

State-of-the-art Aerodynamic Measurements on a Wind Turbine Airfoil

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Abstract

In this lab you will investigate the aerodynamic properties of the DU 96 airfoil which is commonly used for wind turbines. The experiment takes place in VT Stability Wind Tunnel. You have to measure the mean pressure distribution on the airfoil surface. The mean pressure distribution integrated over the airfoil surface provides the lift and moment coefficient. Before and after the experiment you will perform computations with the well known XFOIL code and compare the results with your measurements. Note that the confining walls of the wind tunnel have some influence on the pressure distribution on the airfoil compared to flow in "free air". A method is outlined to make the measurement comparable to the free air condition of the computation.

Introduction

You will measure the mean pressure distribution of the DU 96 airfoil for a Reynolds number of 1.65 million and angles of attack ranging from -20° to 20° . You will calculate the profile polar C_l over α and C_m over α . These measurements are a part of a larger measurement series with the goal to characterize the noise and aerodynamic characteristics of several airfoil shapes typically used for wind turbine blades.

Aerodynamic noise is a key factor for onshore installed wind turbines. It is strongly dependent on the flow speed at the wind turbine blade. Noise regulations limit therefore the rotational speed of the rotor and the rotor size of wind turbines installed in rural areas. Introducing low noise technology allows increasing the rotational speed and/or the rotor size which leads directly to a decrease of the cost of energy.

You will perform the experiment in the VT Stability Wind Tunnel (the largest university operated anechoic facility in the US). The test section of the wind tunnel is the first in the world which combines good aerodynamic performance and low background noise level, a requirement for acoustic measurements. Traditional aerodynamic wind tunnels have a hard walled test section. They cannot be used for acoustic

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measurements, because the background noise level is much higher than noise emanating from the airfoil model. Additionally, you face difficulties when placing microphones in the air flow, because the pressure fluctuations caused by the flow are normally several orders of magnitude higher than acoustic pressure fluctuations. Therefore most traditional aeroacoustic wind tunnels use an open-jet configuration which allows you to place the microphones out of the flow. However, such facilities suffer from poor aerodynamic performance, because the open jet is deflected by the airfoil model. A new test section for the VT Stability Wind Tunnel extending its capabilities to include far-field acoustic measurements was developed in 2005. In this novel test section design (now also implemented in the Japan Aerospace Exploration Agency 2m×2m wind tunnel) acoustically permeable Kevlar cloth contains the test section flow and separates it from the surrounding anechoic chamber(s). This arrangement provides a substantial reduction in aerodynamic interference (the influence of the tunnel walls on airfoil pressure distribution compared to “free air” conditions) when compared to the more conventional open-jet configuration. It also eliminates the need for a jet catcher.

When the test section was designed it was assumed that the aerodynamic performance will be decreased compared to the configuration with a hard-wall aerodynamic test section. However, the contrary was the case. The effect of wind tunnel blockage by the airfoil model was significantly smaller due to the flexibility of the Kevlar walls. Blockage corrections are found to be a fraction of those which would be associated with a hard-wall test section of the same size, and are negligible in most cases.

The VT Stability Wind Tunnel

The Virginia Tech Stability Wind Tunnel is a continuous, single return, subsonic wind tunnel with 7.3m long removable rectangular test sections of square cross section 1.83m on edge. The general layout is illustrated in Figure 1.

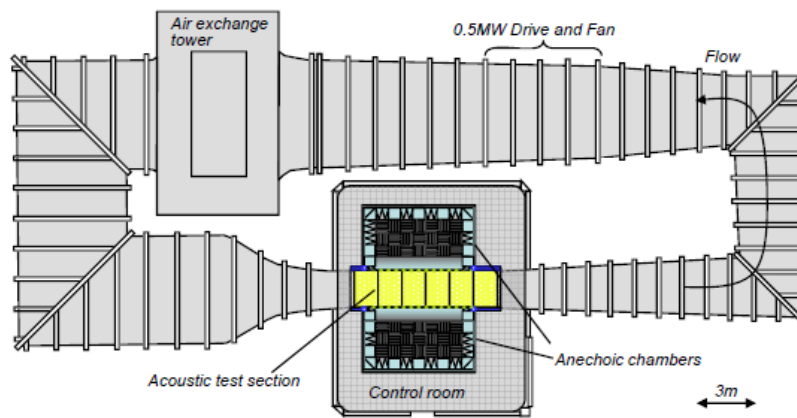


Figure 1: plan view schematic of the Virginia Tech Stability Tunnel in anechoic configuration.

The tunnel is powered by a 0.45MW variable speed DC motor driving a 4.3m propeller at up to 600 r.p.m. This provides a maximum speed in the test section (with no blockage) of about 75m/s and a Reynolds number per meter up to about 5,000,000. The tunnel forms a closed loop, but has an air exchange tower open to the atmosphere to allow for temperature stabilization. The air exchange tower is located downstream of the

fan and motor assemblies. Downstream of the tower the flow is directed into a 5.5x5.5m settling chamber containing 7 turbulence-reducing screens each with an open area ratio of 0.6 and separated by 0.15m. Flow exits this chamber through the 9:1 contraction nozzle which further reduces turbulence levels and accelerates the flow to test speed. Flow in the empty test section is closely uniform with turbulence levels of about 0.02%, increasing slightly with speed.

At the downstream end of the test section flow passes into a 3-degree diffuser. Sixteen 0.16m high vortex generators arranged at intervals of 0.39m around the floor, walls and ceiling of the flow path at the entrance to the diffuser serve to mix momentum into the diffuser boundary layer, minimizing the possibility of separation and the consequent instability and inefficiency. The four corners in the flow path (two between the air exchange tower and settling chamber, and two between diffuser and fan) are equipped with diagonal 6 arrays of shaped turning vanes. Spacing between the vanes is 0.3m except in the corner immediately ahead of the settling chamber where the spacing is 0.076m.

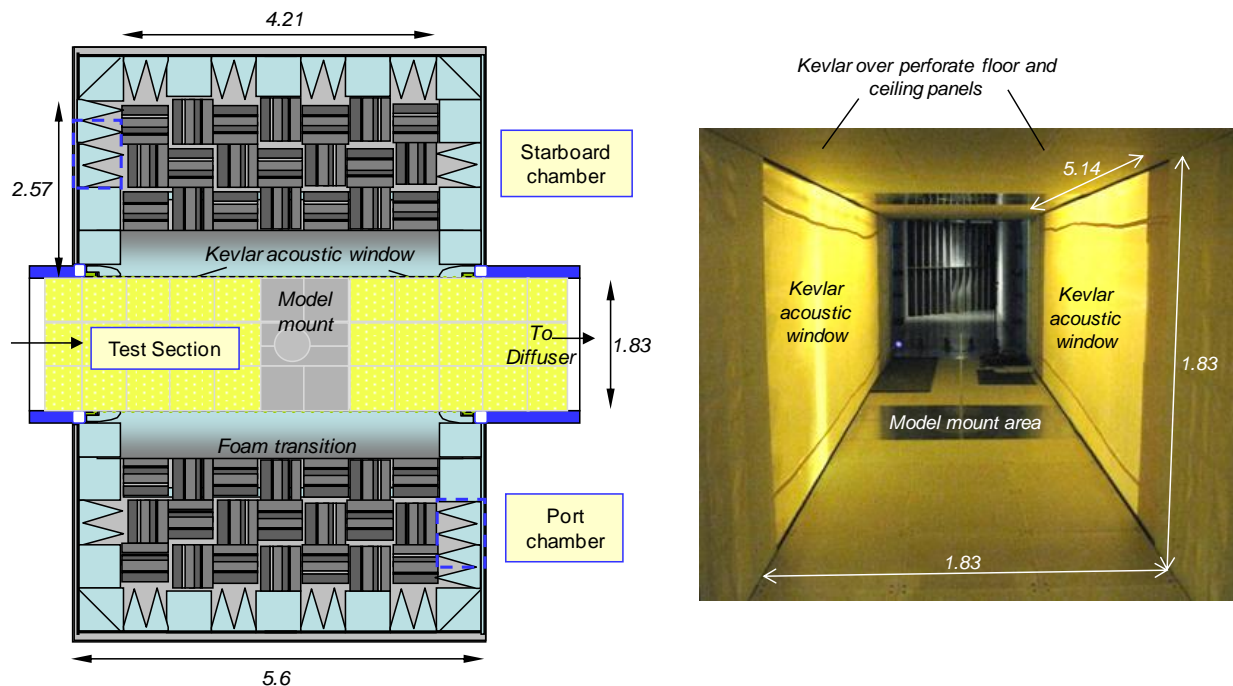


Figure 2: VT Stability Wind Tunnel Anechoic System. Dimensions in meters. Left: cross section through the anechoic test section and chambers as seen from above. Right: Photograph taken from upstream showing the test section interior.

The 1.83x1.83m acoustic test section and anechoic chambers of the Virginia Tech Stability Wind Tunnel are depicted in Figure 2. The test section consists of acoustically treated upper and lower walls that run the full 7.3m length of the test section and partial side walls at the test section entrance and exit. Large rectangular openings in the side walls which extend 5.14m in the streamwise direction and cover the full 1.83m height of the test section serve as acoustic windows. Sound generated in the test flow exits the test section through these acoustic windows into the anechoic chambers on either side. Large tensioned panels of Kevlar cloth cover these openings permitting the sound to pass while containing the bulk of the flow. The test section arrangement thus simulates a half-open jet, acoustically speaking. The Kevlar windows eliminate the need for a

jet catcher and, by containing the flow, substantially reduce the lift interference when airfoil models are placed in the test flow.

The upper and lower walls of the test section are formed from a series of perforated metal panels bonded to a layer of Kevlar cloth that forms a smooth, quiet, but acoustically transparent flow surface. The volume behind this flow surface is filled with 0.457m-high foam wedges designed to eliminate any acoustic reflections at frequencies above 190Hz. The upper and lower walls contain hardware for the (vertical) mounting of two dimensional airfoil models. Models are rotated to angle of attack about their $\frac{1}{4}$ -chord location centered midway between the acoustic windows and 3.6m from the upstream end of the test section. Models completely span the vertical height of the test section.

Plain weave Kevlar 120® cloth (58grams/m²) is used to form the acoustic windows. The cloth is stretched on 5.37x2.51m tensioning frames. The Kevlar windows are sewn from 3 lengths of Kevlar cloth. When mounted, the two 40mm-wide seams run streamwise along the test section 0.19 to 0.28m below the upper wall and a similar distance above the lower wall.

Two anechoic chambers are positioned on either side of the test section. Each chamber has a streamwise length of 6m, extends 2.8m out from the test section acoustic window, and has a height of 4.2m. The chamber walls are constructed from medium density fiberboard, supported by a network of external steel beams, and lined internally with 0.610-m high acoustic foam wedges. Quarter-elliptical foam sections surround the acoustic windows so as to form a smooth transition between the lower and upper walls of the test section, on the inside of the windows, and the acoustically treated walls of the anechoic chambers on the outside of the acoustic windows.

Airfoil Model and Aerodynamic Measurements

Aerodynamic measurements are performed on an airfoil model with DU 96 section. The section shapes is shown in Figure 3. The airfoil is 18% thick and the leading edge radius is 0.89% chord.

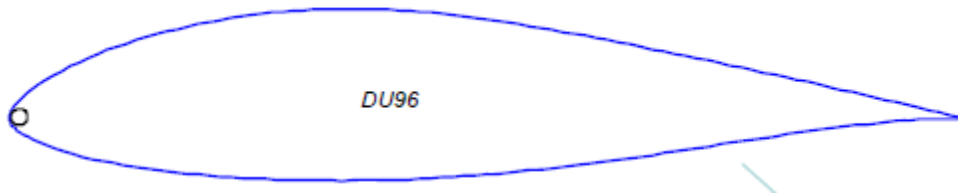


Figure 3: The DU 96 airfoil

The model, constructed by Novakinetics LLC, was designed to span the complete vertical height of the test section. It has a 1.8m span and 914mm chord and is built around 88.9-mm diameter steel tube that forms a spar centered on the quarter chord location. The model has a fiberglass composite skin and a fill of fiberboard and polyurethane foam. The steel tube projects 166mm the ends of the airfoil and is used for mounting.

The model is instrumented with about 80 pressure taps of 0.5mm internal diameter located near the midspan. The nominal chordwise locations of the pressure taps are the same on both sides of the airfoil and are listed in Table 1. The taps are connected internally to 1.6mm Tygon tubing that exits the model through the centre of the steel tube. In order to provide access to the interior of the model in the area of the pressure taps, a hatch is provided on one side of the model fixed in place using a series of flathead bolts countersunk into the

airfoil surface. Both bolt heads and the slight step at the edge of hatch are covered with 0.05mm Scotch tape during testing.

0.00E+00	7.50E-02	3.50E-01	7.50E-01
2.50E-03	1.00E-01	3.75E-01	8.00E-01
5.00E-03	1.25E-01	4.00E-01	8.50E-01
7.50E-03	1.50E-01	4.25E-01	9.00E-01
1.00E-02	1.75E-01	4.50E-01	9.20E-01
1.25E-02	2.00E-01	4.75E-01	9.40E-01
1.50E-02	2.25E-01	5.00E-01	9.60E-01
1.75E-02	2.50E-01	5.50E-01	9.80E-01
2.00E-02	2.75E-01	6.00E-01	
2.50E-02	3.00E-01	6.50E-01	
5.00E-02	3.25E-01	7.00E-01	

Table 1: Nominal chordwise locations (x/c) of pressure taps on both sides of the airfoil models

Measurements examining the accuracy of the model airfoil profile at midspan, relative to machined profile shapes provided by Novokinetics, suggested deviations no larger than 0.5mm. Pressure port locations were not measured, but assumed to be the same as their nominal values.

Flow speed is monitored via a wall mounted pressure ports in the settling chamber and contraction. These ports do not sense the pure stagnation and static pressure and thus calibration factors are used to relate these to the true free stream values. Mean surface pressure distributions on the airfoil model and the reference pressures are measured using an Esterline 9816/98RK pressure scanner with a range of ± 2.5 psi. The system has a rated accuracy of $\pm 0.05\%$ full scale. Temperature in the test section is monitored using an Omega Thermistor type 44004 (accuracy $\pm 0.2^\circ\text{C}$) and the ambient absolute pressure is determined using a Validyne DB-99 Digital Barometer (resolution 0.01" Hg).

The drag is measured using a pneumatic rake made of 90 Pitot probes (1.6mm diameter stainless steel tubes) and 5 Pitot-static probes (3mm-diameter model 167 Dwyer probes). The probes are mounted in a wing-profiled beam for minimum interference. The rake is mounted 0.92m from the floor of the test-section and 2.08m downstream of the trailing edge of the airfoil. The pressures from the probes are recorded using three 32-channel DTC Initium pressure scanners with ± 2.5 psi range and $\pm 0.05\%$ full scale accuracy.

Aerodynamic coefficients and flow similarity

When we look at absolute aerodynamic quantities like the lift force or the pitching moment, we find that those quantities depend on a large number of fluid parameters like the density, viscosity and temperature and additionally on the flow speed and model size. That makes it very difficult to apply experimental results to applications. A much better way is to non-dimensionalize the aerodynamic quantities with flow speed and model size and to look at similarity parameters rather than fluid parameters. In our example we are interested in the time average of the lift and moment coefficient for low speed flow (i.e. Mach number smaller than 0.3). It turns out that those coefficients only depend on one single similarity parameter, the Reynolds number, and the angle between free stream velocity and chord line, the angle of attack.

The Reynolds number

The Reynolds number is defined as the ratio of inertial to viscous forces. It is given by

$$Re = \frac{\rho U_{\infty} c}{\mu}$$

with the density ρ , the flow speed U_{∞} , the characteristic length (in our case the chord length) c and the dynamic viscosity μ .

The lift and moment coefficient

The lift force is the force acting on the airfoil section perpendicular to the mean flow direction. The pitch moment is the moment about the quarter chord point, positive when nose up. We measure aerodynamic quantities in the middle of the airfoil section and assume that the flow is approximately two-dimensional. In this special case it is convenient to look at the force and moment per unit span. The section lift coefficient is then defined as

$$C_l = \frac{l}{\frac{1}{2} \rho U_{\infty}^2 c}$$

with l the lift force per unit span. The section moment coefficient is respectively

$$C_m = \frac{m}{\frac{1}{2} \rho U_{\infty}^2 c^2}$$

with m the pitch moment per unit span.

The pressure coefficient

The non-dimensionalized surface pressure on the airfoil is given by the pressure coefficient as

$$c_p = \frac{p - p_{\infty}}{\frac{1}{2} \rho U_{\infty}^2}$$

The distribution of the pressure coefficient integrated along the airfoil section contour yields the lift and moment coefficient.

Wind Tunnel Corrections

Obviously the flow field in a wind tunnel with confining walls is not the same as in “free air”. For traditional aerodynamic wind tunnels with solid walls corrections have been developed over the years to make the wind tunnel measurements comparable to free field conditions. The main difference between the new test section implemented in the VT Stability Wind Tunnel and traditional ones is the transpiration of air into or out of the test section through the acoustic windows and the deflection of those windows under aerodynamic load. A correction method for this new kind of test section was developed in [1] and the results are summarized in the following. You will have to apply those corrections to your measured data.

A Simplified Model for the Angle of Attack Corrections

A series of calculations revealed that the effects of the porosity of the test section walls on the airfoil pressure distributions were found to be almost indistinguishable from the effects of an angle of attack change. We therefore define the proportionate angle of attack correction δ as,

$$\delta = \frac{\alpha_g - \alpha_e}{\alpha_g - \alpha_{0l}} \quad (1)$$

where α_g is the geometric angle of attack, α_e is the observed effective angle of attack and α_{0l} is the zero lift angle of attack of the airfoil section. That is, the effective angle of attack is given by

$$\alpha_e = \alpha_g - (\alpha_g - \alpha_{0l})\delta \quad (2)$$

Sets of panel method calculations were performed to determine the dependence of the effective angle of attack correction δ first on angle of attack for two substantially differently shaped airfoils. It was found that the correction is not a function of the airfoil shape. The flow situation in the wind tunnel was simplified to derive the analytical correction formula (Figure 4). The deflection of the test section walls was not taken into account at this stage. An analytical expression which matches the behavior observed in the calculations is

$$\delta \approx \frac{\text{sgn}(C_l)}{\alpha_g - \alpha_{0l}} \frac{C}{U_\infty} \left(\frac{1}{2} \rho U_\infty^2 \frac{|C_l| c}{\pi h} A \right)^n \quad (3)$$

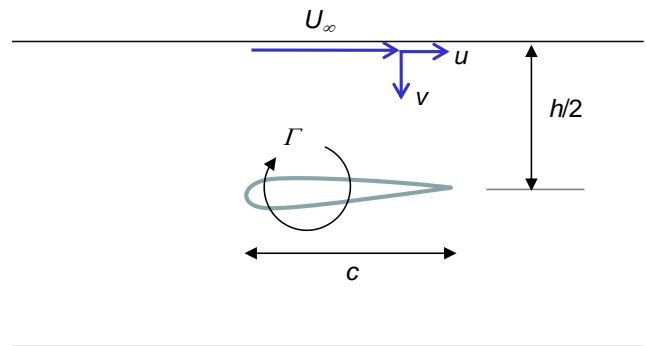


Figure 4: Schematic showing variables used in the derivation of the porosity correction formula

where c , h and U_∞ are defined in Figure 4, C_l is the section lift coefficient, n is an empirical exponent ($n = 0.5734$), A is a dimensionless constant simply set to $A=1$ and C a dimensional constant ($C = 0.03879 [m^{1+n} s^{-1+2n} kg^{-n}]$).

The Blockage Effect

In the presence of confining solid walls introducing an airfoil in the test section will effectively reduce the area through which the air must flow compared to free flow conditions. Upon that, the airfoil model also creates a wake that has a mean velocity lower than the free stream [2]. By continuity and Bernoulli's equation, the velocity in the vicinity of the model and outside of the wake must increase. Measurements in the VT Stability Wind Tunnel have shown that due to the new kind of test section with flexible walls blockage effects are strongly reduced. You can ignore the effect of blockage when processing your data.

Measurement Procedure

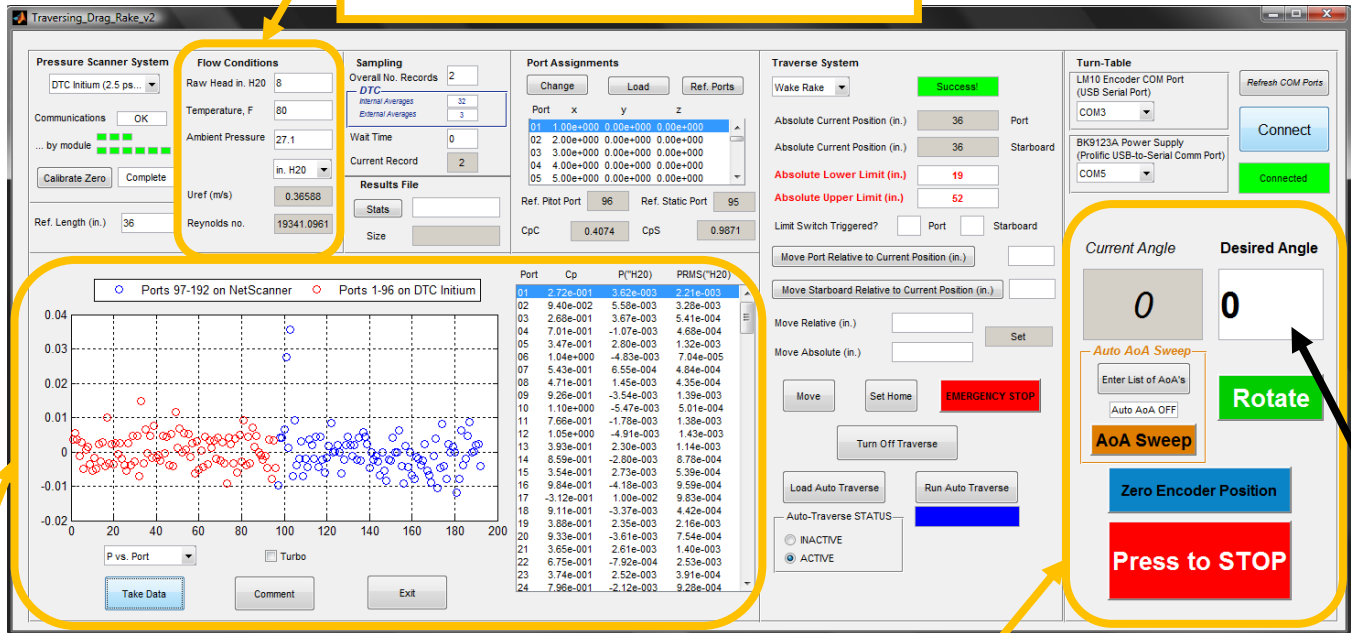
Bring your own digital camera when you do the experiment and take pictures of the wind tunnel test section and the airfoil model!

Send the measurement data to your VT email account when finished with the experiment!

You will perform an angle of attack sweep from -20° to 20° in steps of 1° . The measurements have to be done for a Reynolds number of 1.65 million based on the chord length of the airfoil model. Calculate the corresponding velocity. The wind tunnel is operated by a lab technician. He will adjust the flow speed but you will be responsible for the data acquisition and the angle of attack of the airfoil.

When you arrive in the tunnel, the data acquisition software (developed in house using Matlab) will be up and running (see Figure 5 for a screenshot of the code).

Flow condition panel: for each angle of attack, enter the flow dynamic pressure (raw head in H₂O), temperature (in F) and atmospheric pressure (in inHg). Ask the tunnel operator to locate these values.



Plotting panel: for each record, the code plots the raw pressures or coefficient of pressure for all the ports. Ports on the drag rake (number 1-96) are shown in red. Ports on the airfoil (number 97-182) are presented in blue. The data used to create these plots is given in the table on the right.

Angle of attack panel: To change the angle of attack, enter the value for the desired angle (in degrees) here. Then press **ROTATE**. The value of the "Current Angle" display will change until the airfoil reaches the "Desired Angle" value.

Figure 5 Screenshot of the Matlab data acquisition, rake and angle of attack control software

Below you will find a quick description on how to operate the code:

1. Zero calibration.
 - a. Press "Calibrate Zero". The fan of the wind tunnel has to be switched off at this moment. The zero calibration compensates for the internal offset of the pressure scanner.
 - b. Set number of records in the "Sampling" panel to 10 and make a measurement by pressing the button "Take Data".
 - c. Check in the graph that all port measure roughly 0inH₂O. A deviation of 0.05inH₂O is tolerable. You might have to choose "pressure vs port" in the drop down menu below the graph to visualize the pressure. Note channels with an offset in your measurement log.
2. Load the port location file.
 - a. Press "Load file" in the menu "Pressure Port Assignment".
 - b. Open the file **DU96PortLocation.txt** which is in the desktop directory.

3. Set the reference ports for flow speed monitoring.
 - a. Press “Ref. ports” in the menu “Pressure Port Assignment”. A warning window will pop-up. Press OK and a menu as displayed in Figure 6 will pop up.

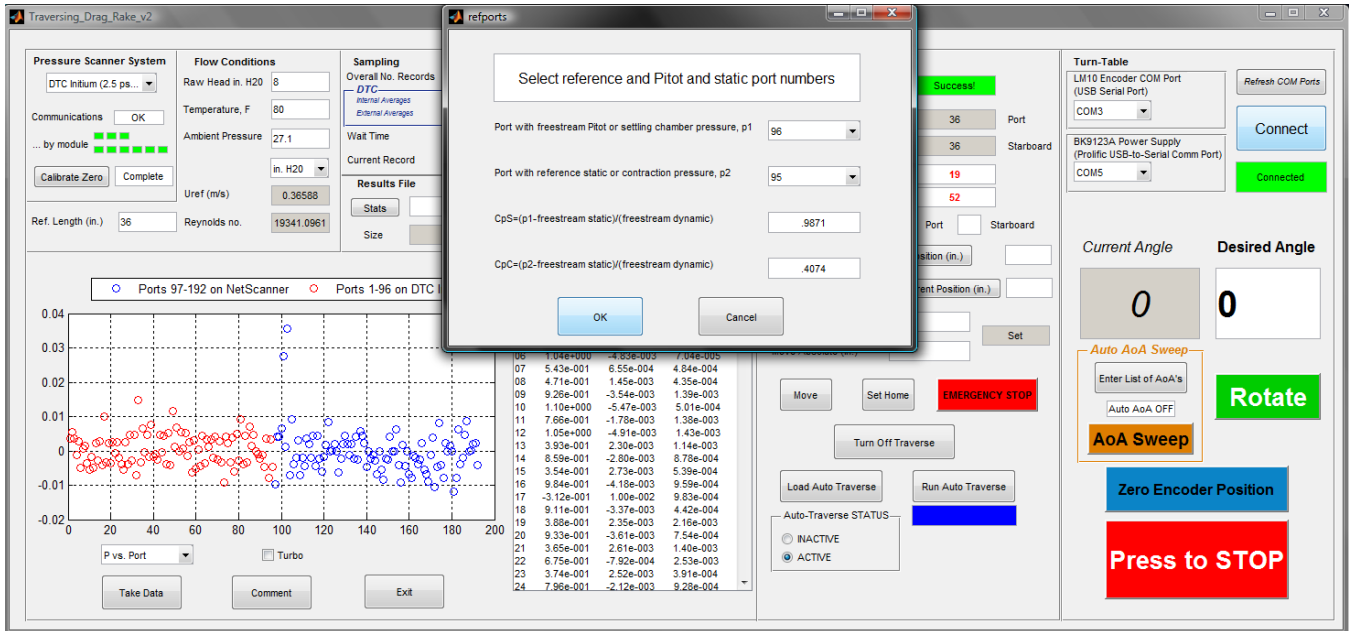


Figure 6: Screenshot of the data acquisition program with menu for pressure port assignment

- b. Set the settling chamber pressure port to 96.
 - c. Set the contraction pressure port to 95.
 - d. Info: The total and static pressure in the test section deviate somewhat from the pressures in settling chamber and downstream of the contraction. The calibration factors CpS and CpC compensate for this deviation.
 - e. Set CpS to 0.9871.
 - f. Set CpC to 0.4074.
4. Final check.
 - a. Tell the lab technician to set the wind speed to 10m/s.
 - b. Take a measurement by pressing “Take data”.
 - c. Check that the raw head in the menu “Flow Conditions” shows the same value as “Q” on the screen for the wind tunnel control.

If your last check was positive you have set up the data acquisition system correctly and can proceed with the measurement. Follow the list below to start the measurement.

1. Set the reference length in the field “Ref. Length” to 36 in. (the chord length of the airfoil model).
2. Press “Stats” in the menu “Sampling” and you will be asked for the file name in which your data will be stored. Name the file *DU96_Re1.65m_cp.stat*.

3. Press “Comment” in the plotting panel. A new window will pop-up asking you to enter information about your run (this information will be saved as a header each time you press the “Take Data” button). See below.

4. Tell the lab technician to run the wind tunnel to your desired velocity and to set the model to the desired angle of attack.
5. Enter 50 in the field “No. Records” to acquire 50 samples (gives decent accuracy of the mean value).
6. When the lab technicians tells you that the flow is stable, enter the flow temperature and the ambient pressure (make sure that the unit is in. Hg) in the fields “Temperature F” and “ambient pressure”. You can read these values from the wind tunnel control screen.
7. Press “Take data”.
8. Repeat the procedure for the next angle of attack. You have to update the temperature and ambient pressure every time before you take data, because it will change during operation of the wind tunnel.
9. Press “Stats” after the last measurement to save the file *DU96_Re1.65m_cp.stat*.

Processing of the Measured Data

Lift

Use the MATLAB script *CICmDU96_2011.m* to obtain the sectional force coefficient normal and axial to the chord line as well as the moment coefficient about the quarter chord point. A window will pop up to ask you for the file with your c_p data. Open the file *DU96_Re1.65m_cp.stat*. The lift coefficient can be obtained from normal and axial force coefficient by the relation

$$C_l = C_n \sin \alpha - C_a \cos \alpha.$$

α is the angle of attack which is not known until you apply the interference correction given by equation (3). However, the interference correction depends on the lift coefficient. To obtain a very accurate result, one is forced to iterate with those two quantities. However, for reasons of simplicity, it is ok if you use the geometric angle of attack to calculate the lift coefficient and then calculate the effective angle of attack by equations (1) to (3). Write your own MATLAB code to read the file with your c_p data and plot c_p over x/c .

Drag

Run the MATLAB script ***WakeProfilePlotDrag_2011.m*** to obtain the drag. The code will plot the wake profile for each angle of attack and determine the edges of the wake (used a integration limits for the drag). The edges will be shown by two dashed lines. The code will ask you to enter a vector of bad ports (explained below) and a trip condition.

To obtain reliable drag values you will have to ensure that there are no bad ports in the profiles (such ports would appear as obvious outliers). Once you have identified bad ports, just enter the port number(s) during the initial calling of the function.

The code will produce the Cd variable that contains the drag value at each geometric angle of attack. You can then plot the drag against the effective angle of attack determined in the lift analysis.

Computations

To compare with your measurement results you have to perform free-flight computations with the XFOIL code.

The XFOIL code

XFOIL's inviscid formulation is based on the potential flow theory. The airfoil geometry is described via a source panel method. Boundary layer and wake of the airfoil are neglected in this formulation. In the viscous formulation of XFOIL, boundary layers and wake are modeled with a two-equation lagged dissipation integral boundary layer formulation. Transition from laminar to turbulent is modeled via standard e^N -criterion. The incompressible inviscid flow outside the boundary layer and wake region is solved in the same way as the flow field in the inviscid formulation.

A correct prediction of the transition from laminar to turbulent boundary layer flow is very difficult. The e^N -criterion used in XFOIL is a much simplified way to describe reality. Upon that there is also an uncertainty in the choice of the N-factor. N depends mostly on the turbulence level of the flow around the airfoil section, but other factors like for example the airfoil shape itself play a role. Table 2 can help you to chose N.

n_{cr}	Tu [%]
4	0.563
5	0.371
6	0.245
7	0.161
8	0.106
9	0.070
10	0.046

Table 2: Dependence of N factor on turbulence intensity according to XFOIL vpar menu

Before the Lab

Perform inviscid and viscous computations with the XFOIL code for the DU 96 airfoil and for a range of angles of attack from -20° to 20° in steps of 1° . Compare the results to see the effect of the boundary layer. In the viscous formulation the Reynolds number plays an important role. Start with a Reynolds number of 1.65 million and vary it in another series of calculations to see its impact on the result. You should also make a parameter study with varying N for a fixed Reynolds number of 1.65 million.

Interpret the C_l over α polar for a Reynolds number of 1.65 million and $N=9$. Write a MATLAB program to read and plot the surface pressure distribution from the XFOIL output files and use these plots for your interpretation. Compare the polar plots (C_l over α and C_l over C_m) and surface pressure distributions for varying parameters. If you are not familiar with XFOIL see Appendix A for help to get started. You find an example which shows you how to calculate the surface pressure distribution of a NACA0015 airfoil section for two angles of attack and write the polar data for these two angles to another file.

After the experiment

Make a viscous calculation with XFOIL using the same Reynolds number as in your experiment and vary N to get a good agreement between the XFOIL results and the measurements. A good starting point is $N=9$.

Presentation of your Results

In plotting and presenting data, normalize x (chordwise distance) with the chord length of the airfoil model. Present the profile polar C_l vs. α and C_d vs. α and C_d vs. C_l . Plot a couple of the C_p vs. x/c distributions.

Use the plots to illustrate the influence of the Reynolds number and N-factor on your XFOIL computations. Show the difference between viscous and inviscid computations.

Compare the results of your measurement with the XFOIL calculation. Explain differences between the calculation and measurement.

Point out what the measurements and computations tell you about the airfoil properties (maximum lift coefficient, stall behavior etc.).

Note that plots of experimental data should consist of symbols connected by straight lines – no curves between points.

Be aware that neither the XFOIL program nor the wind tunnel experiments are perfect. You will not get THE ANSWER from one of them. Keep that in mind when you interpret your results!

Report Expectations

1. Title Page and Abstract
2. Introduction
3. Description of the Experiment
 - a. Describe the wind tunnel and airfoil model (use the information provided in this document and pick out the most important things, write in your own words!)
 - b. Make a 2D sketch of the airfoil in the wind tunnel with the most important dimensions
 - c. Describe the data acquisition system and the pressure measurements
 - d. Discuss the calibration procedure of the pressure measurement system
 - e. Describe the test conditions (Reynolds number and angle of attack)
4. Computations
 - a. Describe briefly the XFOIL code
 - b. Comment on the settings of the calculations (put your batch file in the Appendix if necessary)
5. Results of Experiment and computations
 - a. Polar and c_p distribution plots with explanation of shapes
 - b. Comment on the influence of the Reynolds number, N-factor and viscosity on computations
 - c. Describe the corrections you applied to the measured data and how it changed them
 - d. Compare computations and measurement
 - e. Comment on the overall airfoil performance
6. Conclusions
 - a. Repeat major findings
 - b. What did the lab show you

References

[1] Devenport, W.J., Burdisso, R.A., Borgoltz, A., Ravetta, P. and Barone, M.F., "Aerodynamic and Acoustic Corrections for a Kevlar-Walled Anechoic Wind Tunnel", AIAA paper 2010-3749, 2010.

[2] Barlow, J.B., Rae, W.H. jr. and Pope, A., "Low-Speed Wind Tunnel Testing", 3rd Edition, John Wiley & Sons Inc., 1999.

Appendix A: How to run XFOIL

To run XFOIL in batch mode

Copy the XFOIL application *xfoil.exe* and the batch file *example_cp2.inp* in your working directory. Open a dos window and go to the path of your working directory. Type *xfoil<example_cp2.inp* to run XFOIL with the batch file *example_cp2.inp*.

In the following two example batch files are presented which show you how to compute the pressure distribution and the lift curve for a NACA0015 airfoil. You have to change the file that you can load the DU96 airfoil into the program and you have to adjust the flow parameters.

If you are not familiar with XFOIL, please read the manual before (provided in the file *xfoil_doc.txt*). You can find a nice XFOIL tutorial on the web page <http://openae.org/airfoil-resources/xfoil-tutorial/xfoil-resources-tutorial-part1>.

The batch file *example_cp2.inp* to compute the polar and the pressure distribution of a NACA0015 airfoil in the viscous formulation

! An empty line in the batch file has the same effect as pressing the *ENTER* button in the XFOIL program!

! By pressing enter you move from a sub menu to the next higher level, it is important not to change the number of empty lines in the batch file!

```
naca 0015          ! loads the NACA 0015 airfoil
pane              ! smoothes the airfoil geometry
ppar             ! show/change paneling
n                ! command to enter a new number of panels
140              ! number of panels
                 ! you can increase the number of panels to achieve a better accuracy, but if you
                 ! choose to many panels to calculation might not converge
oper             ! go to menu to choose operation mode (viscous/inviscid)
visc             ! viscous mode
1.8e6            ! Reynolds number
mach            ! command to enter the mach number
0.0876           ! mach number
vpar            ! go to the change boundary layer parameter menu
n               ! command to enter the N factor of the transition criterion
8               ! N factor

vpar            ! go to the change boundary layer parameter menu
xtr             ! fix the laminar-turbulent transition point xtr/c, 1.0 means natural transition according
                ! to eN-model
```

1.0 ! forced transition location x/c on the suction side
1.0 ! forced transition location x/c on the pressure side

pacc ! enables point accumulation for polar calculation
naca0015pol.dat ! file name in which polar data is stored

alfa ! prescribe angle of attack for first calculation point
0 ! value of the angle of attack in deg.
cpwr ! Output x vs C_p to file at specified calculation point
cp_NACA0015_AOA00.out ! name of file in which x vs C_p is stored
padd ! Add point to polar save file
alfa
2
cpwr
cp_NACA0015_AOA02.out
pacc ! turn off point accumulation

quit ! quit XFOIL