Aerodynamics

Newton's first law of motion states "The velocity of a body remains constant unless the body is acted upon by an external force." In other words, if there is no net force on a body the velocity will stay constant. Aeronautics demonstrates this principle fairly well, due to there being 4 main forces acting upon an aircraft while it is in flight. Those forces are:

Lift- Is the force that "raises" an aircraft up, produced by the airplane when it is moving through the air. It is created when there is lower pressure of air on top of the airfoil and a higher pressure on the bottom, causing the wing to be "pulled" upwards. There is a lower pressure of air on the top of the airfoil because the air takes longer to get to the same point.

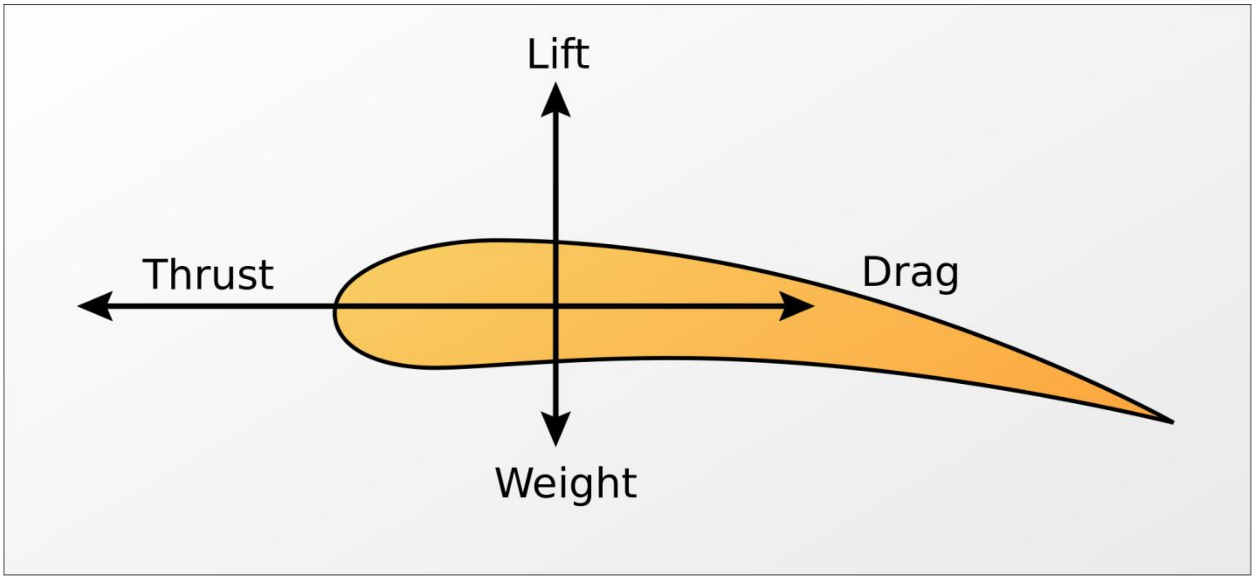
And lifts counterpart…

Weight- Is the force based upon the mass of the object and the force of gravity acting upon it.

Drag- Is the amount of air that is slowing down the aircraft in flight.

And drags counterpart…

Thrust- Generated by the engines on the aircraft, this propels the aircraft forward and overcomes the drag. The aircraft must maintain a net thrust force greater than the drag in order to stay at a constant velocity.



Above is a picture of the four forces acting on an airfoil. An airfoil is also known as a wing.

In order to calculate the lift of an object we need an equation for lift. That equation is

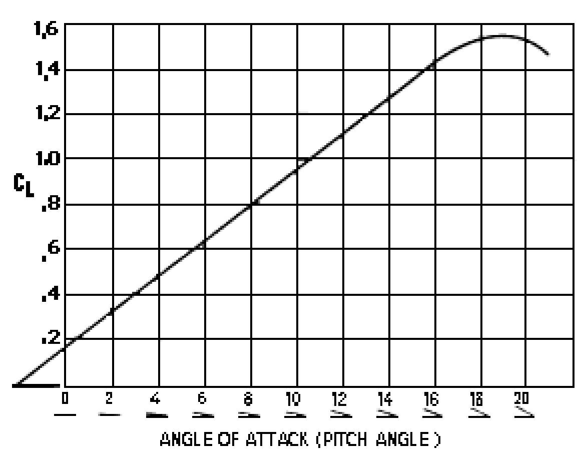
$$L = (1/2) \cdot p \cdot v^2 \cdot s \cdot CL$$

P=Density of the air. This is based upon the altitude of the aircraft. (Values can be found in I.C.A.O standard atmosphere table)

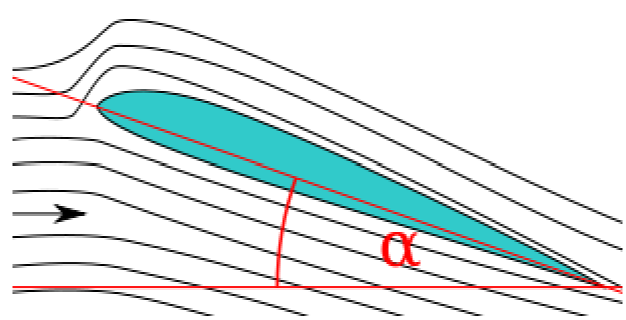
V=velocity of the aircraft in feet/second

S=The wing area of the aircraft in square feet.

CL=Coefficient of lift. This is determined by the airfoil and angle of attack. It’s usually very experimental and will be given.



The “angle of attack” is the angle between an airfoils reference line and the oncoming flow. The reference line is a horizontal line coming from the back end of the airfoil. The angle of attack is a part of the coefficient of lift and is factored into the equation. The chart above shows the optimal degree for the angle of attack, in order to get the highest coefficient of lift.



Since we now know how to calculate the lift of the airfoil we now need to find the amount of drag that the airfoil will have. The equation for calculating the drag is the same as the equation for lift, except the coefficient of drag is substituted for the coefficient of lift.

$$D = (1/2) \cdot p \cdot v^2 \cdot s \cdot CD$$

Cd=Coefficent of drag. This is determined by a number of factors (just like lift) and is usually found experimentally.

P=Density of the air.

V=The velocity of the aircraft/airfoil (and/or the wind coming at the airfoil)

S=The wing area of the aircraft in square feet.

Example

1) We have a light aircraft with a wing area of $\mathbf{250ft^2}$. The lift coefficient for this aircraft is **0.40**. Drag coefficient of **.05**, and a velocity of $\mathbf{150}\,{}^{ft}/_{s}$. How much lift and drag does this aircraft have? The air density is $\mathbf{2.38\cdot10^{-3}}\ {}^{slugs}/_{ft^3}$. After finding the lift and the drag, how much must the aircraft weigh and how much thrust must the aircraft have to maintain a constant velocity?

   In order to calculate the lift we must plug in this information into our lift equation.
   $L = (1/2)\cdot p\cdot v^2\cdot s\cdot CL$, so

   $$½\cdot(\mathbf{2.38\cdot10^{-3}})\cdot(150)^2\cdot(\mathbf{250})\cdot(.\mathbf{4}) = \mathbf{2677.5}\,{}^{\boldsymbol{ft}}/_{\boldsymbol{lbs}}$$

   We can calculate the drag the same way.
   $D = \left(\frac{1}{2}\right)\cdot p\cdot v^2\cdot s\cdot CD$, so

   $$½\cdot(\mathbf{2.38\cdot10^{-3}})\cdot(150)^2\cdot(\mathbf{250})\cdot(.\mathbf{05}) = \mathbf{334.6875}\,{}^{\boldsymbol{ft}}/_{\boldsymbol{lbs}}$$

   In order to maintain a constant velocity the aircraft must weigh exactly 2677.5 lbs to level off with lift, and it must be outputting more than 334.7 lbs of force in order to maintain a constant velocity.

Problems

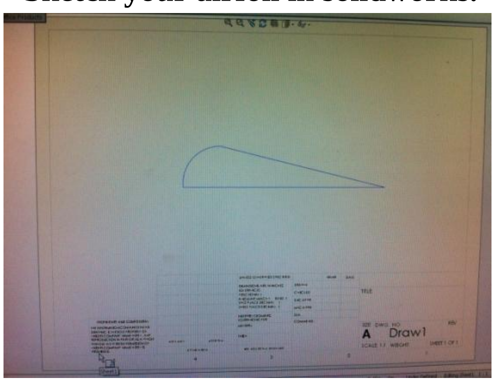
1) A Boeing 747 with a wing area of 5,500 $ft^2$is cruising at 813${}^{ft}/_{s}$. The air density is .000737 ${}^{slugs}/_{ft^3}$at 35,000 feet. The coefficient of lift is .3. The coefficient of drag is .08. Find the lift and drag the aircraft has.

Project:

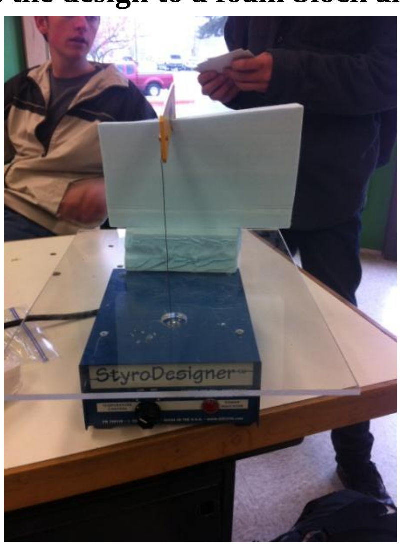
You will be using a foam block to construct an airfoil that will create the most lift and drag. The airfoil will be tested in the Kelvin Air tunnel, and the foam will be cut with the Kelvin foam cutter. The design of the airfoil will begin in solidworks, where you will get all of your measurements accurate. Then you can print it out, glue it to the foam block and cut it out from there. In designing your airfoil you want to get the most lift possible with minimal drag. You should take into consideration the angle of attack and how aggressive of an arc you want on your airfoil.

The blocks of foam are approximately 7” by 2 ½” by 2 ½”. Your airfoil should be no longer than 5 inches and no wider than 2 ½. From the bottom of your airfoil to its highest point should be about an inch and a half.

1. **Sketch your airfoil in solidworks.**

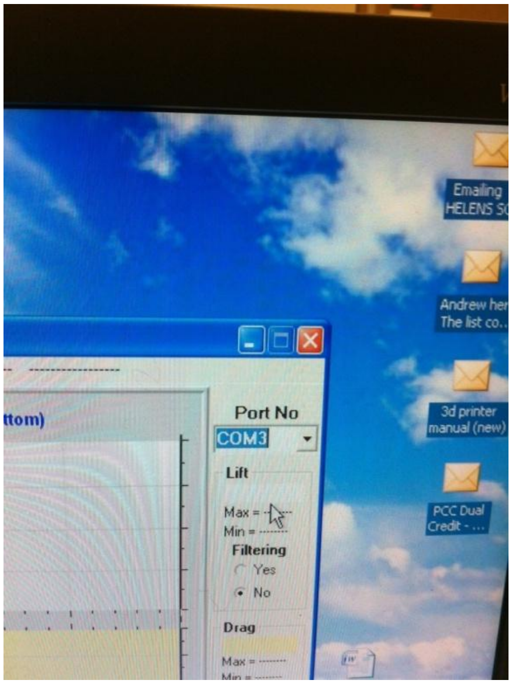


2. **Glue the design to a foam block and cut it out with the foam cutter.**



3. Cut a hole for the screw to be able to fit through (Make sure to make the angle of the hole the desired angle of attack)

4. Open the Kelvin "KelSensor" application and set the port to "COM 3." Hit "Calibrate Sensors" and press start.



5. Flip the switch on the Kelvin wind tunnel and let it run for a few minutes. After the data has been collected, stop the wind tunnel and write down the highest amount of lift and the highest amount of drag your airfoil got.

Finding the coefficient of lift and drag for your airfoil

Since we now know the lift and drag of our airfoil we can calculate the coefficient of lift using the same equation as earlier.

$L = (1/2) \cdot p \cdot v^2 \cdot s \cdot CL$

The air density is .00238 slugs/ft^3 at sea level. We will use that for the sake of simplicity. Our velocity is 45 MPH (The speed of the wind tunnel), or 66 ft/s. To calculate the surface area of your airfoil you can wrap a piece of paper around the curve of your airfoil, mark the edges, and then unfold it to create a rectangle. Do the same for the underside, add the lengths together and multiply by the width to get your area.

Now that we know all of our variables we can plug them in. On my airfoil I had a lift and drag of .14 and .09. My surface area was .193ft^2. So,

$.14 = 1/2(66)^2 \cdot (.00238) \cdot (.193) \cdot CL.$

After solving that I got CL=.0115.

We can now do the same for drag using $D = (1/2) \cdot p \cdot v^2 \cdot s \cdot CD$

$.09 = 1/2(66)^2 \cdot (.00238) \cdot (.193) \cdot CD.$

Therefore CD=.015.

Now what?

Since we now have our lift and drag coefficients for our airfoil we can discover the amount of lift and drag that our aircraft would have at any altitude and at any speed. Because our surface area is so tiny we are going to have to scale it up in order to get an accurate answer. In order to do this take your airfoils surface area and multiply it by 1250. This will amplify the size of your airfoil which will give us an accurate answer. Input the different velocity and air density values and calculate your lift and drag. Feel free to try your airfoil at any velocity/altitude.

For my example my surface area was initially .193 ft^2. Therefore, (.193)(1250)=241.25ft^2.

Test your airfoil at 300ft/s and at 35000 feet. The air density is 000737 $^{slugs}/_{ft^3}$ at 35,000 feet

$$L = \left(\frac{1}{2}\right)(300)^2(.000737)(241.25)(.0115)$$

$$L = 92.01\ ft/lbs$$

Now because we have two wings on our aircraft we will multiply that number by two.

$$L = 184.025\ ft/lbs$$

Test your airfoil at 500ft/s and at 50000 feet. The air density is .000364$^{slugs}/_{ft^3}$ at 50,000 feet