UNIVERSITY OF SHEFFIELD

AER325 AERODYNAMIC DESIGN

THE EFFECT OF AEROFOIL DESIGN ON AIRCRAFT PERFORMANCE

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1 NOMENCLATURE

C_L	3D Lift Coefficient	α	Angle of Attack (°)
C_D	3D Drag Coefficient	ρ	Air Density (kg/m^3)
c_l	2D Lift Coefficient	b	Wing Span (m)
c_d	2D Drag Coefficient	AR	Aspect Ratio
L	Lift Force (N)	R_e	Reynolds Number
S	Wing Surface Area (m^2)	c	Chord (m)
D	Drag Force (N)	v	Kinematic Viscosity (m^2/s)
V	Air Velocity (m/s)	C_{Di}	Induced Drag
e	Span efficiency factor		

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2 INTRODUCTION

The aim of this study is to investigate the effect the aerofoil of a wing has on the performance of a given aircraft. In order to ensure the aerofoil design is the only variable under investigation, the aspect ratio and wing area remain constant throughout the study. Xfoil was used to predict the aerodynamic performance of various different aerofoil sections, which were then compared to a given datum aerofoil in order to select an improved aerofoil design. As aircraft performance can have a huge variety of factors, this study focussed on the performance at cruise. Coefficients of lift and drag were recorded at varying angles of attack, from which drag polars and cl/cd v alpha graphs were drawn and analysed.

3 XFOIL

Xfoil is a programme which was used in this study for viscous analysis of existing aerofoils [1]. In order to calculate the pressure distribution on an aerofoil, hence the lift and drag characteristics, Xfoil needs to be given the coordinates specifying the shape of the 2D aerofoil and the Reynolds number. In this study only NACA four-digit aerofoils were investigated, therefore the shape can be specified by the maximum camber, maximum camber position and thickness-to-chord ratio, represented by the first, second and last two digits respectively.

The Reynolds number can be calculated using the data given of the Cessna 172 aircraft, as shown in table 1, converting cruise speed into m/s for air velocity and finding the kinematic viscosity of air at 1000ft using interpolation. Chord length can be calculated using equations 1, 2 and 3, and put into equation 4 to find the Reynolds number.

$$AR = \frac{b}{c} = \frac{b}{c} x \frac{b}{b} = \frac{b^2}{S} \qquad (1)$$

$$b = \sqrt{AR \ x \ S} = \sqrt{4.2 \ x \ 28.64} = 10.968 \qquad (2)$$

$$c = \frac{b}{AR} = \frac{10.968}{4.2} = 2.611 \qquad (3)$$

$$R_e = \frac{Vc}{v} = \frac{51.44 \ x \ 2.611}{1.497x10^{-5}} = 8.97x10^6 \qquad (4)$$

Table 1: Table of aircraft and flight specifications

Aircraft	Cessna 172
Aircraft mass	1061 kg
Cruise speed	100 knots
Datum aerofoil	NACA 1412
Datum wing Aspect Ratio	4.2
Total wing planform area:	28.64 m^2
Span efficiency factor:	0.9
Altitude:	1000 ft

4 AEROFOIL ANALYSIS

As NACA four-digit aerofoils were used in this investigation, the geometric parameters (max camber, max camber position and thickness) are therefore the variables in this study. In order to compare the effects of each parameter, three separate studies were carried out. In each study a different geometric parameter would vary while the other two would be fixed. By running each of these aerofoils in Xfoil at angles of attacks between 20° and -10°, coefficients of lift and drag were recorded for each one. Using these values graphs of cl v cd and cl/cd v alpha were drawn.

As this study is focused on the aircraft's performance at cruise, while analysing the drag polars it is most important to find the aerofoil with the lowest drag coefficient at cruise, as the lift coefficient at cruise will be constant and can be calculated using equations 5 and 6. As the aircraft is at cruise, lift is equal to weight and therefore can be taken as mass multiplied by gravity. Due to 3D effects the wing C_L calculated in equation 5 is 90% of the 2D aerofoil c_l in Xfoil, hence the need to divide it by 0.9, as shown in equation 6, to find the correct coefficient of lift for the Xfoil data.

$$C_L = \frac{2L}{\rho V^2 S} = \frac{2\ x\ 10408.41}{1.19\ x\ 51.44^2\ x\ 28.64} = 0.23 \qquad (5) \qquad\qquad c_l = \frac{0.23}{0.9} = 0.26 \qquad (6)$$

4.1 Maximum Camber

Camber is the difference between the mean camber line and the chord line, essentially how curved the aerofoil is. Upon research prior to the study, it was found that the lift of a given aerofoil at a given angle of attack is largely determined by its camber [2]. It is known that lift is produced by a difference in pressure below and above the wing, and that this difference in air pressure is due to the air moving faster over the top of the wing. Camber increases this effect, therefore increases lift.

Camber is represented as a percentage of the chord and can vary between 0 and 9.5. In this study maximum cambers of 1,2,3,4 and 5% were tested with a constant camber position of 40% and thickness of 14%. As shown in figure 1, the increased camber does increase lift as discussed earlier from prior research. Although increasing camber increases lift, it also decreases the aircrafts stall angle, therefore having a massive effect on aircraft performance. From figure 1 it can be seen that NACA2414 had an increased lift-drag ratio than the datum aerofoil however did not lower the stall angle by much. From this the conclusion can be drawn that an improved maximum camber is 2%.

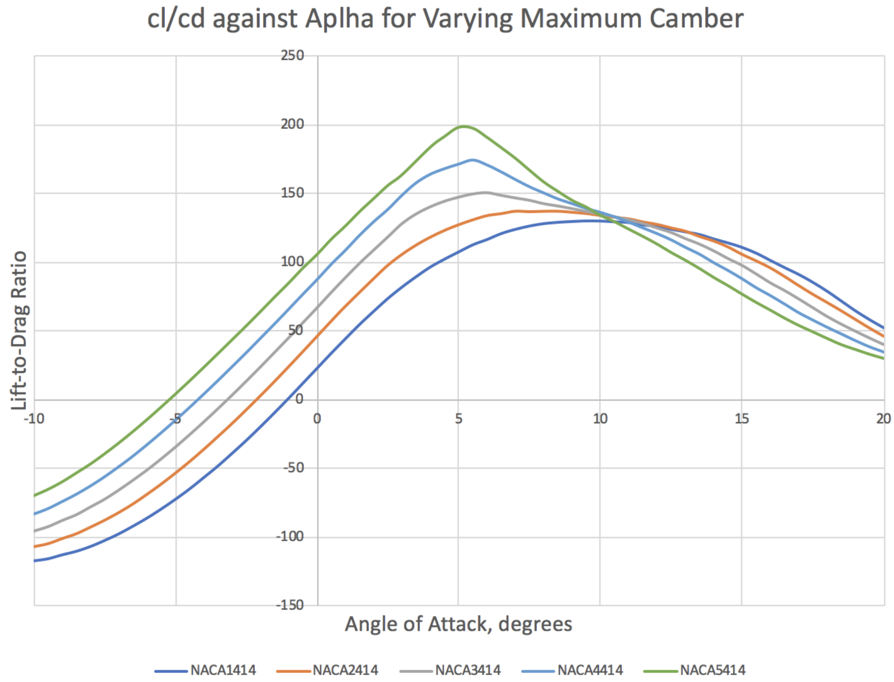

Figure 1: Graph showing lift-drag ratio vs angle of attack for varying maximum camber.

4.2 Maximum Camber Position

Maximum camber position is a percentage of the chord length between 0 and 90 representing where the maximum camber lies along the chord. In order to investigate the effect, it has on aircraft performance, maximum camber position varied from 0-8 (0-80%) in increments of 2, while thickness was fixed at 14% and maximum camber at 2%, except in the case where camber position was 0 because this meant there was no camber therefore maximum camber was also 0.

From figure 2 it can be interpreted that an aerofoil with maximum camber position at 80% may perform better than one at 40% as the lift-drag ratio is higher for most of the graph and does not compromise the stall angle much. Also looking at figure 3, at the cruise coefficient of lift, 0.26, the drag coefficient is slightly lower at camber position 80%, than at 40%. Therefore, the aircraft will have greater lift.

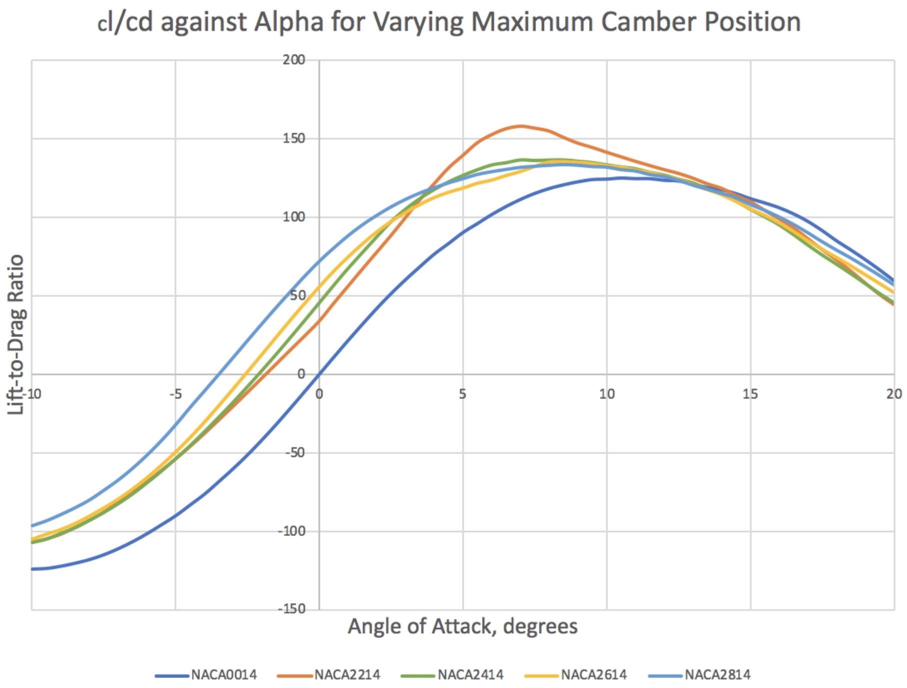

Figure 2: Graph showing lift-drag ratio vs angle of attack for varying maximum camber position.

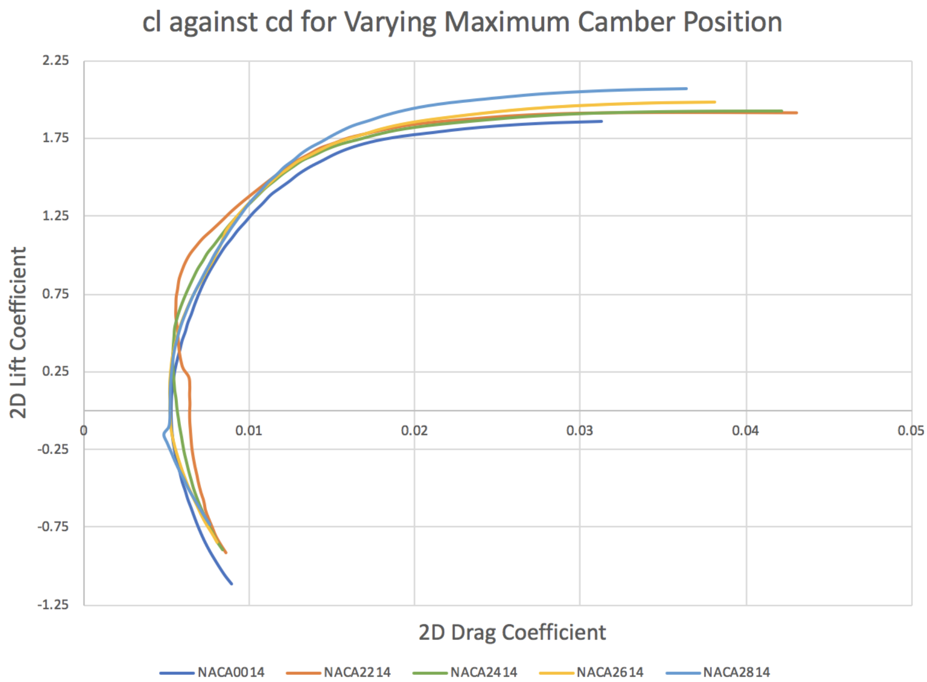

Figure 3: Graph showing cl v cd for varying maximum camber position.

4.3 Thickness

Thickness refers to the maximum thickness of the aerofoil and is a percentage of thickness-to-chord ratio. In order investigate the effect thickness has on aircraft performance, maximum camber was fixed at 10% and maximum camber position at 40%, while thickness varied from 8-24% in increments of 4. From figure 4 it can be seen that the dark blue line, representing NACA1408, has the lowest drag coefficient at cl 0.26, therefore suggesting an aerofoil of thickness of 8% will perform better at cruise than one the datum aerofoil with a thickness of 12%.

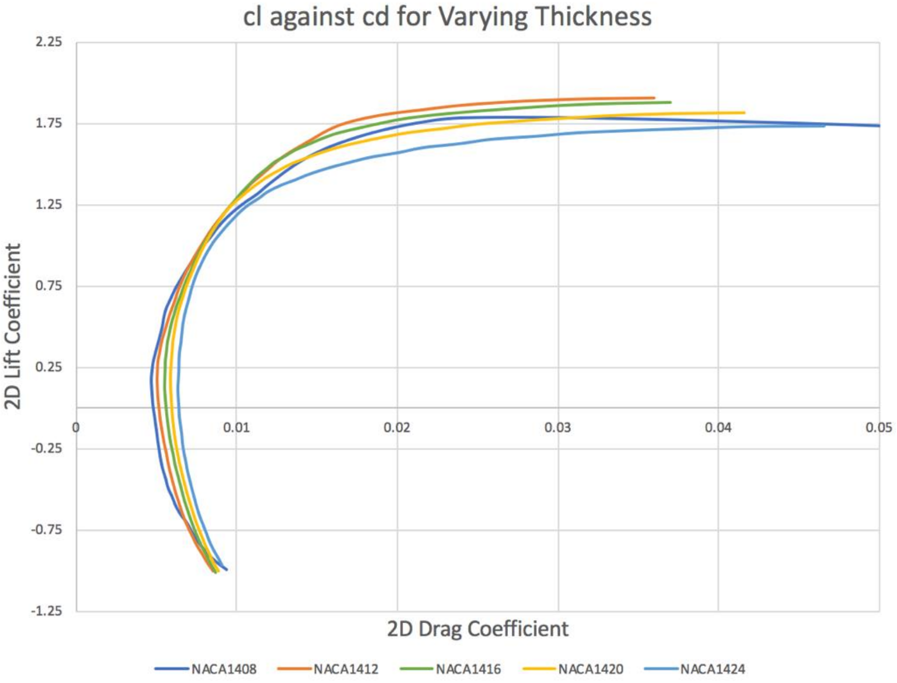

Figure 4: Graph showing cl v cd for varying thickness.

5 IMPROVED AEROFOIL SELECTION

From the graphs it could be interpreted that an improved aerofoil at cruise would have a maximum camber of 2%, a maximum camber position of 80% and a thickness of 8%, therefore a NACA2808 aerofoil. As mentioned earlier, Xfoil analyses just the aerofoil therefore works in 2D, whereas a wing in real life is 3D. Therefore, in order to best compare the datum aerofoil with the improved aerofoil some calculations must be carried out in order to find the 3D lift and drag coefficients and from these the 3D lift-drag ratio.

As mentioned previously 3D wing lift coefficient C_L is 90% of the 2D aerofoil lift coefficient c_l, therefore to find 3D C_L, the c_l values produced by Xfoil must be multiplied by 0.9. As Xfoil does not deal with induced drag, in order to find 3D wing, drag coefficient, equation 7 must be used and then added to the Xfoil c_d values, as shown in equation 8. Where e is the span efficiency factor and is equal to 0.9.

$$C_{Di} = \frac{C_L{}^2}{\pi e AR} \quad (7) \qquad\qquad C_D = c_d + C_{Di} \quad (8)$$

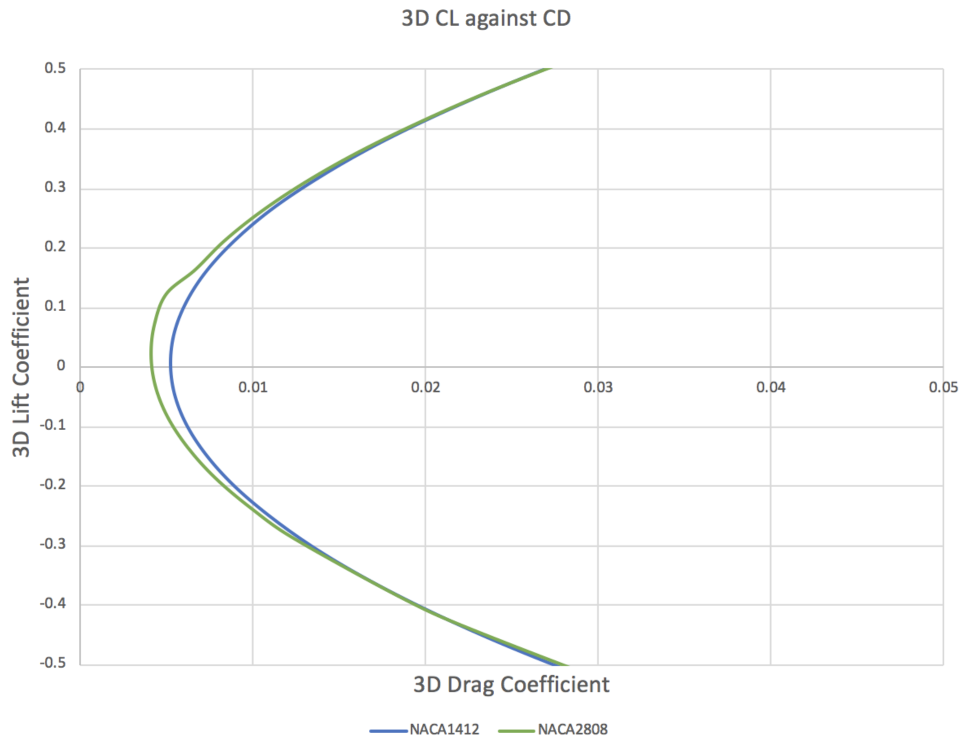

Figure 5: Graph showing cl v cd for datum aerofoil NACA1412 and improved aerofoil NACA2808.

From figure 5 it can be seen that the selected aerofoil does have a slightly lower drag coefficient at cruise C_L, 0.23, therefore can be said to have a better aircraft performance at cruise.

6 CONCLUSION

From gathering data of various different aerofoils using Xfoil and plotting polar drags and cl/cd v alpha graphs the following conclusions can be drawn;

- Maximum camber has the most significant effect on aircraft performance out of the three parameters.
- A maximum camber of 2% shows the best performance at the cruise in this study.
- A maximum camber position of 80% shows an improved performance at cruise than the datum aerofoil.
- The thicker the aerofoil the greater the drag coefficient at cruise, therefore a thickness of 8% was found to show the best performance in this study.

7 REFERENCES

[1] XFOIL, *Subsonic Airfoil Development System*, [Reviewed November 2019] https://web.mit.edu/drela/Public/web/xfoil/

[2] Aircraft Owners and Pilots Association, *CAMBER IS THE KEY: UNDERSTANDING BASIC AERODYNAMICS*, [Reviewed November 2019] https://www.aopa.org/news-and-media/all-news/2004/november/flight-training-magazine/camber-is-the-key