Tests and numerical simulation of aerodynamic characteristics of airfoils for general aviation applications

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Abstract: This paper was to validate the effects of airfoil thickness ratio on the characteristics of a family of airfoils. Research was carried out in different ways. First, tests were conducted in the wind tunnel. And numerical simulation was performed on the basis of tests. Results from calculation were consistent with tests, indicating that numerical method could help evaluate characteristics of airfoils. Then the results were confirmed by compared with empirical data. The study also showed that the determining factor of lift is not only the thickness ratio, but the angle of attack, the relative camber and the camber line. The thickness ratio appears to have little effect on lift coefficient at zero angle of attack, since the angle of zero lift is largely determined by the airfoil camber. According to the research, numerical simulation can be used to determine the aerodynamic characteristics of airfoils in different environment such as in the dusty or humid air.

Key words: aerodynamic characteristics; numerical simulation; airfoil; thickness ratio

1 Introduction

The main parameters that influence lift coefficient of airfoil are airfoil geometry, surface condition (i. e. smooth or rough), angle of attack, Reynolds number (Re) and Mach number (Ma). In this paper, effects of airfoil geometry on the aerodynamic characteristics of an initial low-speed family of airfoils were investigated.

This paper presented results from tests and computational simulation of the aerodynamic characteristics of a low-speed family of airfoils. Then the results of the two methods were compared to validate the precision of computational simulation. Models for the tests were 13 %, 17 % and 21 % thick airfoil sections designed for general aviation applications[1]. The tests were conducted at a Ma of about 0.06 and Re about 5.5×10^5. The angle of attack varied from $-20°$ to $30°$. And simulation was performed with CFX.

2 Wind tunnel test

2.1 Wind tunnel and models definition

The size of wind tunnel test chamber section was 1.20 m × 0.80 m, with the length of 2.00 m.

This airfoil family was obtained by linearly scaling the mean thickness of the 17 % computer-designed airfoil[2,3]. Therefore, all three airfoils had the same camber distribution. The airfoil section profiles and the mean camber line were shown in Fig. 1. Where x is the airfoil chord coordinate, z is the airfoil thickness coordinate and c is the length of airfoil chord.

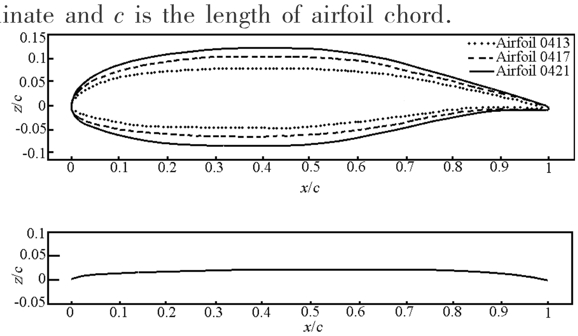

Fig. 1 Airfoil section profiles and mean camber line

2.2 Wind tunnel test results

The LS_0413, LS_0417 and LS_0421 airfoils were tested at a Ma of about 0.06 and Re about 5.5×10^5. The angles of attack were $-20°$, $-17.5°$, $-15°$, $-10°$, $-5°$, $0°$, $5°$, $10°$, $15°$, $20°$, $22.5°$, $25°$, $27.5°$, and $30°$. The following figure was the lift coefficient of the three airfoils with different angles of

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attack α.

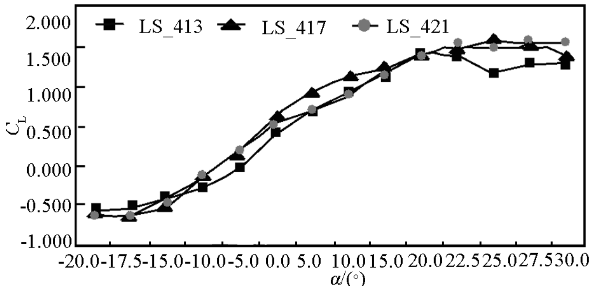

Fig. 2 Lift coefficient for three airfoils.

Fig. 2 showed that the lift coefficient (C_L) increased with the angle of attack α. It also indicated that the determination of lift was not the thickness ratio, since the values of lift coefficient of three airfoils were similar. Ref. [1] showed that the determining factors were angle of attack, the relative camber and camber line. The thickness ratio appeared to have little effect on lift coefficient at zero angle of attack, since the angle of zero lift was largely determined by the airfoil cambe0r.

3 Calculation

3.1 Numerical simulation

Computational fluid dynamics (CFD) method has become an indispensable tool for designing an airplane. The CFD software CFX was used in this research to evaluate the characteristics of the airfoils. The governing equations are the Reynolds averaged 3-D incompressible Navier-Stokes (RANS) equations and the continuity equation[4,5]:

$$\frac{\partial u_i}{\partial t} = -u_j\frac{\partial u_i}{\partial x_j} - \frac{1}{\rho}\frac{\partial p}{\partial x_i} + \nu\frac{\partial^2 u_i}{\partial x_j \partial x_j} \qquad (1)$$

$$\frac{\partial u_i}{\partial x_i} = 0 \qquad (2)$$

Where, $\boldsymbol{u}$, p, ρ and ν represent components of velocity vector, pressure, density and kinematic viscosity, respectively.

The $k-\varepsilon$ turbulence model with integration to the wall and pressure gradient effect was employed. The $k-\varepsilon$ model was selected due to its capability of taking into account of turbulent boundary layer history effect by solving the complete transport equations of k and ε, and the $k-\varepsilon$ model was more capable than algebraic models to predict the separated flows, which occurred when the airfoil stalled at high angle of attack. The transport equations for the standard $k-\varepsilon$ model are:

$$\frac{\partial(\rho k)}{\partial t} + \nabla\cdot(\rho U k) = \nabla\cdot\left[\left(\mu + \frac{\mu_t}{\sigma_k}\right)\nabla k\right] + P_k - \rho\varepsilon \qquad (3)$$

$$\frac{\partial(\rho\varepsilon)}{\partial t} + \nabla\cdot(\rho U \varepsilon) = \nabla\cdot\left[\left(\mu + \frac{\mu_t}{\sigma_\varepsilon}\right)\nabla \varepsilon\right] + \frac{\varepsilon}{k}(C_{\varepsilon 1}P_k - C_{\varepsilon 2}\rho\varepsilon) \qquad (4)$$

Where, μ_t is the turbulence viscosity, $C_{\varepsilon 1}$, $C_{\varepsilon 2}$, σ_k and σ_ε are constants, and P_k is the turbulence production due to viscous and buoyancy forces.

The full turbulent boundary layer assumption was used and was consistent with the tripped boundary layer in the tests. Mesh refinement study was conducted for a few selected points to ensure that the solutions were mesh size independent. Since the CFD solutions were obtained from the steady state calculations based on RANS model, the unsteady details of the shear layer mixing entrainment and large coherent vortex structures were not able to be captured.

The total pressure, total temperature and velocity components were set as the inlet boundary conditions. And the static pressure was set as the outlet boundary condition, the periodic interface as the span-wise boundary condition.

According to Fig. 3 and Fig. 4, it was shown that the chord-wise pressure coefficient between tests and simulation was similar but not totally consistent. Since the accuracy of computational simulation was correlative to the computational method and the quality of mesh. With the following equation the mean lift coefficient along the chord was obtained:

$$C_L = \int_0^1 (C_{p\text{-lower}} - C_{p\text{-upper}})\cos\alpha\, d\left(\frac{x}{c}\right) \qquad (5)$$

Where, C_L and C_p is lift coefficient and pressure coefficient respectively.

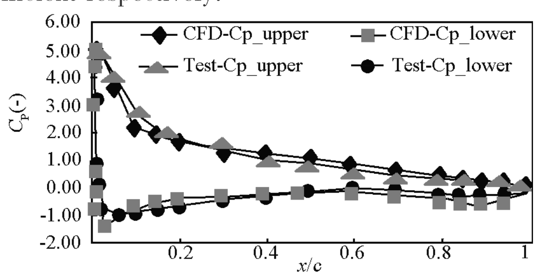

Fig. 3 Comparison of pressure coefficient between tests and simulation (LS_0421, α = 20°) *

Then the values of mean lift coefficient from tests and simulation were 1.416 and 1.498 respectively, which were quite similar.

* According to aerodynamics theory, upper surface C_p of airfoil is negative and lower surface is positive. C_p(-) means the opposite of C_p, which was used in order to put upper surface pressure coefficient curve in the upside and lower surface curve downside.

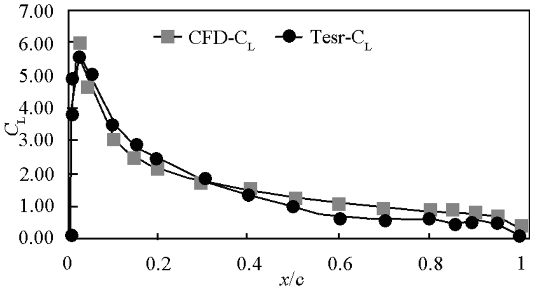

Fig. 4 Comparison of lift coefficient between tests and simulation (LS_0421, α =20°)

Comparisons of mean lift coefficient resulted from the test and simulation were shown in Table 1. Most results from simulation agreed with those from the test, and the relative errors were within 15 %. However, with the increase of thickness, it was more difficult to get the reasonable result. Sometimes several turbulence models had to be changed before the calculation became convergent.

Table 1 Comparison of mean lift coefficient resulted from test and simulation

α/(°)	LS_0413		LS_0417		LS_0421	
	Test	CFD	Test	CFD	Test	CFD
-20.0	-0.533	-0.668	-0.608	-0.663	-0.610	-0.840
-17.5	-0.531	-0.571	-0.635	-0.714	-0.630	-0.778
-15.0	-0.403	-0.474	-0.393	-0.395	-0.507	-0.662
-10.0	-0.256	-0.232	-0.131	-0.186	-0.118	-0.152
-5.0	-0.044	0.054	0.173	0.114	0.189	0.116
0.0	0.433	0.437	0.626	0.455	0.544	0.451
5.0	0.681	0.799	0.919	1.000	0.675	0.822
10.0	0.928	1.122	1.131	1.058	0.900	0.938
15.0	1.160	1.169	1.226	1.328	1.196	1.313
20.0	1.421	1.150	1.417	1.496	1.416	1.498
22.5	1.416	1.275	1.512	1.414	1.526	1.517
25.0	1.168	1.312	1.573	1.530	1.480	1.509
27.5	1.286	1.353	1.543	1.482	1.537	1.458
30.0	1.312	1.410	1.402	1.328	1.540	1.305

3.2 Comparison with empirical data

To check the accuracy of tests, analysis about the maximum lift coefficient C_{Lm} was performed according to the theory in Ref. [4]. The maximum lift coefficient was presented in terms of incremental lift coefficient, ΔC_L, added to the lift coefficient at zero angle of attack, C_{L0}. The models of test could be supposed as a smooth aerofoil, since the small band of minimum roughness was sufficient to fix boundary-layer transition. And this was the common practice for present wind-tunnel test. On the basis of this hypothesis, the quantity of C_{L0} and ΔC_L were calculated respectively.

$$C_{Lm} = (C_{L0} + \Delta C_L) F_S F_M \quad (6)$$

Where F_S is the factor on C_{Lm} for modern airfoils; F_M is the factor for effect of Mach number on C_{Lm} [6].

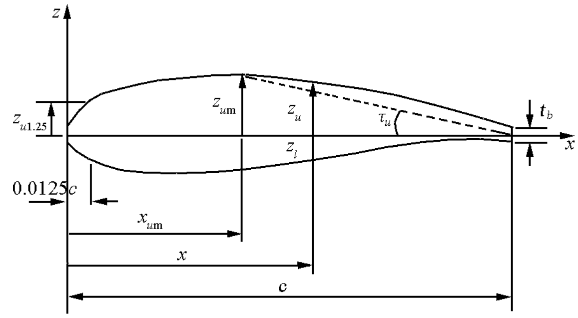

Fig. 5 Aerofoil geometry

The coefficient C_{L0} was obtained by combining the lift-curve slope, $(a_1)_0$, for incompressible flow with the zero-lift angle, α_0.

$$\alpha_0 = -\frac{\pi}{90}\sum_{i=1}^{14}(B_i z_{ci}/c) \quad (7)$$

Where, B_i is coefficient used in estimation of α_0 in Ref. [6], and $z_{ci} = [z_u(x_i/c) + z_l(x_i/c)]/2$. The meaning of other parameters were shown in Fig. 5.

With the values of coefficient B_i [6], α_0 was got and then, the lift coefficient at zero angle of attack was given by

$$C_{L0} = -\alpha_0 (a_1)_0 \quad (8)$$

The coefficient ΔC_L was got according to the profile curve of the airfoil. Firstly the quantity of $z_{u1.25}/c$ was calculated, for airfoils with $z_{u1.25}/c \leqslant 0.017$, ΔC_L was a function of $z_{u1.25}$ and Reynolds number Re. For $z_{u1.25}/c \geqslant 0.017$, ΔC_L was a function of $\tan\tau_u$ and Re, and $\tan\tau_u$ is given by

$$\tan\tau_u = \frac{(z_{um}/c)}{(1-x_{um}/c)} \quad (9)$$

Take the airfoil LS_0417 as an example, with the coordinate of airfoils[1] and lift coefficient curve from the test, the values of α_0 and $(a_1)_0$ were got. Then the lift coefficient at zero angle of attack C_{L0} was calculated as 0.604. The coefficient ΔC_L was got as 0.795. Therefore, C_{Lm} = 1.40.

From the results of tests, C_{Lm} was 1.57, and the relative error was 10.8 %. The accuracy of the test was confirmed. But there were still some difference between tests and empirical data because the surface of airfoil model was not smooth and was not totally in accord with empirical method.

4 Conclusions

Aerodynamic characteristics of an initial low-speed family of airfoils for general aviation applications were investigated at a Mach number of about 0.06 and Reynolds number about 5.5×10^5. Wind tunnel test was carried out to find the difference among a series of airfoils. Then CFD simulation was conducted for 3D airfoils with $k-\varepsilon$ turbulence model. With the results of test and computational simulation, some conclusions were obtained.

Firstly, computational simulation was further confirmed by the test. The lift curve from CFD simulation agreed well with the test at low angle of attack. The increase of lift coefficient with angle of attack was effectively linear until the onset of flow separation at a particular angle. This was dependent on the airfoil geometry profile and the free-stream conditions. Further increase in angle of attack resulted in greater extents of flow separation and reduction in the slope of the lift coefficient curve until the lift coefficient reached the maximum value, C_{Lm}, and the airfoil stalled.

Secondly, the accuracy of test was validated by compared with empirical data. The maximum lift coefficients from the test and empirical method were similar, 1.57 and 1.40 respectively (taking the LS_0417 airfoil as an example), and the relative error was 10.8 %. Since the maximum lift coefficient was dependent on the lift coefficient at zero angle of attack, C_{L0}, and incremental lift coefficient, ΔC_L, it was important to find the characteristics of the two parameters. The angle of zero lift was largely determined by the airfoil camber, and C_{L0} was determined by the angle of zero lift and slope of lift curve.

Finally, since the results of numerical simulation were consistent with test, and the low-speed family of airfoils were for general aviation applications, the outside conditions can be changed (such as in the dusty or humid air) to estimate the aerodynamic characteristics in other environment. It will be helpful to improve the characteristics of airfoils and develop new aircraft airfoils.

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