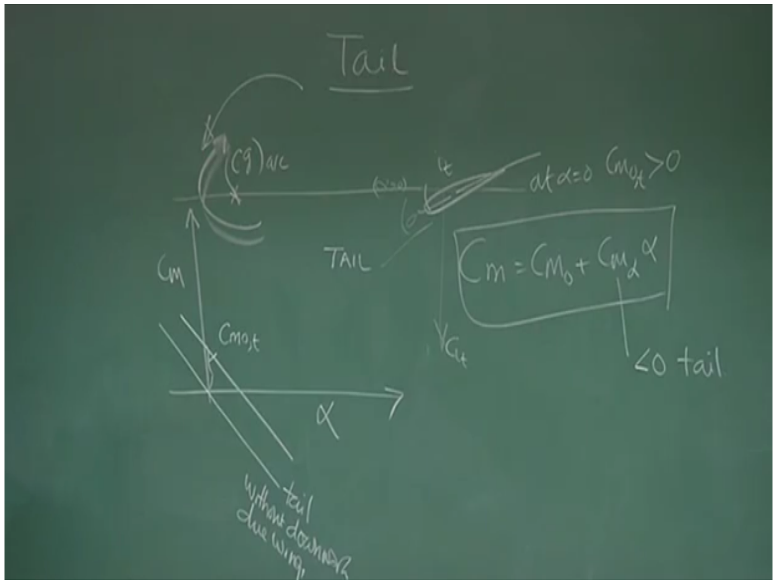

So in this course we have to go on doing all these sort of an example exercise, because just writing some expression will not make you a flight mechanics man, just writing for expression will not make you a designer, we have to go on doing integration to put numbers see it try to get feel out of that numbers.

Then all you could be a designer, then in my opinion then only you are fit for aerospace flight mechanics man, else you are as good as any good mathematician or anything for that matter, right. So I will be doing this in this lecture module you will find many times I will be searching from derivations to numerical, numerical to derivations your different interpretations at different level so that, you get the insight of all those expressions.

If I take a case of a tail, let’s see, tails are typically symmetric tails, unless it is the CG of the aircraft, right. so if I plot CM versus Alpha for tail, what will be the slope again before you start thinking you please create a map in your mind CM = CM0+CM Alpha into Alpha, and we are talking about tail now, we are also assuming that there is no influence of wing on the tail as per as downwash etcetera, etcetera is concerned which already you have seen but to develop physics of the situation to make the situation simpler.

I am neglecting it downwash and all, if the tail is after CG of the aircraft, what is going to be its contribution, we know, very well now, since AC of the tail, is behind CG, so naturally it will have a stabilizing component, fundamentally you could see if there is a Alpha disturbance this will give a lift, this will give a lift, and that will give a moment you nose down about CG so DCM by DCL because of tail is negative.

So I know slope is negative, so this part because of tail is less than 0 right. Now, what about CM0? If there is a Alpha=0 if it sees, I am neglecting all downwash effect of wing etcetera, etcetera, or fuselage in this isolated case if Alpha=0 if this is a symmetric tail, it will not produce any lift so it will not give any CM at Alpha=0. This is clear? So this again this will be tail contribution on the something like this. Tail without downwash due to wing.

So I have neglected that, you will understand why I am saying that, but I want, slope is negative is fine, stabilize but I want it should also gives CM0 positive because finally for the aircraft I want CM0 of the whole aircraft positive, and we have seen that once we use the cambered aerofoil where every possibility having larger CM0 wing negative, so you have to neutralize that so you have generate CM0 positive value through tail so how do I do that? The moment I, put some setting angle say I, T.

What will happen now? Let us again see CM Alpha is okay, AC is behind CG it is negative, but at Alpha = 0 what happens now? Now because of tail setting angle at Alpha = 0 there is a lift acting downward which will give movement upward, right. So now at Alpha = 0, CM0 is greater than zero because of tail, clear? At Alpha = 0, force will act downward, which will give a nose up movement, so that Alpha = 0, that is CM0 tail which is greater than 0.

So, the moment I put tail setting angle this will become something like this, where this is CM0 tail, that is why we give tail setting angle. So, what is the understanding? If I want to give positive CM0 generate positive CM0 one of the option is, you can put tail setting angle. Negative tail setting angle. We have been discussing about static stability of airplane, and we have understood one thing that, if there is a positive angle disturbance to an airplane.

About its trim suppose trim means about the airplane is flying at Alpha = 2 degree, and because of some gust there is a say point five degree disturbance, so the aircraft will be set to be in static stability if the aircraft automatically generates a nose down movement nose down pitching movement ensuring it has initial tendency, to come back to equilibrium that is two degrees, and then I say the aircraft is statically stable.

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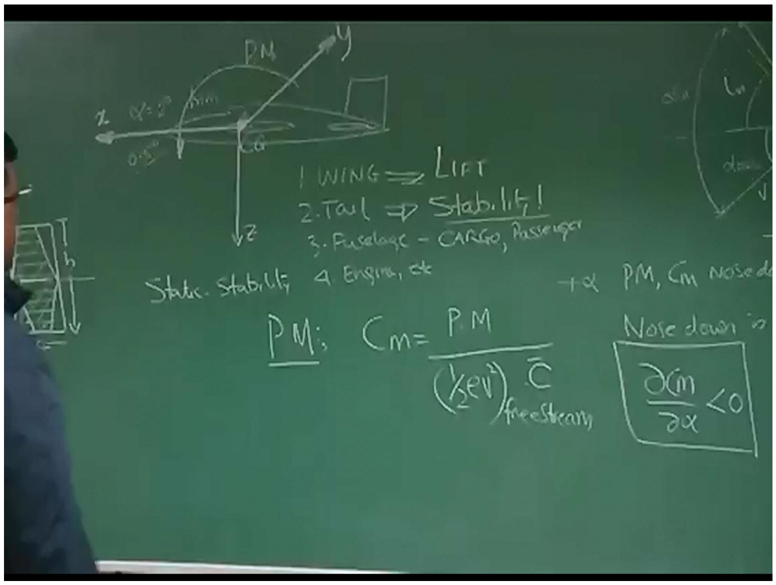

Just to illustrate this point, suppose this is an airplane, this is the horizontal tail, this is the wing, and let's say this airplane is flying, at Alpha = 2 degree, which is trim that means it is in equilibrium. Let's say cruise flight it is flying at Alpha 2 degree, so that lift equal to weight, thrust equal to drag as there is no net pitching movement also, now because of some disturbance let's say,

Some point five degree, disturbance is encountered by this airplane may be because of upward gust, then If the airplane automatically generates, nose down pitching movement, okay. then we say it has a initial tendency to nullify this point five degree, so that it has the tendency to come back to Alpha = 2 degree or we say it has a initial tendency to come back to the equilibrium, then we say this airplane is statically stable, okay.

Now the question is who will help in generating this nose down movement nose down pitching movement, so that it is statically stable, before you understand that, we should again recall that the airplane has three basis components, one is wing, another is tail, then third is fuselage, and then fourth engine and etcetera, right.

Landing here so many other things miscellaneous components could be there. If I try to see what the role of the wing, wing primary role is to generate enough lift so that it can do its mission but if it is equal to weight mission that is cruise mission, is the wing is primary responsible to generate that much of lift.

If it's the manure, then load factor to be generated then again the wing is responsible to generate that much of extra lift. What about tail? Tail we have seen and we will see also in more explicit manner, its primarily, it is there to provide stability. okay. First task what about fuselage, fuselage is basically for cargo, passenger, etc. and of course engines primary role you know that to deliver trust, okay.

So when I am talking about static stability I am being talking so many times about pitching movement, let us understand what is the sign convention to understand the positive pitching movement, negative pitching movement, so what we agreed upon is, as per sign conversion is concerned if I draw it here, if I put it as an X axis which is, located at CG, is Y and Z axis is like this body fixed axis.

So these XYZ body are fixed axis located at center of gravity, okay. And what is the pitching motion? Pitching motion is about Y axis, so if this is the airplane this is the Y axis pitching movement is about Y axis, and nose up by convention is positive and nose down by convention is negative, okay.

So, we are talking about pitching movements. We will not talk about any other movement now because the aircraft can rotate about Z axis or about X axis which we will discuss later at present we are talking about angular motion about Y axis that is pitching movement and you know we work in a non-dimensional quantity so we will define CM as pitching movements non-dimensionalize with half row V square to dynamic pressure free stream and of course you need the length storm.

So, if your C bar which is mean aerodynamic chord you will be knowing something about mean dynamic chord. But, I will just touch upon here and while solving a problem, we will understand how to compute it. If this is the span of the wing and this is the chord, here the chord is constant so its mean aerodynamic chord is also C in terms of magnitude in terms in terms of its location.

However, if it is something like this it is a trapezoidal wing like this then you see that chord is changing at every span location. So, there is a need to define mean aerodynamic chord at present you understand that something like an average chord and which is, which will be located somewhere here and it will be categorized by the length of the mean aerodynamic chord and also the location where from its starting so, with this simple background.

I am moving forward at present you think some sort of the chord some sort of its average chord and less dimensions to see that this is not dimensional so as far as static stability is concerned in pitch motion what we have realized that if there is a positive Alpha disturbance then it should generate pitching movement and hence, I call it CM nose down and as per the convention nose down is negative that's why you say.

Delta CM by Delta Alpha this slope should be less than zero for static stability we have been discussing that right. But now the question is who will be generating those forces and movement so that for a positive Alpha it will indeed give a negative pitching movement. So, who are the component wing, tail, and fuselage and engine so we will first start with what is the wing contribution towards generating

A nose Down movement for a positive angle of a attack or in a more general way we will now find out through formulation what is the contribution of wing towards static stability okay? So that is the first exercise we will be doing.

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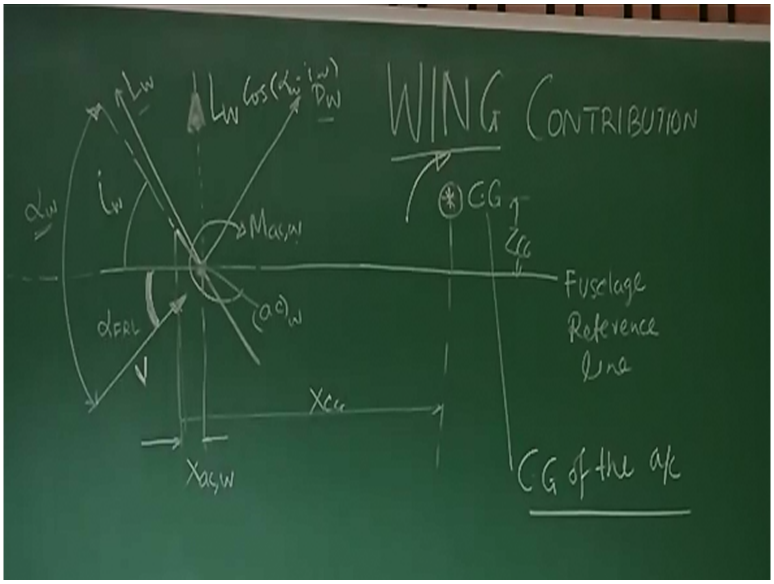

Let us draw our at10tion here what I have represented here is the wing through mean aerodynamic chord. This is the wing is the aerodynamic center of the wing this is the V the free stream velocity. When I say stream means? It is the velocity of relative velocity which is not disturbed by the presence of the aircraft right. So that is why we said theoretically at infinite what is the direction and magnitude of velocity or dynamic pressure in general.

So, this the velocity direction and let say this wing is set such that it has IW setting angle with respect to fuselage reference line so the wing if a normal aircraft mostly we will find the wing will be if this is the wing if it is most of the airplane we find it is set like this. So wing setting angle is zero if it is set like this then, I say it has a positive wing setting angle and which is denoted by IW. So, now tell me if this is the velocity vector and this is the mean aerodynamic chord, the angle of a attack seen by that wing or mean aerodynamic chord is Alpha W.

Which is angle of a attack seen by the wing and for our mathematical explanation, we are also denoting the angle between velocity vector and fuselage reference line as Alpha FRL is stand for fuselage reference line okay. And you could see in this wing I have put MAC wing and it is a general case if it is a cambered aerofoil MAC wing will be negative or CMAC will be negative with symmetric aerofoil the CMAC wing will be zero right.

And this is the general description of CG location here so I am measuring every dimensions of the length with respect to leading edge of the wing so I am denoting the CG location along X is XCG the location of aerodynamic center as XAC and ZCG is the vertical location of CG of the airplane with respect to fuselage reference line. Please understand this CG is CG of the aircraft, it's very important, right.

That means, it has tail, it has engine, it has cargo, it has everything, fuel finally that is the CG, CG of the airplane, don't get confuse it is not the CG of the wing it is the CG of the airplane. okay.

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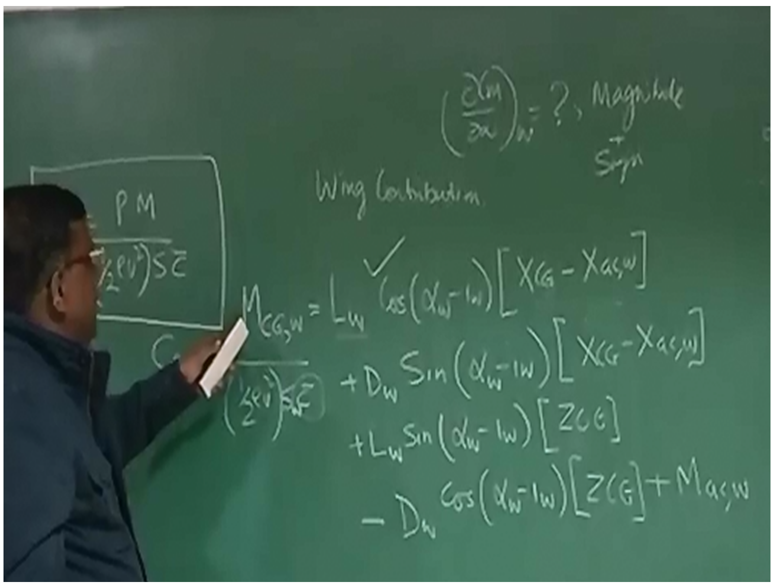

So, now, what is our aim, our aim is to see DCM by D Alpha because of wing is what in terms of magnitude in terms of sign positive or negative correct.Now, let us also understand very clearly in free air or in once the aircraft leaves the ground. It is suppose to rotate above the axis passing through center of gravity. That is why we will be writing all the moment contribution of the wing next you will see about tail, fuselage or engine all about center of gravity of the airplane.

So, let us first see since we are doing the wing contribution let us write this wing under this condition of V what are the forces it will see? It will see experience of drag force to DW drag of the wing, it will experience lift on the wing because cambered aerofoil there is a MAC of the wing and if I now try to find out the effect of this force in terms of, moment about CG what will be the expression? So, let me write the expression you can easily see this M moment about CG because of wing I can write it as let me write this LW COS Alpha W - IW into XCG- XAC wing.

And DW because of drag of wing SIN Alpha W - IW into XCG - XAC wing similarly because of SIN component of lift we have another term - DW into COS of Alpha W - IW to ZCG then of course there is term MAC wing let us see this term, let us see how these terms are here, if I see this if I take LW component along here, this will be LW COS of Alpha W - IW right you can see from this angle so this will give about CG what type of moment?

It will give forces like this to give a nose up movement so there is a nose up contribution okay? Similarly DW SIN you could see here DW is here take the SIN component and you will again get this term all these terms come straight forward, and I'll advice you draw the vector and find out these terms okay, what is our aim? Our aim is to find DCM by D Alpha.

Because of wing so this is M from M how do I come to CM see, CM is pitching movement coefficient, and CM is defined as pitching movement non-dimensionalize with free stream dynamic pressure, reference area and mean aerodynamic chord okay. So I have to go for expression for DCM by D Alpha so I need CM so what I will do I will divide left hand side and right hand side by half row V square SC bar.

When I understand S is the wing area sometime I may use S instead of SW but in general, please remember if I write S it is the reference area for aircraft in the wing area so if I divide it by this then I get for this CM similarly I have to divide every term here by half row V square S so what will happen? You see half row V square S comes here under this and the C bar.

I take it under this is it clear? if it is not clear let me do one term for you so that you understand what I am telling. Since I need to convert pitching movement M to CM I am dividing left hand side and right hand side by half row V square SC bar so I will be doing in the right hand side also.

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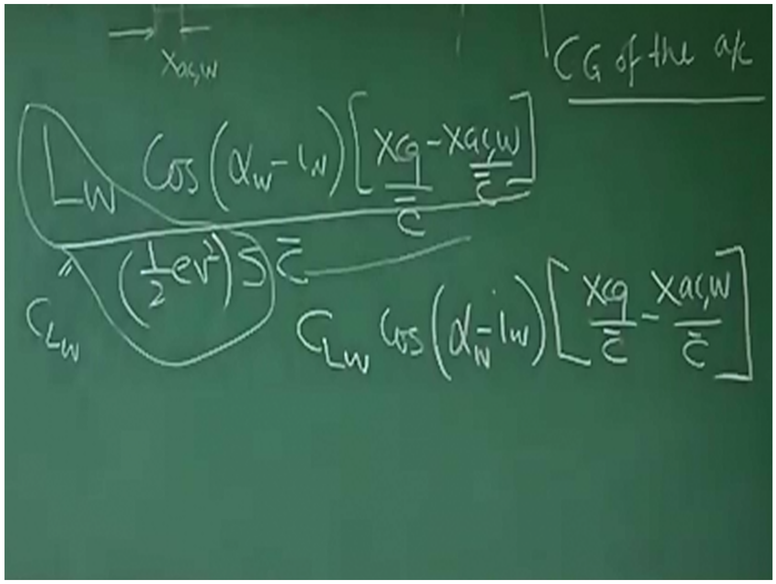

So I am taking the first term LW COS Alpha W - IW XCG - XAC wing. So, I will be dividing this by half row V square SC bar so what I do? LW by half row V square, for this I write this as CLW non-dimensional and the C bar I take it here so, I write it by C bar by C bar so what happens this term get reduced to CLW into COS of Alpha W - IW into XCG by C bar - XAC wing by C bar this is clear okay.

Similarly, I can now do for all the other terms so if I am smart which you are smarter then what I can do I need not write any other expression, I simply erase this I write here CL, I write here CM so CLW and here I write divide by C bar divide by C bar similarly here, I write + CDW and divided by C bar divided by C bar and here again I write + CLW and again divided by C bar and here I write CDW and divided by C bar, and this MAC wing becomes CMAC wing is this clear or not? Let me repeat.

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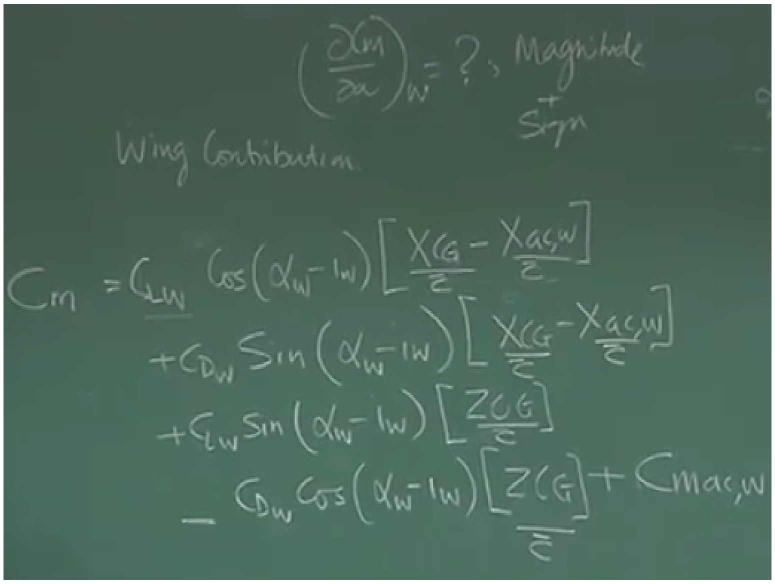

From moment, the momentum was LW COS of this into XCG - XAC wing since we are non dimensionalze into CM I am dividing by half of the V square S and C bar what I have done force divided by half of the V square S so this become non-dimensional, this becomes CL wing and C bar I took underneath the length term so, I get CLW COS of this into XCG by C - XAC wing by C accordingly.

I modify whole expression clear. And what is the CM? This CM because of wing about CG of the airplane this is wing contribution this is CM because of wing about CG of the whole airplane clear?

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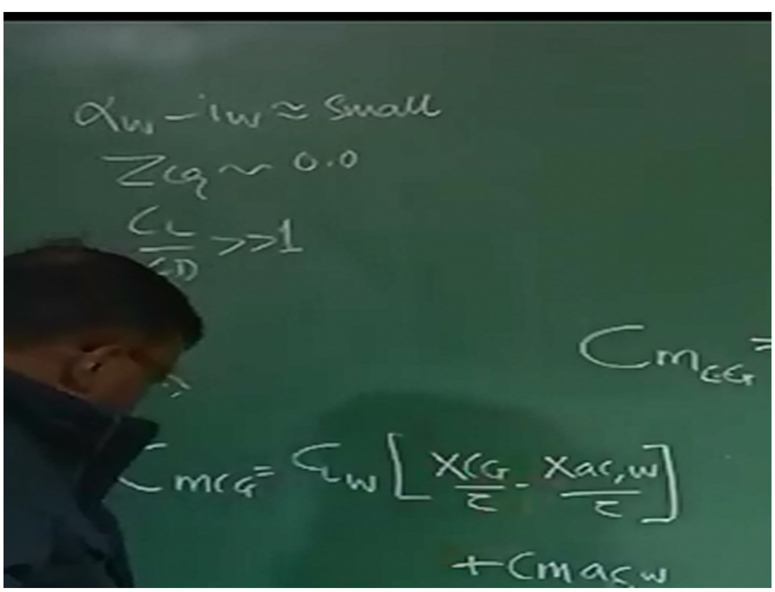

Okay, now we do approximation, that Alpha W – IW is very small right ZCG almost 0 that is very small and CL by CD is greater than one that is CL is much higher than CD okay, so the aircraft these are the approximation I am doing to get some feel for numbers right, in actual practice you need not do that you can actually compute what is the CM because of wing about the CG of the airplane.

If I do all these things then you find what is happened to the first term this becomes one because this is very small so CL, CM, CG will have only CLW into XCG by C bar - XAC wing, by C bar what about this term this term we will go because, we have neglected CDW, CD of the wing we have neglected right? What happens + SIN this number is also small. What about here although CLW is there you cannot neglect it but we have neglected ZCG.

We have said ZCG is 0 that means the center of gravity is on the fuselage reference line what about the third one again same you see ZCG we have taken 0, so this term was vanishes but this term remains so this is this + CMAC wing okay. Is this clear? I repeat again this is the general expression we have assumed that Alpha W - IW is very small so this is one, we have assumed that, CL of the wing is higher than much higher than CD of the wing in fact practically we're neglecting the drag contribution.

So this goes off here ZCG we have assumed to be zero that means the CGS on the fuselage reference line so this term goes see because ZCG is zero however this remains finally, I get a expression this.

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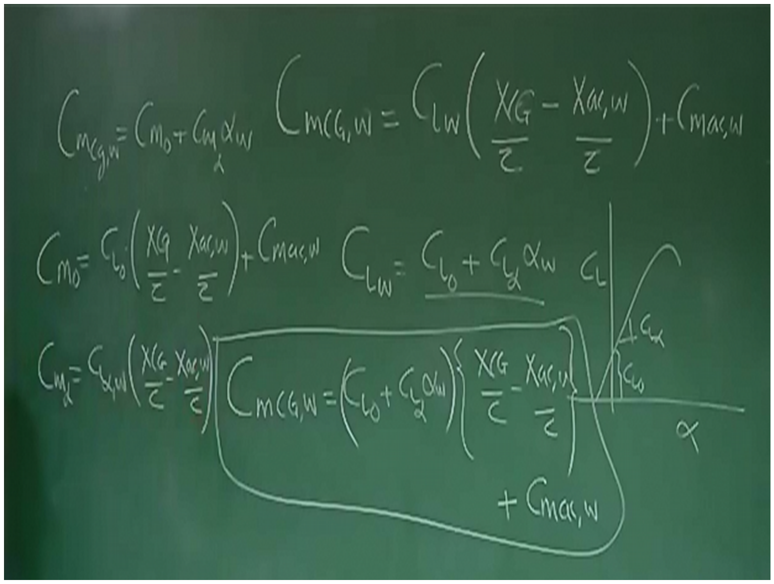

So let me write this in a neat fashion let us see CMCG wing is now with the simplification it becomes CL wing into XCG - by C bar of course, - XAC wing by C bar correct. This is there + CMAC wing correct now we that CLW can be written as CL0 + CL Alpha into Alpha wing which is obvious because you know.

We have assumed a general aerofoil for the wing so if it is a cambered then you know this part is CL0 the slope is CL Alpha this is CL, and this is Alpha so CL can be presented as CL0 + Alpha into Alpha wing. And if I substitute this expression, here then I get CMCG wing is = CL0 + CL Alpha into Alpha wing this is for this term then, this is XCG by C - XAC wing by C this is this + CMAC wing correct?

I repeat what is the expressions telling me this is the contribution of wing for pitching movement above center of gravity which is now after simplification, can be represented by this to expression where CMAC wing is the wing because of cambered aerofoil if it is a symmetrical aerofoil that value is zero, for a cambered it is minus right. Now with these two expression, I can write within this just closely see this it has two terms one is constant term that is CL0 into XCG by C - XAC wing by C + CMAC wing constant term another term.

Which is changing with Alpha W so I can write this as CMCG wing as CM0 + CM Alpha into Alpha wing where CM0 = CL0 into XCG by C - XAC wing by C + CMAC wing and CM Alpha as CL Alpha wing into XCG by C - XAC wing by C.

So if I write this two terms again here for clarity and try to see what is the interpretation and designer would like to have from this that is very important, as I have been telling you again and again these expressions need one time derivation after that as a designer you should know.

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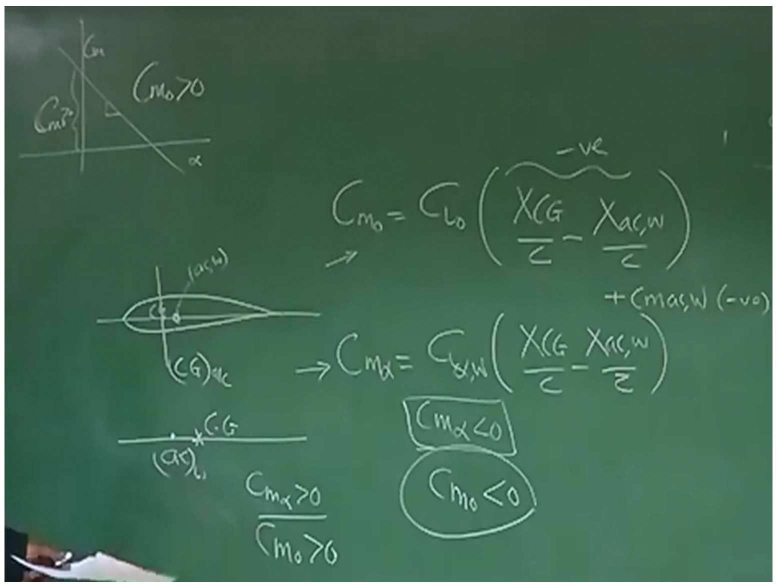

What is happening. And for CM Alpha, we have CL Alpha wing into XCG by C - XAC wing by C bar let us concentrate here what do you mean by that what is the message I am getting from these two expressions. Remember we have discussed that if I want to trim a machine at a positive angle of a attack I need to have CM0.

Greater than 0 right. this portion this is CM this is Alpha. So to get CM0 greater than 0 who will contribute to CM0 greater than 0, we are seeing that wing also will contribute also for the slope wing also will partially contribute okay now, if I see here in this case if this is my cambered aerofoil wing and let’s say this is the CG of the airplane it clearly tells me if the AC of the wing is behind CG if AC of the wing is here right then definitely CM Alpha will be less than zero see here if AC of the wing is behind CG then.

If I check CM Alpha AC is behind CG that means this is absolute while this is larger than this so this sign will be negative, so CM Alpha will be less than zero we will say the contribution of the wing towards stability stabilizing. You also know that if AC is behind CG it will be stabilizing CG of the airplane right however if I do that then what see in CM0 then this becomes negative, because XAC is behind CG isn't it so this part will give negative this is also negative for the cambered aerofoil. So what will happen this CM0 will become further negative.

That means whatever our initial aim was there to have a CM0 positive if you are making the wing giving stabilizing contribution towards aircraft static stability, then you should be sure that it is going to travel you in terms of CM0 it will give you a CM0 negative right and that is perhaps one of the reason you will find most of the cambered wing in airplane you will find that the AC of the wing is actually little ahead of CG of the airplane by doing that.

What has being done CM Alpha has become positive this stabilizing however CM0 has become greater than zero because, now XCG is greater than XAC so this becomes positive so CM0 we are getting positive however CM Alpha is also getting positive so, though wing is destabilizing in this case does not matter the tail will take care of stability part right you will see in detail when you solve some problem.