Fluid Mechanics & its applications
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Lecture 26

Welcome back.

In this lecture, we would cover the development of lift on airfoils and wings.

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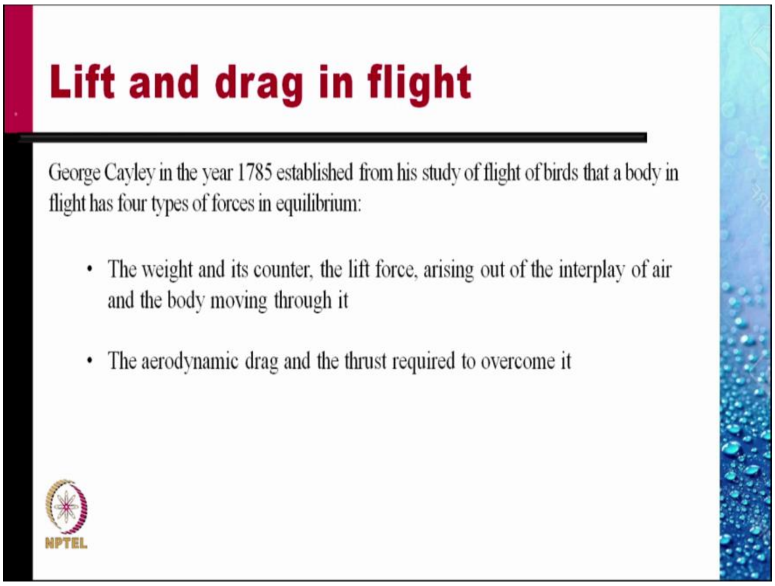

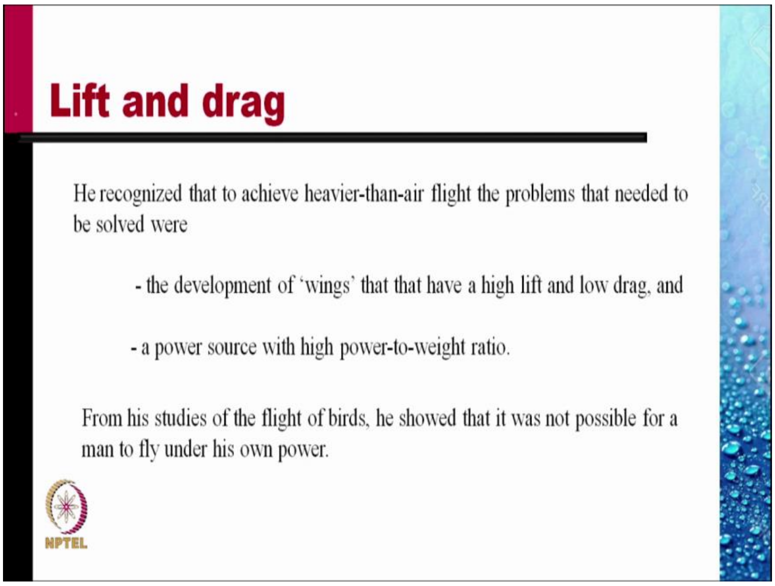

George Cayley in 1785 established from studies on the flight of birds that the body in flight has four types of forces in equilibrium, the weight and its counter, the lift force arising out of the interplay of air with the body moving through it, and the aerodynamic drag and the thrust required to overcome it. He recognized that for heavier than air flight, the problems that needed to be solved were the development of wings that have a high lift and low drag, and a power source with high power to weight ratio. From his studies on the flight of birds, he showed that it was not possible for a man to fly under his own power by attaching wings.

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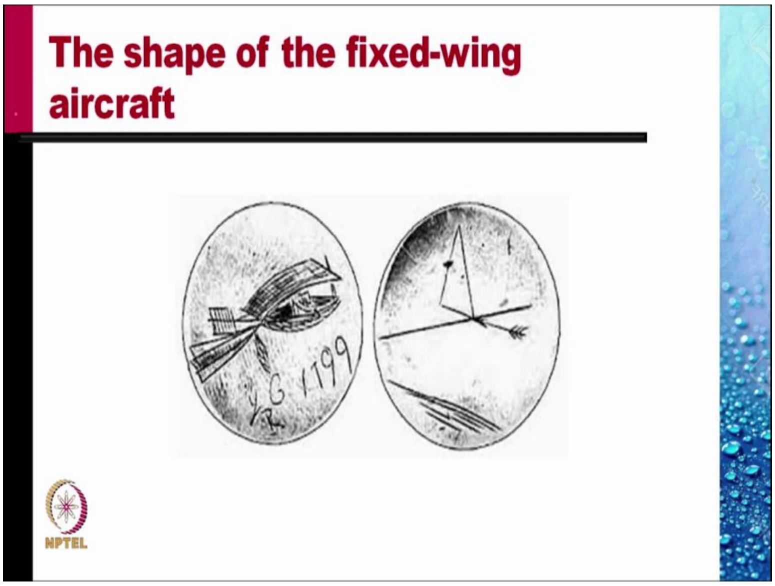

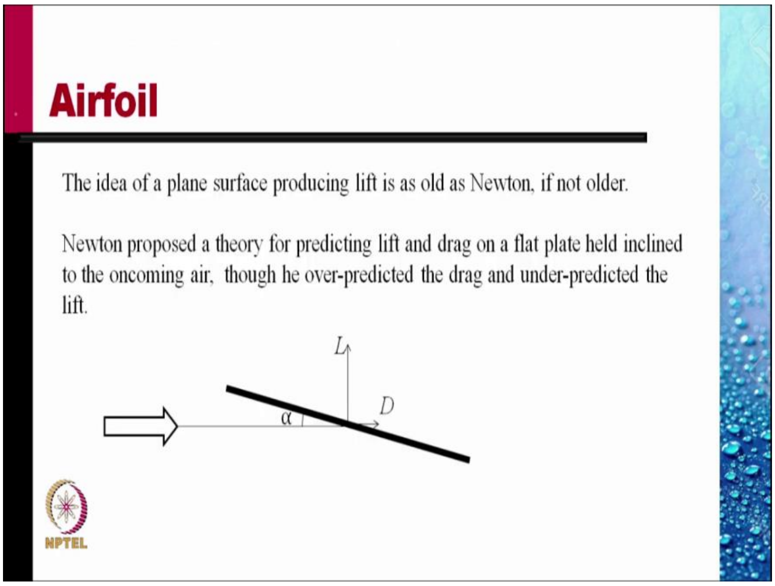

In fact, he was so sure of his thinking that he got a silver medallion etched in 1799 describing what he thought was an essential configuration of an aircraft. This was the first time when an aircraft was thought to have a tail and sail-like wings. The idea of a plane surface producing lift is as old as Newton, if not older. Newton proposed the theory for predicting lift and drag on a flat plate held inclined to the incoming air as shown. Though he over predicted the drag and under predicted the lift.

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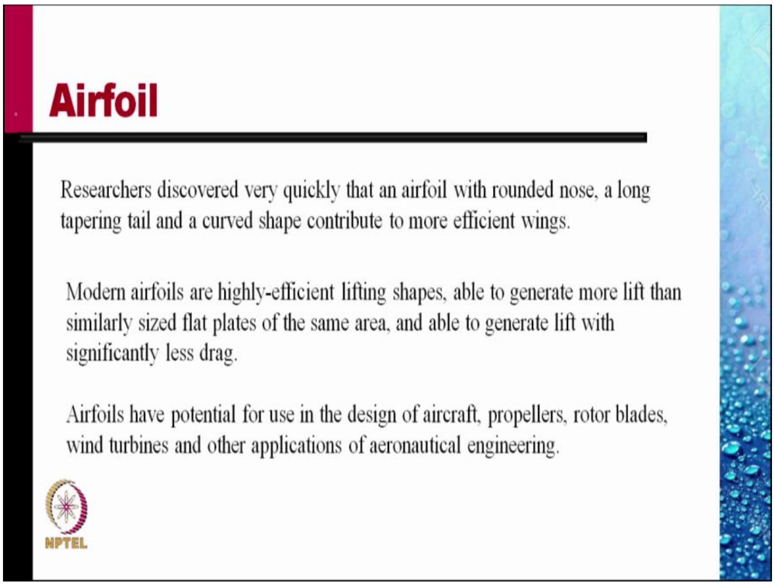

Researchers discovered very quickly that an airfoil with rounded nose, a long tapering tail, and a curved shape contributes to more efficient wings. Modern airfoils are highly efficient lifting surfaces, able to generate more lift than similarly-sized flat plates of the same area, and able to generate lift with significantly less drag. Airfoils have the potential for use in the design of aircrafts, propellers, rotor blades, wind turbines, hydro foils, and other applications of aeronautical engineering.

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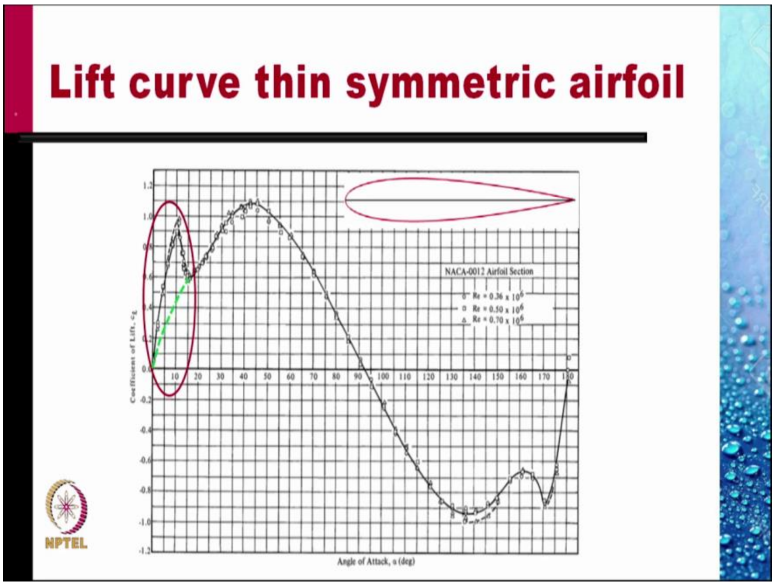

This graph shows the development of lift on a thin airfoil. The airfoil NACA 0012, which is a symmetric airfoil with 12 percent thickness at its maximum. And this is the whole curve for lift from for angles of incidence from 0 degrees to 360 degrees or rather 180 degrees, because it is symmetric after that. You notice that in the beginning, the lift rises very fast with the angle of attack, with the angle of inclination of the airfoil. This is a very important discovery.

In fact, Wright brothers are credited with this discovery. When they suspected that the data they were working with was over-simplified, and conducted their own experiments, where they found that the lift produced by airfoils rose sharper than α for a flat plate. The rest of the curve, except for this region, the rest of the curve behaves as if it is a flat plate.

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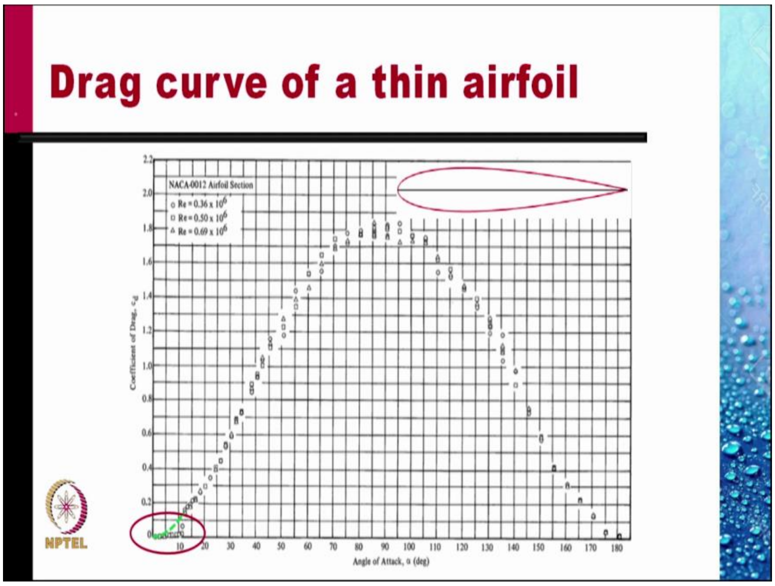

Not only this, but the drag curve also shows an anomaly at low angles of attack. Notice that the actual drag obtained is very much smaller than the drag that will be obtained on a flat plate.

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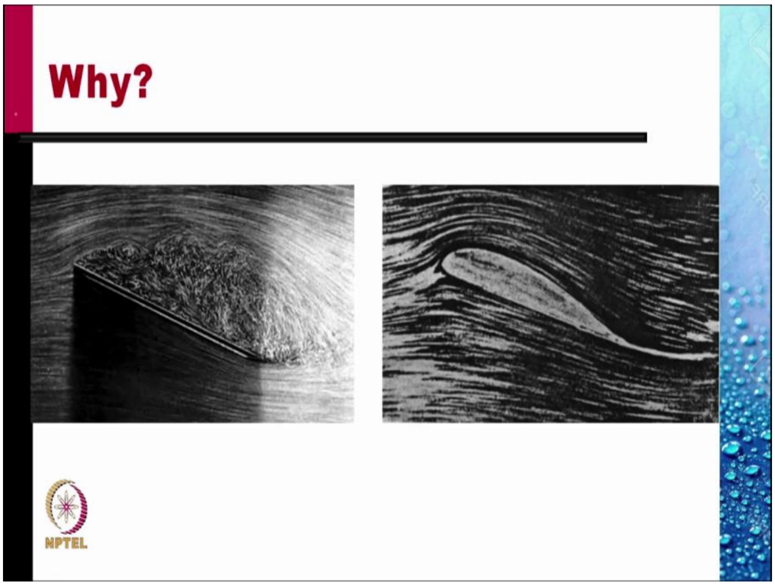

Why is this happening? This is happening because of the rounding of the nose. The first picture on the left shows a flat plate. The flow separates right the nose producing a thick wake. This thick wake results in comparatively large drag and not so high a lift. While the picture on the right is flow about an airfoil in which the nose is rounded. And we see that because of the rounding of nose, the flow does not separate on the top edge, it goes around the body surface. And because of this, it produces a negative pressure on the top which contributes to increasing lift, the kind of lift we saw in the graph earlier.

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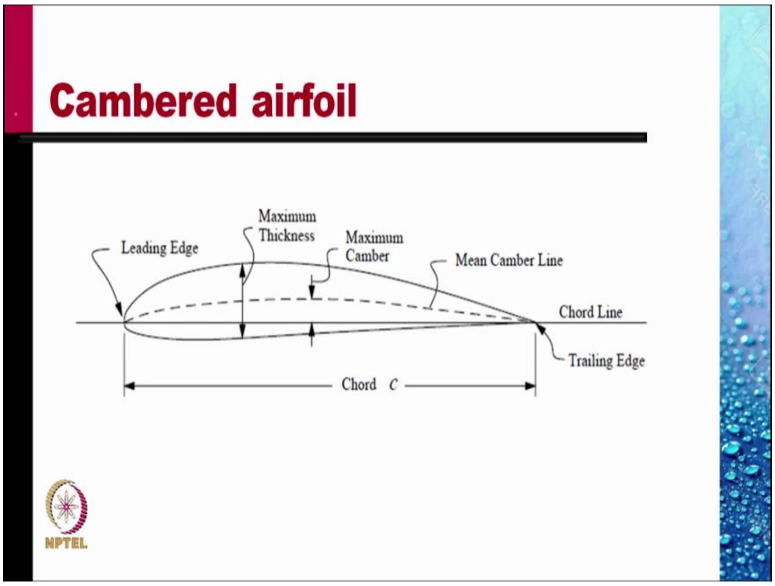

Most airfoils used in industry are cambered airfoil. Cambered airfoil means the airfoil which is not symmetric, but curved. There is a top surface of the airfoil and the bottom surface of the airfoil. The median line between the two surfaces shown here by a broken line is called the mean camber line. This is exactly midway between the top surface and the bottom surface. It is the curvature of mean camber line that results in lift even at no angle of attack.

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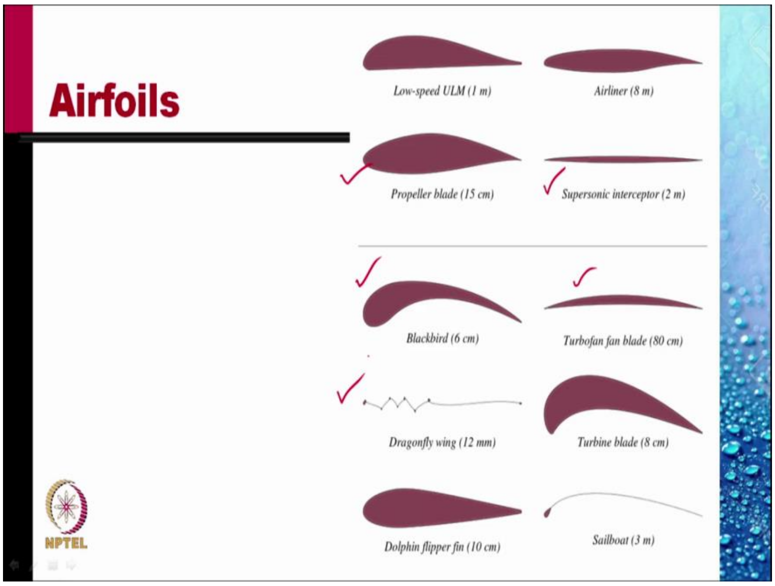

Large numbers of airfoils have been designed and are being continually designed. This picture shows a collection of airfoil shapes. On the top two, we see the low-speed airfoils. The third one is the airfoil for a propeller blade. Supersonic aircrafts have very thin sharp edged airfoils. A bird, the wings of the bird, are also airfoil-shaped. Turbofan blades are airfoil shaped, but with sharp leading edges. This is the cross section of the wing of a dragonfly. It is only 12 millimeter long. A sailboat with a sail inflated does also have a shape of an airfoil as it is shown in this picture.

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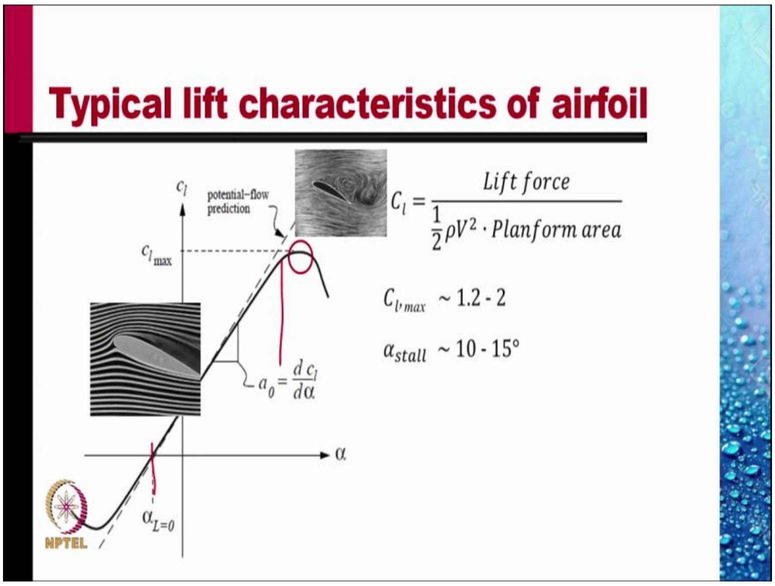

Typical lift characteristic of airfoil is shown by this graph. On the horizontal axis is alpha, the angle of attack. This is the angle at which the chord of the airfoil is inclined to the oncoming flow and the lift coefficient C_L, which is defined as the lift force per unit length of the airfoil divided by $\frac{1}{2}\rho V^2$, times the chord of the aircraft, non-dimensional lift. Over quite a bit of range, this is a straight line with almost a constant slope. This constant slope can be predicted by potential flow theory, and we will give a glimpse of this potential flow theory right here.

But when the angle alpha increases, after a certain value, the lift coefficient does not rise any further, and we get a $C_{L,max}$ value. Clearly in the linear portion of the graph, the flow is rather smooth over the nose with separation near the tail of the airfoil. But at the peak $C_{L,max}$, as you go beyond it, there is a large scale flow separation on the top surface which spoils the flow, kind of, and we get a large drag. This phenomenon is called stall in the airfoil, or the corresponding wing is said to be stalled. A pilot flying an aircraft has to be very careful about not reaching the stall. In fact, the aircraft operates largely within this range of alpha.

The typical value of $C_{L,max}$ that we obtain various between 1.2 and 2. There are ways to get a higher $C_{L,max}$, but that is obtained not with plain wings, but wings with articulating attachments, and the angle of attack at which such a wing stalls is typically between 10 to 15 degrees.

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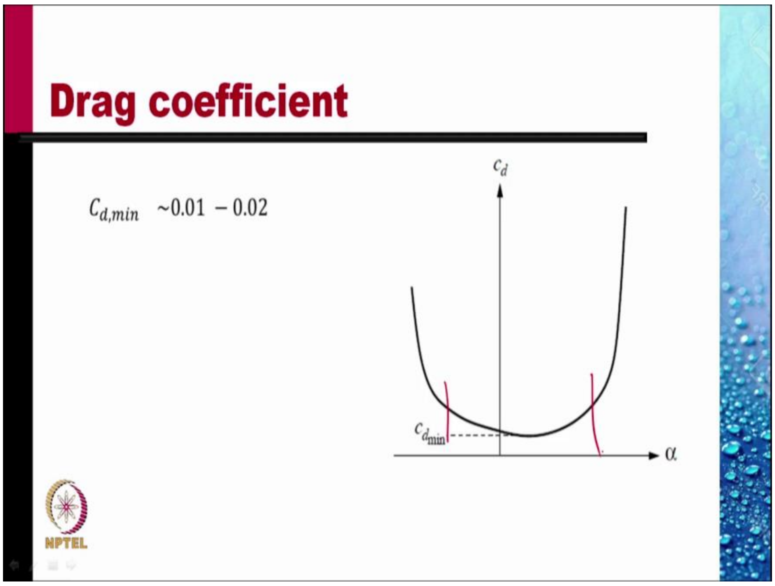

The drag coefficients are surprisingly small. The minimum value of drag is between 0.01 and 0.02, the drag coefficient which is the non-dimensional drag force, which is non dimensionalized by the stagnation pressure value $\frac{1}{2}\rho V^2$times the planform area. There is a large range of alpha in which C_D is small and beyond that, it rises suddenly. This is when the stall occurs and a large wake results, causing large drag.

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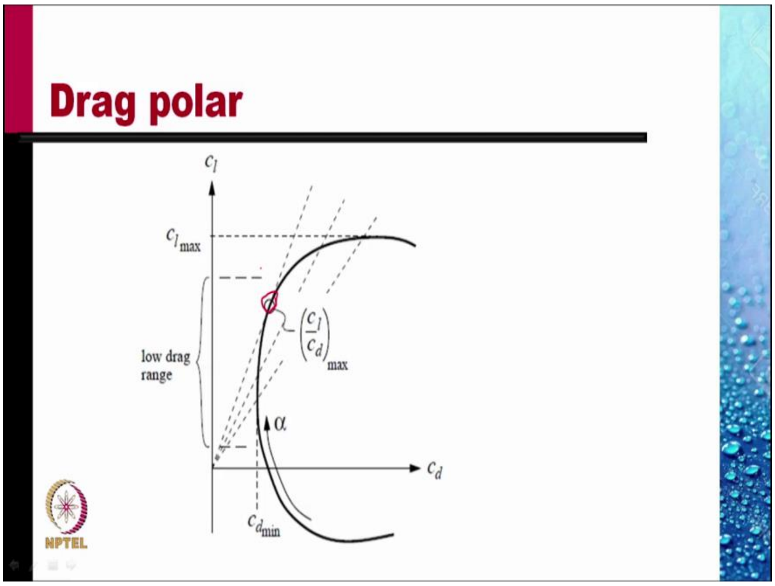

One of the interesting things to study is the drag polar. It is the polar plot of the lift coefficient and the drag coefficient, C_L versus C_D, and the curve shape is like this. Beyond $C_{L,max}$, the drag increases very fast. This point here represents the point of the maximum value of C_L/C_D. People who analyze the performance of aircrafts have determined that when the aircraft flies at an angle of attack, such that C_L/C_D is maximum, it can have a maximum range or maximum endurance depending upon what kind of engine does it have. A glider with no engine has the maximum range from a given height, when C_L/C_D is maximum.

In fact, C_L/C_D is termed as the aerodynamic efficiency of the airfoil, or of the wing. For competition gliders, C_L/C_D could have values between 70 to 100, that is, they slope downwards at less than 1 degrees. That is, if they start at an altitude of 1,000 meter, they would cover 70 to 100 kilometers in uniform gliding.

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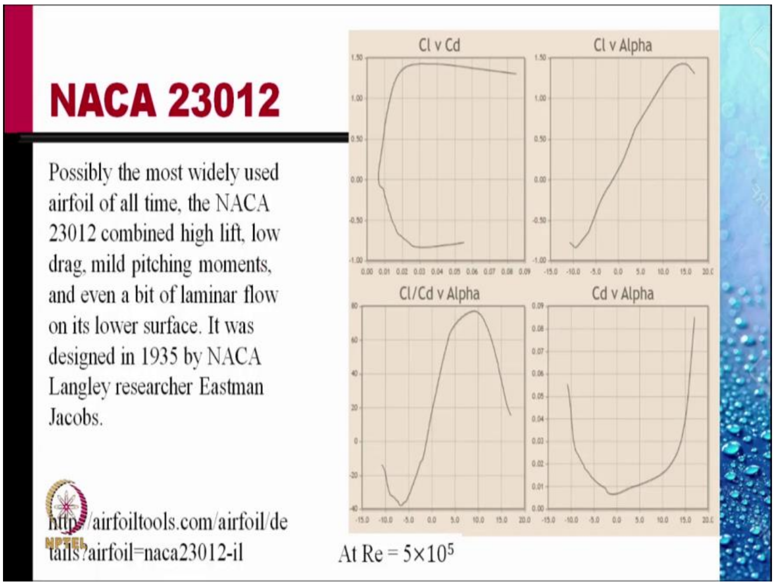

NACA, the predecessor of NASA, has codified the various airfoils. In this airfoil, NACA 23012, which is a 12 percent thick cambered airfoil was designed in 1935 at Langley Research Center. It combines high lift, low drag, very mild pitching moment and even a bit of laminar flow or its lower surface. The curves for NACA 23012 are shown here. It has a stall angle about 14 degrees, a maximum C_L of about 1.4, and a maximum C_L/C_D of about 75.

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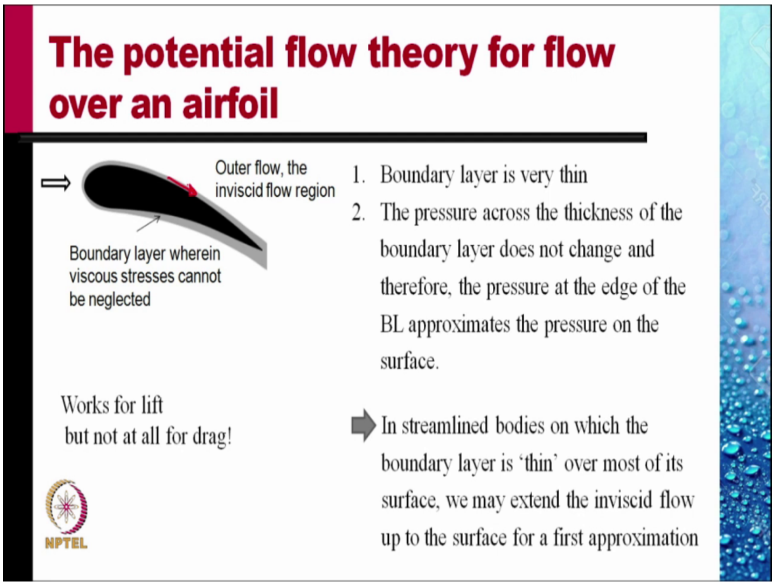

I had mentioned earlier that the potential flow theory can predict the lift of an airfoil. Why is that so? Consider the airfoil as shown on the left. Suppose the flow is not separated, the boundary layer is thin, and within this boundary layer, the viscous stresses cannot be neglected. Outside this boundary layer, in the region called outer flow region, the flow can be treated as inviscid. And this inviscid flow is potential flow with an inner boundary condition that the velocity is tangential at the edge of the boundary layer, right here.

But the boundary layer is very thin, and as can be shown in the development of boundary layers, that the pressure across the thickness of the boundary layer does not change. And therefore, pressure at the edge of boundary layer is the same as the pressure on the surface of the airfoil. So, to predict the pressure on the surface of the airfoil, we need to calculate the outer flow up to the edge of the boundary layer. And whatever is the pressure profile obtained at the edge of boundary layer can be impressed on the surface of the airfoil. But the boundary layer is very thin. And if it is thin, to the first order approximation we can neglect this at all, so that we can treat the outer flow, as a first approximation, to extend right up to the surface of the airfoil, provided the boundary layer is thin, that is, the flow does not separate. That is, for low angles of attack, or rounded nosed airfoil, and that is what makes the potential flow theory so powerful in aerodynamics.

We can use completely aerodynamic solutions, forget about the no slip condition at the surface of the airfoil, treat the surface of the airfoil simply as an impervious body, and calculate the pressures there. To summarize, in streamlined bodies in which the body layer is thin over most of its surface, where we extend inviscid flow up to the surface, as a first approximation. This works for lift, but does not work for drag at all. We cannot predict the drag. In fact, this predicts drag to be exactly 0, which is obviously not the case. We will have to bring in the viscous effects to predict the drag.

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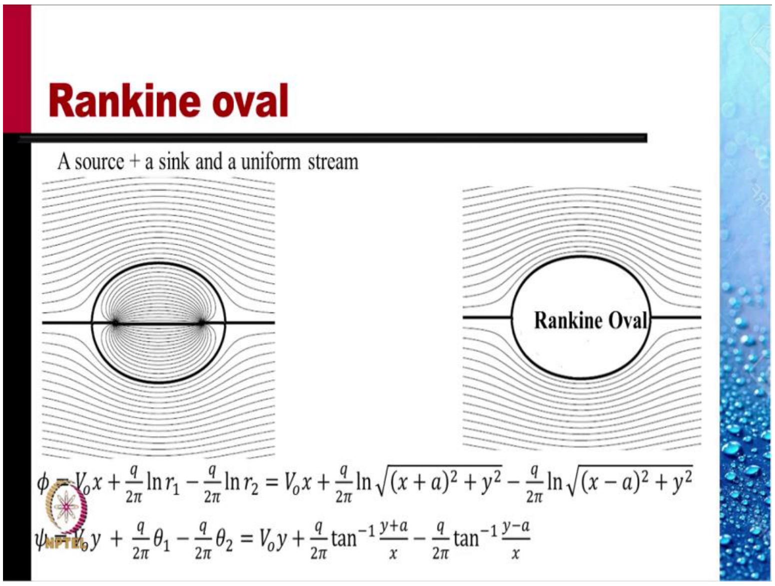

While discussing potential flows, we have done flow over a Rankine oval, which was obtained by a source and a sink, separated by distance, and immersed in a uniform stream. The pattern of stream lines was as shown here. And as was discussed, any stream line can be replaced by a solid surface. This closed stream line, which is shown as a dark oval, now can be replaced by a surface of a body termed as Rankine oval, and the potential due to a source and a sink separated by a distance, and superimposed by a uniform flow, then predicts the flow about the Rankine oval. The potential was written as the potential due to the free stream, due to the source, and due to the sink. Source and sink are situated as $x = -a$ and $x = +a$. Similarly, we could write the stream functions for this. We had done this in the last lecture.

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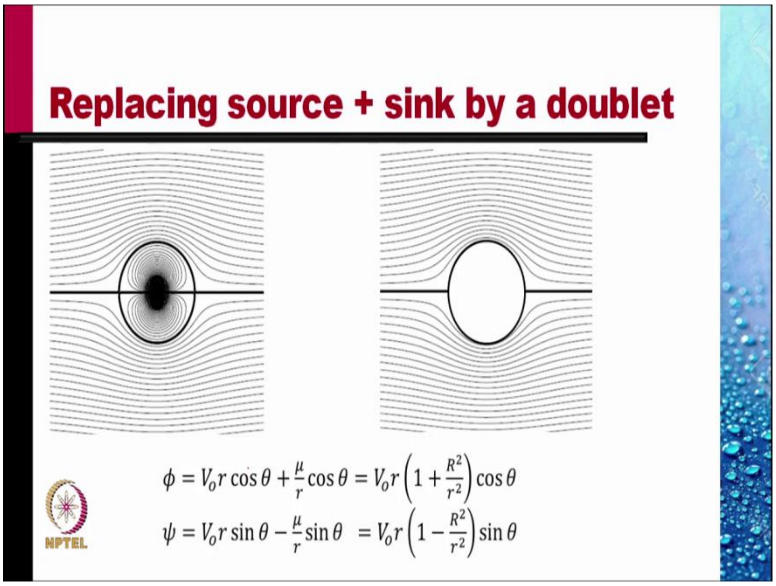

We had also done in the last lecture, that as the source and sink approach one another such that, as the distance a tends to 0, the source strength q tends to infinity, such that the product qa remains constant equal to μ. Then we get a doublet, and when a doublet is superposed with a uniform stream, we get a circular streamline. A circular streamline separates the flow inside that streamline with the flow outside. The inside area could be replaced by a cylindrical body. And so, this potential function and this stream function give us the flow about a circular cylinder.

We discussed this in the last class. We had also calculated the velocity distribution on the surface of the cylinder, and using on the surface of the cylinder this velocity profile, we had calculated the pressure distribution using the Bernoulli equation.

If the flow did not separate over this cylindrical surface, then that is exactly the pressure distribution that we will get, or that is approximately the pressure distribution that we will get. But the pressure does separate over a cylinder. It would not separate if the body was streamlined like an airfoil, and in that case, the predicted pressure distribution would be the same, or very nearly the same, as the actual pressure distribution.

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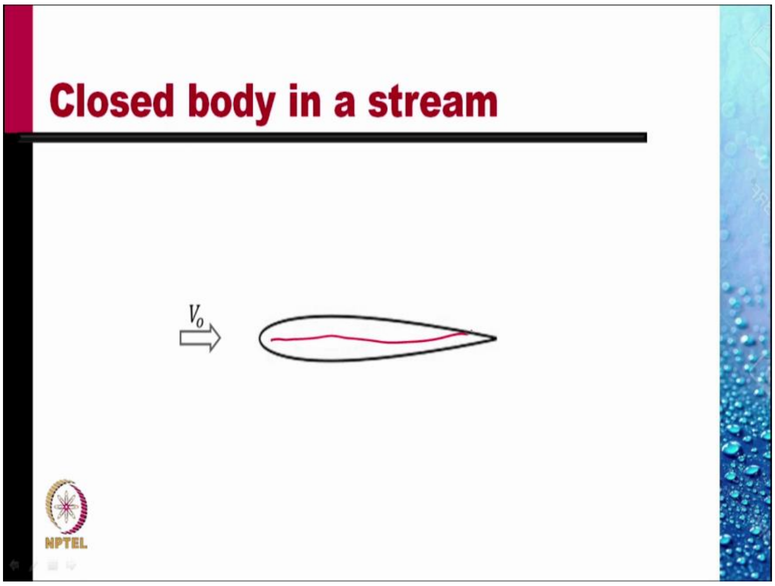

You notice that in both Rankin body and the circular cylinder, there was a source and a sink of equal strength enclosed within a closed stream line, an oval or a circle. In fact, if we distribute sources and sinks over a line such that the sum of the sources and sinks strengths is equal to 0 over the entire line, then there would be a resulting streamline which would be closed. The problem now is to predict this distribution of sources in sink such that we get a closed stream line which resemble the shape of an airfoil.

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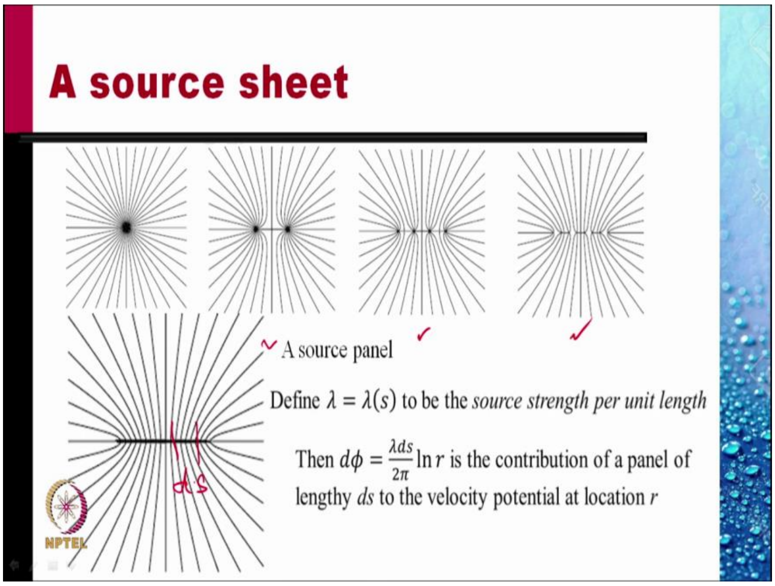

For this it is more convenient to work with a source sheet rather than individual sources and sinks. Consider a source of strength q, let us divide this into two parts, of q/2, and q/2. Then we divide this into four parts, and then we divide this into eight parts. And as we increase this division, what we have is a source sheet, a source panel. The advantage of this source panel is that it does not have a singularity like the singularity we had in a source, because the velocity there varied like ln r. So, r tends to 0, there was a singularity. On the source panel, there is no singularity of that kind. We define λ, lambda as a function of s to be the source strength per unit length over this source panel. Then $d\phi$, the contribution to velocity potential by a source panel of length ds is $\frac{\lambda ds}{2\pi}\ln r$. This is the velocity potential at location *r* due to this element of length *ds* of the source panel.

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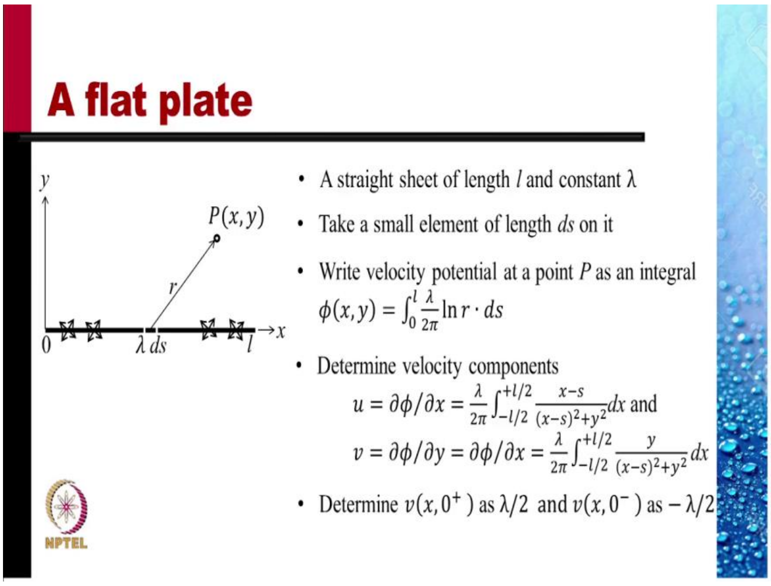

We solve a little problem. We assume a flat plate of length *l*. Let this be a source panel of a uniform source panel strength of λ. So, λ is no longer a function of x. Let us consider a small element *ds* at s, and then find out what is the potential at point P at the location x, y. We write the velocity potential at point P as an integral $\phi(x,y) = \int_0^l \frac{\lambda}{2\pi}\ln r \cdot ds$. $\frac{\lambda}{2\pi}\ln r\ ds$s is a contribution due to length *ds*.

So, integrate over s from 0 to l to get the total contribution to potential at that point. Then we determine the velocity components u and v at any point x and y, by differentiating this velocity potential. We obtain the integral with respect to x and with respect to y, and we get these expressions. Now, these integrals are not easy to evaluate, but most often we are not interested in evaluating them everywhere. We are interested in evaluating them close to the panel, a distance slightly above the panel, and slightly below the panel.

And if we do this, then the vertical component of velocity v is obtained as λ/2 on a point slightly above the plate, and −λ/2, slightly below the plate. So, there is a jump of velocity going from below the plate to above the plate. Across the plate, there is a jump of velocity equal to λ, the source panel strength.

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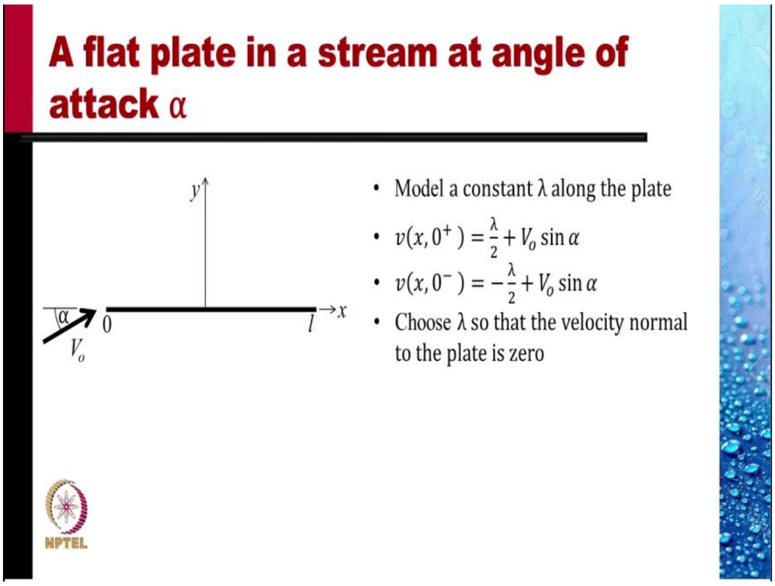

Now, once we found out the ϕ for a source panel, let us immerse this in a free stream, a uniform stream at the velocity V_o which is inclined at an angle α to this plate, so that this plate is now is at a positive angle of attack, α. So, we model a constant λ along the plate. So, $v(x, 0^+)$, a point slightly above the plate, is $\frac{\lambda}{2} + V_o \sin\alpha$, the contribution because of the panel plus $V_o \sin\alpha$ is the contribution to the vertical velocity above the panel because of the uniform stream. And *v* slightly below the plate would be $-\frac{\lambda}{2} + V_o \sin\alpha$.

We choose α so that the velocity normal to the plate is 0. Now obviously, this cannot be satisfied both at the top surface and at the bottom surface of this plate, because the normal velocities are different there. So, they cannot be. There is only one unknown λ. So, you can choose λ such that this condition is satisfied at both the points, a point below the plate and a point above the plate, without invoke physicality. The flow on the top of this plate would be separated flow. So, potential flow would not apply there.

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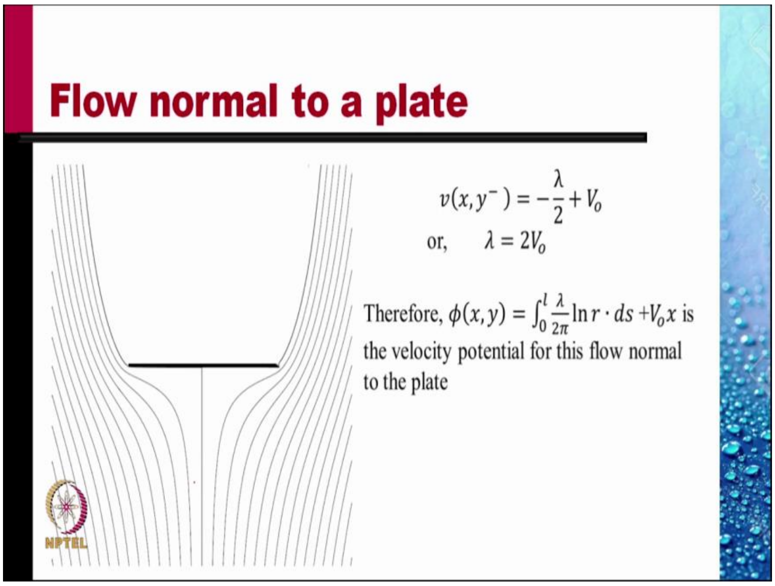

So, let this condition be applied to only the bottom of the plate. And when we apply this condition to the bottom of the plate, and we make the velocity zero there, then we can obtain the flow. This is the picture we obtain in which α is 90 degrees. The flow separates at the edges, but the rest of the flow region is very well predicted by the potential developed in the last couple of slides. We can, from that potential, determine the horizontal velocity, the only component of velocity, present in the bottom of the plate, and that would lead to calculation of pressure along this plate using the Bernoulli equation.

And so, we can calculate the net force on this plate, which is very close to what we actually get. This would require that λ is equal to $2V_o$. Therefore, $\phi(x, y) = \int_0^l \frac{\lambda}{2\pi} \ln r \cdot ds \ + V_o x$, where λ is equal to $2V_o$, is the velocity potential for this flow normal to the plate, to a finite length of plate.

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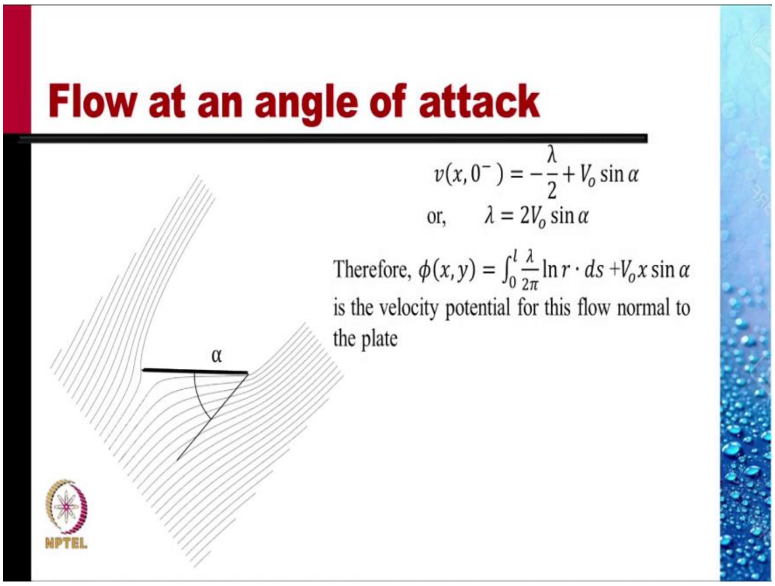

If the angle of attack was not 90 degrees, but α then the flow pattern obtained would be something like this. And the $\lambda = 2V_o \sin\alpha$ to give this flow pattern.