Incompressible Aerodynamics Wind Tunnel Lab Report

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Word Count	1190
Page Count	5

Abstract

This is an experiment to obtain aerodynamic characteristics of a NACA 0012 airfoil, and then compare results to those from XFoil and also those from Thin Airfoil Theory, it is conducted in a low-speed wind tunnel, and then the data is processed to calculate local pressure coefficient, lift coefficient, moment coefficient and also drag coefficient (although not in a direct way), the results don't seem to be so far from each others especially at low angles of attack, but they differ somehow at large angles of attack due to increase of viscous effect and occurrence of stall.

1 Introduction

Our goal is to analyze the airfoil NACA 0012 for different angles of attack using 3 methods: experimental, XFoil analysis and thin airfoil theory. We need to calculate the aerodynamic coefficients: C_l, C_d, C_m. C_l and C_m are calculated using the local pressure coefficient over the surface of the airfoil, however, C_d is a challenge because we can't measure axial force on the airfoil directory. So, a solution is to use rake pitot tubes behind the airfoil to detect the vertical change in pressure in order to calculate the total drag due to the presence of the airfoil inside the flow.

2 Methodology

2.1 Equipment

The experiment is to be performed in a low speed open-return wind tunnel with a test section measuring 0.457m x 0.457m and whose length is 1.4m, the airfoil has a chord $c = 0.152$m. and a span filling the width of the test section and is put in the middle of the cross-sectional area of the test section as shown in figure 1b.

There are openings in the upper and lower surface of the airfoil in order to insert the pressure measuring tubes, there are also tubes behind the airfoil to enable us to calculate the drag, the exact distribution of the pressure tubes on the airfoil and tubes of the wake rake are given in [1].

2.2 Steps

1. The conductor of the experiment operates the wind tunnel (regardless of the velocity of the freestream) and gradually rotates the airfoil until the mercury level in all tubes is equal, this means that the net pressure above and below the airfoil is zero, which in turn means that the airfoil is set to zero angle of attack (because it's a symmetric airfoil).

2. Adjust the velocity of the free-stream to be 20 m/s.

3. Traverse the wake rake until the wake position is found.

4. Read the pressure for all the pressure taps above and below the airfoil and also in the rake.

5. Repeat steps $2, 3$ and 4 for different angles of attack: $\alpha = 3^\circ,\ 6^\circ,\ 9^\circ,\ 12^\circ,\ 15^\circ$.
6. Repeat all steps to obtain 4 datasets of each measured quantity to overcome some uncertainties.

(a) Wind Tunnel

(b) Airfoil Fixed in the Wind Tunnel

Figure 1: Wind Tunnel Images, credits to Eng. Abdallah Lotfy

2.3 Processing

1. Since we have four datasets for each reading in the experiment, we shall take the average of these values.
2. For a given angle of attack α, calculate the local pressure coefficients $C_{p,i}$ on the upper and lower surfaces of the airfoil, using:

$$C_{p,i} = \frac{p_{s,i} - p_\infty}{p_T - p_\infty} \tag{1}$$

3. Calculate normal force coefficient C_n, using trapezoidal integration but for discrete integration:

$$C_n = \sum_i^{11} \frac{C_{p,u}(i+1) + C_{p,u}(i)}{2} * \left(\left(\frac{x}{c}\right)_{i+1} - \left(\frac{x}{c}\right)_i\right) - \sum_i^{10} \frac{C_{p,u}(i+1) + C_{p,u}(i)}{2} * \left(\left(\frac{x}{c}\right)_{i+1} - \left(\frac{x}{c}\right)_i\right) \tag{2}$$

4. Calculate C_l:

$$C_l = C_n \cos(\alpha) - \cancel{C_A \sin(\alpha)}^{\,0} \tag{3}$$

5. Calculate $C_{m,c/4}$ with $x_{ref} = \frac{1}{4}$:

$$C_m = \overbrace{\frac{2}{c}\sum_i^{10} \frac{\left(c_{p,l_{i+1}} + c_{p,l_i}\right)}{2}\left[\left(\left(\frac{x}{c}\right)_i + \left(\frac{\left(\frac{x}{c}\right)_{i+1} - \left(\frac{x}{c}\right)_{i-1}}{c_{p,l_{i+1}} + c_{p,l_i}}\right) c_{p,l_{i+1}}\right) - \frac{1}{4}\right]\left(\left(\frac{x}{c}\right)_{i+1} - \left(\frac{x}{c}\right)_i\right)}^{Lower} - \overbrace{\frac{2}{c}\sum_i^{11} \frac{\left(c_{p,u_{i+1}} + c_{p,u}\right)}{2}\left[\left(\left(\frac{x}{c}\right)_i + \left(\frac{\left(\frac{x}{c}\right)_{i+1} - \left(\frac{x}{c}\right)_{i-1}}{c_{p,u_{i+1}} + c_{p,u_i}}\right) c_{p,u_{i+1}}\right) - \frac{1}{4}\right]\left(\left(\frac{x}{c}\right)_{i+1} - \left(\frac{x}{c}\right)_i\right)}^{Upper} \tag{4}$$

6. Still, we're with the same α, calculate $\frac{u(y)}{V_\infty}$, and assume $P_{s,w} = P_{s,\infty}$:

$$\frac{u(y)}{V_\infty} = \sqrt{\frac{P_{T,w}(y) - P_{s,w}(y)}{P_{T,\infty} - P_{s,\infty}}} \tag{5}$$

7. Calculate the value $var_i = \left(\frac{u(y)}{V_\infty}\right)_i * \left(1 - \left(\frac{u(y)}{V_\infty}\right)_i\right), \ \forall\ i = 1, 2, 3, ..., 13$

8. Calculate C_d:

$$C_d = \frac{2}{c} \int_{y=-60}^{y=+60} \frac{u(y)}{V_\infty}\left(1 - \frac{u(y)}{V_\infty}\right) dy = \frac{2}{c} \sum_{1}^{12} \frac{var_{i+1} + var_i}{2} * (y_{i+1} - y_i) \tag{6}$$

9. Now, repeat all these steps for each angle of attack.

3 Results and Discussion

Wind Tunnel Corrections

Due to the blockage effect of the wind tunnel, there are three modes of calculations going wrong:

1. The so-called "solid blockage" due to a lateral constraint to the flow pattern about the aerofoil. This effect is a function of model thickness and size, and it results in an equivalent increase in dynamic pressure and hence an increase in forces and moments measured at a given angle of attack.

2. The so-called "wake blockage" due to a lateral constraint to the flow pattern about the wake. This effect increases with the size of the airfoil wake and leads to a drag increase of the model.

3. An alteration to the streamline curvature of the flow around the aerofoil due to the constraint from the tunnel ceiling and floor. This results in an increase in the airfoil lift and moment coefficient.

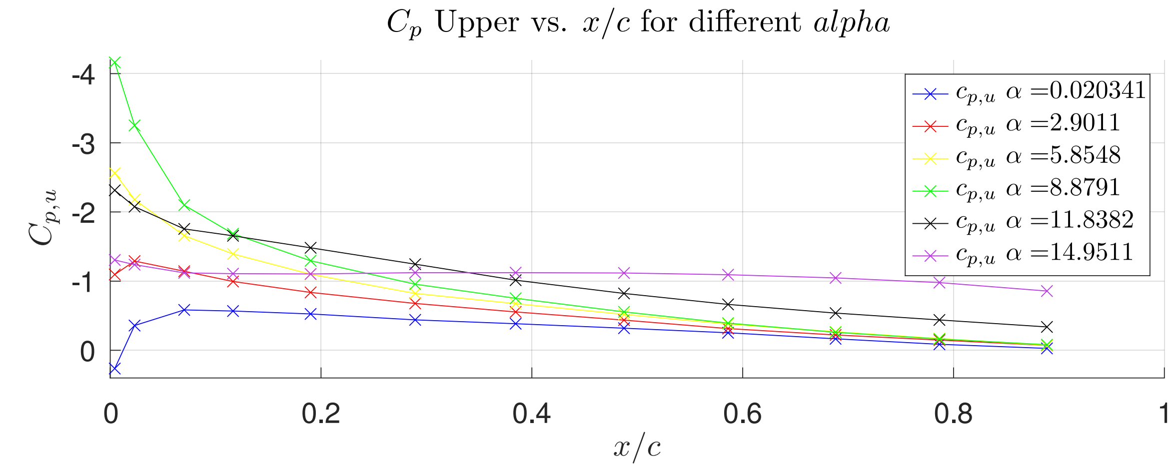
By a rule of thumb, the wind tunnel blockage effect can be neglected when the total blockage ratio of a given model is less than 5%.

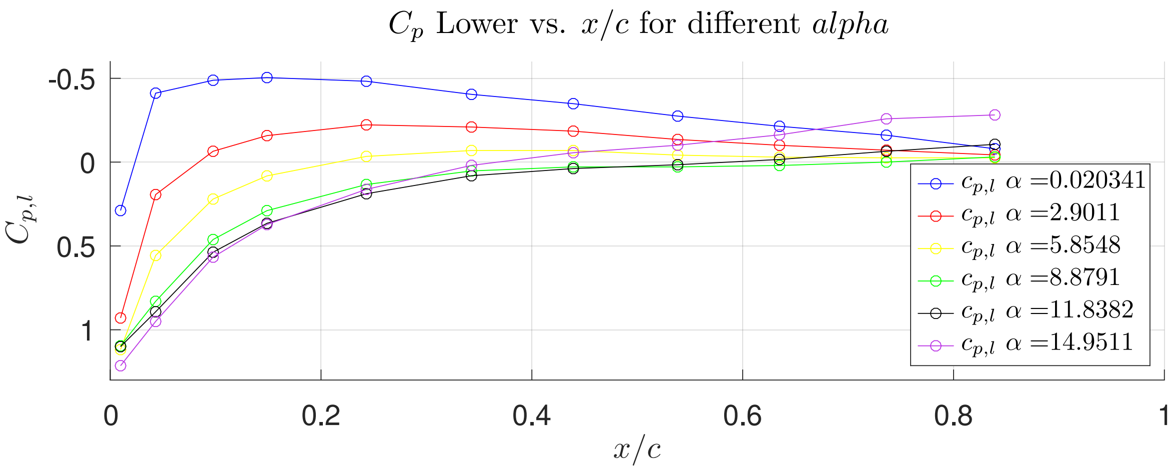
For all the results, ε was calculated and was found to be less than 5% except for the data of $\alpha = 15^\circ$, so the corrections were made using equations in [1], by first defining:

$$\sigma = \frac{1}{48}\left(\frac{\pi c}{h}\right)^2, \qquad \varepsilon = \varepsilon_{sb} + \varepsilon_{wb} = \frac{K_1 V}{A^{\frac{3}{2}}} + \frac{c}{2h} c_{du} \tag{7}$$

where A and h are the cross-sectional area and height of the wind tunnel test section, respectively, K_1 is a constant (for an aerofoil mounted across the entire width of a wind tunnel test section $K_1 = 0.76$), V is the volume of the aerofoil model (for an aerofoil with a chord length of c, a maximum thickness of t and a span of b, $V = 0.7tcb$), then:

$$C_l = C_{lu}(1 - \sigma - 2\varepsilon), \quad C_d = C_{du}(1 - 3\varepsilon_{sb} - 2\varepsilon_{wb}) \tag{8}$$


C_p Upper vs. x/c for different $alpha$
$C_{p,u}$
x/c
$c_{p,u}$ α =0.020341
$c_{p,u}$ α =2.9011
$c_{p,u}$ α =5.8548
$c_{p,u}$ α =8.8791
$c_{p,u}$ α =11.8382
$c_{p,u}$ α =14.9511


C_p Lower vs. x/c for different $alpha$
$C_{p,l}$
x/c
$c_{p,l}$ α =0.020341
$c_{p,l}$ α =2.9011
$c_{p,l}$ α =5.8548
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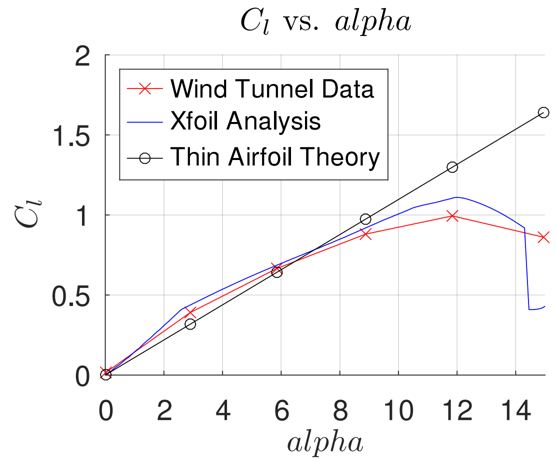
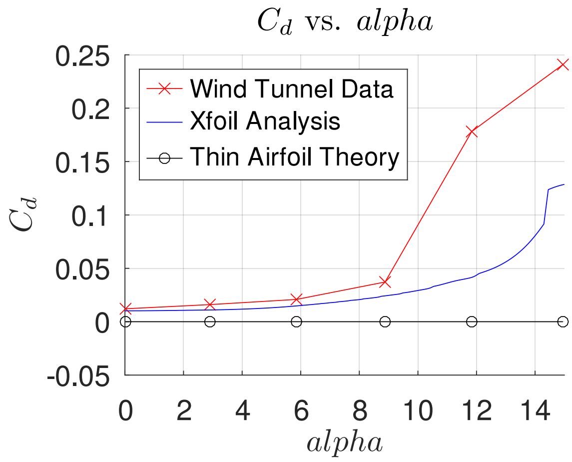
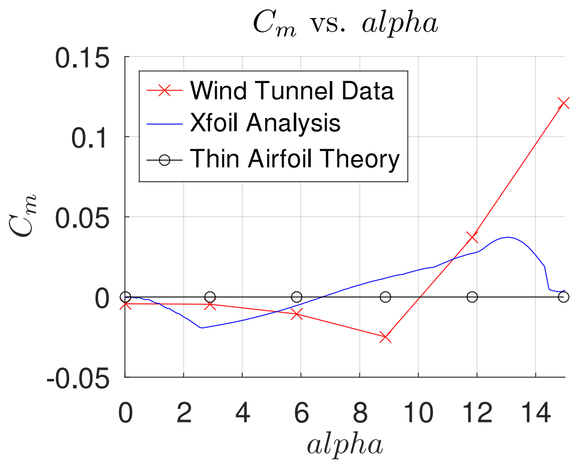
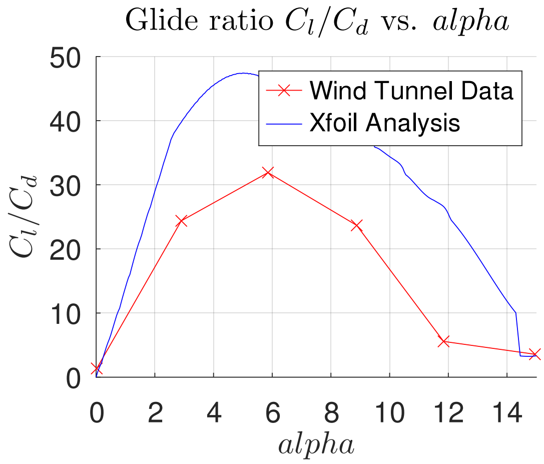

C_l vs. $alpha$
C_l
$alpha$
Wind Tunnel Data
Xfoil Analysis
Thin Airfoil Theory


C_d vs. $alpha$
C_d
$alpha$
Wind Tunnel Data
Xfoil Analysis
Thin Airfoil Theory


C_m vs. $alpha$
C_m
$alpha$
Wind Tunnel Data
Xfoil Analysis
Thin Airfoil Theory


Glide ratio C_l/C_d vs. $alpha$
C_l/C_d
$alpha$
Wind Tunnel Data
Xfoil Analysis

Results

I found it more convenient to conduct my own XFoil analysis, I did it through XFLR5 software, I set the Reynold's number to $Re = \frac{\rho_\infty v_\infty c}{\mu_\infty} \simeq 200\,000$ and used 150 panels for the airfoil.

Those are some of the results obtained presented in the form of numbered points, discussing some of the not so satisfying results in this report, apparently because of the measurements uncertainty and external factors:

1. Many points on the lower airfoil surface have read a local static pressure greater than the total freestream pressure, hence, the local coefficient of pressure has become > 1 at these points.
2. Many points on the rake had read static pressure more than the total freestream pressure, to overcome this, I took the total pressure to be the pressure at the furthest point in the rake $P_T = \max\left[P_{rake}\left(x = -60\right), P_{rake}\left(x = +60\right)\right]$, but surprisingly, many points in the middle of the rake also read static pressure more than the one read at the furthest points in the rake, I could do nothing to solve this.
3. The values of upper surface pressure at $\alpha = 15^\circ$ are so wrong at the trailing edge, they are too high, so C_p doesn't meet at all at the trailing edge between upper and lower surfaces.
4. The pitching moment behavior against change in α obtained from experiment was similar enough to that obtained from XFoil, however, $C_{m,c/4}$ is equal to zero in irrespective of the angle of attack in the thin airfoil theory.
5. Thin airfoil theory does not capture drag at all, because it is a viscous theory.
6. When setting x_{ref} to $0.24c$ or $0.23c$ to calculate C_m from experimental data, the result of C_m was better and closer to become zero than C_m obtained when $x_{ref} = 0.25c$.

4 Conclusions

In this report, we processed the data for the airfoil and achieved the aerodynamic coefficients C_l, C_d, and C_m, we compared the results with those from XFLR5 (uses XFoil analysis for airfoils) and compared both to results from thin airfoil theory, the results were generally satisfying because, the coefficients behavior was somehow relevant. Only some differences between the three methods come from the uncertainty of the measurements, to solve this in the future, more advanced equipment should be used such as digital manometers and so on to obtain better results, but however, some differences come from the nature of the method used (thin airfoil theory is inviscid, so there's no drag, and aerodynamic center always exists at $c/4$).

Author Contribution

Project was done by member no. 1 in team 38.

References

[1] Dr Nabawy M. Lab Handout; 2023.

[2] Dr Nabawy M. Lecture Notes on Incompressible Aerodynamics; 2023.

[3] Khan Academy. Understanding the Trapezoidal Rule;. Available from: https://www.khanacademy.org/math/ap-calculus-ab/ab-integration-new/ab-6-2/a/understanding-the-trapezoid-rule.