

Lab report TEP4160

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Abstract

This lab was realized to compute and obtain the necessary data for the lift of the the NREL S826. The airfoil was set in a wind tunnel and a Scanivalve MPS4264 pressure scanner aquired data from 32 pressure points located around the airfoil. The data were interpolated using a piecewise cubic Hermite polynomials to plot the C_p and C_L curves. The experimental results are close to the analytic ones, but diverge with the error ΔC_L as the maximal error is 5.93% of the total C_L .

1. Introduction

There exists many types of airfoils. They have different lengths, shapes and material, which leads to different performance. Therefore, it can be useful to measure the properties of the different airfoils in order to choose the right one. This experiment is based on the NREL S826 airfoil. On the airfoil, one of the middle section contains 32 pressure ports around the chord, on both the pressure and suction sides of the airfoil. The airfoil will be set at different angle of attacks (AoA), and the pressure will be calculated for each port. Then it will be mapped to a pressure scanner and interpreted with a Matlab code.

2. Theory

Calculation of the pressure coefficient C_p

From the lectures notes, C_p can be obtained by the eq.(1) :

$$-C_p = \frac{P - P_\infty}{\frac{1}{2} * \rho * U_\infty^2} \quad (1)$$

Where P and P_∞ are respectively the local pressure and the atmosphere pressure far away. U_∞ is the wind velocity of the wind tunnel.

Calculation of the lift coefficient C_L

First, the expression for the pressure force acting normal to the chord line is given in the lecture notes as eq.(2),

$$C_n = \oint C_p(-\hat{n} * \hat{y})dl, \quad (2)$$

with C_p the coefficient of lift and \hat{n} the unit normal vector pointing out of the surface, \hat{y} is the unit vector in the direction normal to the chord line. dl is the length of an infinitesimal line element. Similarly, the axial component can be express as eq.(3)

$$C_a = \oint C_p(-\hat{n} * \hat{x})dl, \quad (3)$$

with \hat{x} , the unit vector pointing along the chord line. A piecewise cubic Hermite interpolation (PCHIP) was used to estimate the surface location and C_p values. With some calculation and rearrangement , eq.(3) can be express as :

$$\begin{pmatrix} C_a \\ C_n \end{pmatrix} = C_{p,0} \begin{bmatrix} dy_0 + dy_{N-2} \\ -dx_0 - dx_{N-2} \end{bmatrix} + \sum_{i=1}^{N-2} C_{p,i} \begin{bmatrix} dy_i + dy_{i-1} \\ -dx_i - dx_{i-1} \end{bmatrix} \quad (4)$$

The eq.(4) gives the expression for the pressure force coefficient in the normal and axial direction in the coordinate system where the cord length is in the (x) direction. To get the lift coefficient, the coordinate system need to be rotated by $\theta = -\alpha$ in eq.(4) .This gives the eq.(5):

$$\begin{pmatrix} C_d \\ C_l \end{pmatrix} = \begin{bmatrix} \cos(\alpha) & \sin(\alpha) \\ -\sin(\alpha) & \cos(\alpha) \end{bmatrix} \begin{pmatrix} C_a \\ C_n \end{pmatrix} \quad (5)$$

Rewriting eq.(5) with eq.(4) , the final formula for C_L is given in eq.(6),

$$C_L = \frac{1}{2} * [Cp_0 * (-(dy_0 + dy_{N-2}) * \sin(\alpha)) - (dx_0 + dx_{N-2}) * \cos(\alpha) + \sum_{i=1}^{N-2} Cp_i * (-(dy_i + dy_{i-1}) * \sin(\alpha) - (dx_i + dx_{i-1}) * \cos(\alpha))], \quad (6)$$

with d_x and d_y , the infinitesimal variations of the pressure point position are the differences in respectively the x and y direction between two pressure point.

Errors in the lift coefficient C_L

The error of the lift coefficient C_L is estimated as in eq.(7),

$$\Delta C_L = \sqrt{\left(\frac{\partial C_L}{\partial z_1}\right)^2 * (\sigma_{z_1})^2 + \left(\frac{\partial C_L}{\partial z_2}\right)^2 * (\sigma_{z_2})^2 + \dots} \quad (7)$$

The variances σ_z are estimated using the variance of the time series, and the known accuracy of the measuring equipment, with eq.(8) :

$$\sigma_\alpha = \frac{\sum (x - \mu)^2}{N}, \quad (8)$$

with μ the average value.

To determine the error in the C_L coefficient, it was assumed that there are no significant errors in the location of the points on the airfoil surface. With eq.(7) and partial differentiation of each P_i value, the uncertainty of the lift coefficient C_L was obtained in eq.(9),

$$\begin{aligned} \Delta C_L = & \left[\left(\frac{C_L}{q_\infty} * \sigma_{q_\infty} \right)^2 \right. \\ & + (\sigma_\alpha * (\sin(\alpha) - \cos(\alpha))) * \sum_{i=0}^{N-2} \frac{C_{p,i+1} + C_{p,i}}{2} * \left(\frac{d_{x_i}}{d_{y_i}} \right)^2 \\ & + \left(\frac{1}{2 * q_\infty} * ((dy_0 + dy_{N-2}) * \sin(\alpha) \right. \\ & \left. + (dx_0 + dx_{N-2}) * \cos(\alpha)) * \sigma_{P_0} \right)^2 \\ & + \sum_{i=1}^{N-2} \left(\frac{1}{2 * q_\infty} * ((dy_i + dy_{i-1}) * \sin(\alpha) \right. \\ & \left. \left. + (dx_i + dx_{i-1}) * \cos(\alpha)) * \sigma_{P_i} \right)^2 \right]^{\frac{1}{2}} \quad (9) \end{aligned}$$

3. Method

First, the surface pressure was measured with a pressure scanner (Scanivalve MPS4264). The MATLAB script *get_qinf_T.m* was used to find the target freestream velocity U_∞ .

A first offset measurement was taken with the pressure scanner, sample at 800 Hz for 10 seconds , while matlab was taking an offset measurement. After the offset measurement done , the wind tunnel VFD RPM was set to reach the target U_∞ within $\pm 0.5m/s$. For each of the following $\alpha = [-8 -6 -4 -2 0 2 4 6 7 8 9 10 11 12 13 14 16 18]$, the same procedure was repeated :

The turntable was set to the right angle of attack (as shown in fig.(1)). Then the dynamic pressure and the temperature were taken (1000 Hz for 30 seconds for pressure, and 14 Hz for 10 seconds for the temperature).

While Matlab was taking the data , the pressure scanner was run to take measurement at 800 Hz for 60 seconds. After changing the angle, a break of 5 seconds was taken in order to fully settle the flow into a steady state before taking the next set of measurements.

The post-experiment calculations were realized with Matlab. First, the pressure offset was computed in order to get the right pressure measurement. With the 2 offset measurements and the *getfiledate.m* Matlab code, the time of each offset has been taken. A linear interpolation was realized to get the offset at any time.

The pressure points were linked to the corresponding measurement value of the scanner and the time of each measurement was obtained with the *getfiledate.m* code. The new pressure were finally taken by subtraction of each corresponding time offset to the measurement pressure for every angle of attack.

The lower and upper C_p values were computed with eq.(1). The denominator in the eq.(1) ($P - P_\infty$) correspond to the new pressure calculated by subtraction of the offset . As the pressure points does not surround the airfoil entirely, the C_p curves had to be closed by interpolation of the data points using piecewise cubic Hermite polynomials (PCHIP) for the last three points to estimate a value for the trailing edge. An example of a C_p curve for a certain angle of attack is shown in fig.(5).

Next, the C_L values for each angle of attack were com-

puted using eq.(6). The coordinate system used in eq.(6) is shown in fig.(2). fig.(5) shows the resulting plot of this calculation.

Finally, the errors in the lift coefficient were computed using eq.(9). The different variance values were given in the lab document and calculated using eq.(8). fig.(3) shows the resulting plot of this calculation.

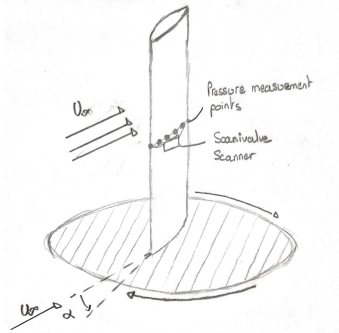


Figure 1: Set up of the airfoil experiment

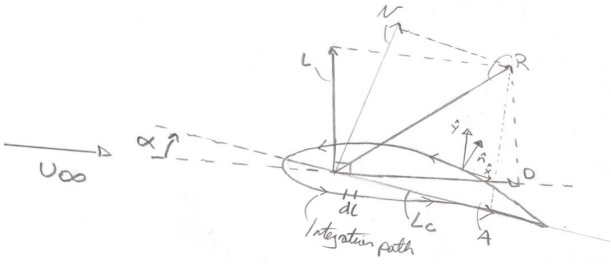


Figure 2: Airfoil spatial measurement

4. Results

The resulting slope of the C_l for $\alpha \in [-8, 8]$ is 6.174 rad and 6.209 rad for $\alpha \in [-4, 4]$. This deviates by 0.1090 and 0.0745 respectively from the 2π value predicted by thin airfoil theory, indicating larger errors for higher AoA's.

The max theoretical error ΔC_l was calculated to be 0.0887 , and occurred at $\alpha = 16^\circ$, which is in the stall region. Outside of the stall region the max error was calculated to be 0.0391 , at $\alpha = 8^\circ$.

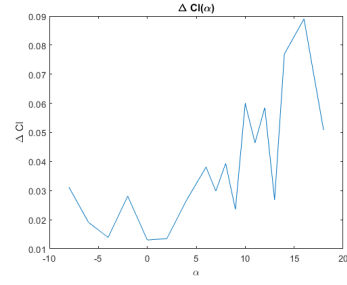


Figure 3: Resulting plot of ΔC_L

The standard deviations presented in tab.1 were used in the result above. $\sigma_{q_{inf}}$, and σP_i were found with eqn (8). However, σP_i is a vector for all of the pressure ports, and will not be presented.

Table 1: Value of variance

σP_0	σ_α	$\sigma_{q_{inf}}$
3.000	0.250	0.453
[Pa]	[deg]	[Pa]

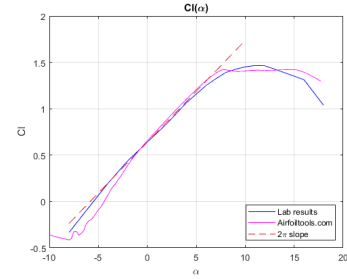


Figure 4: Resulting plot of C_L compared to experimental data

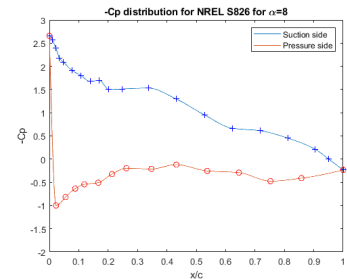


Figure 5: $-C_p$ for $\alpha = 8^\circ$

5. Discussion

According to thin airfoil theory, the C_l curve for cambered airfoils should be straight for low angles of attack with a slope of 2π . It should also have a positive lift at $\alpha = 0^\circ$. The resulting C_L curve clearly follows this trend, albeit not perfectly, especially at higher AoA's. This likely follows from the assumption of a thin airfoil, as the NREL S826 has a non negligible aspect ratio of 5 .

Furthermore, the boundary layer acts as a streamline, essentially adding some minute thickness to the airfoil flow. It would therefore experience a higher adverse pressure gradient due to the curvature, and thus earlier separation. This can also be observed in figure 4, where a high pressure gradient is starting to form already for $\alpha = 8^\circ$ at $\frac{x}{c} \approx 0.2$.

Furthermore, stall can be predicted to be about $\alpha = 12^\circ$ from figure 3. This seems to fit well with previous experimental data shown in pink [2], . Larger theoretical errors are expected in this region, as separation and irregular flow further complicates the theory.

The discrepancies are also likely to be due to the measurement errors described in the theory section. The max calculated error ΔC_L is *5.93* % of the total C_L .

6. Conclusion

The goal of this lab was experimentally measure pressure around an airfoil for different AoA's and to compare the resulting lift data with theory. This was done with numerical integration of the pressure distribution, while also adjusting for measurement errors.

There seems to be good agreement between the lab data and theory. The resulting slope of the C_L curve deviates at a maximum *0.109* from thin airfoil theory outside the stall region. This is probably due to the thickness of the airfoil, as well as the measurement error in the equipment. As expected stall occurs at about $\alpha = 12^\circ$, which can be qualitatively observed in both the C_L and C_P curves.

References

- [1] Scanivalve: MPS4264 Miniature Pressure Scanner Manual, <http://www-cs-faculty.stanford.edu/~uno/abcde.html>
- [2] Airfoil tools: Previous experimental data for the NREL S826, <http://airfoiltools.com/airfoil/details?airfoil=s826-nr>