Characteristics of a typical lifting symmetric supercritical airfoil[†]

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Abstract. The theoretical aerodynamic characteristics of a typical lifting symmetric supercritical airfoil demonstrating its superiority over the NACA 0012 airfoil from which it was derived are presented in this paper. Further, limited experimental results confirming the theoretical inference are also presented.

Keywords. Transonic flow; supercritical airfoil; aerodynamic characteristics.

1. Introduction

All lifting supercritical airfoils, to the best knowledge of the author, have camber which gives wave drag at supersonic speeds. Hence these supercritical airfoils are unsuitable for use in supersonic aircraft. It has been reported that supercritical airfoils were considered for Y.F. 16 and Y.F. 17, but were discarded for the above reason. Also camber gives excessive pitching moment, an undesirable characteristic for helicopter rotor blades. Ramaswamy (1976) suggested that it should be possible to design symmetric airfoils which under design lift and incidence conditions are nearly shock-free. Such lifting symmetric supercritical airfoils would eliminate the disadvantages mentioned earlier. The feasibility of designing such lifting symmetric supercritical airfoils has been demonstrated by Nandan & Ramaswamy (1985).

In this paper, a brief outline of the method used in designing such airfoils is illustrated. Then the theoretical results for the aerodynamic characteristics of a typical airfoil derived from modifying the NACA 0012 are presented to demonstrate the superiority of the modified airfoil over the original one. Finally, limited experimental data obtained on NACA 0012 and the modified airfoil are presented to confirm the theoretical conclusions.

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2. Outline of the method

The method used is essentially the design method of Sobieczky *et al* (1979), called the fictitious gas method. This method is used for modifying a given profile to make it shock-free. The method needs a reliable potential code for computing the transonic flow past the profile to be modified. The code used by Sobieczky is the one developed by Bauer *et al* (1975, the BGKJ code) and the same with the non-conservative option is adopted here. The code solves the potential equation

$$(a^2 - U^2)\phi_{xx} - 2UV\phi_{xy} + (a^2 - V^2)\phi_{yy} = 0, \quad (1)$$

with the relation for density

$$\rho/\rho_\infty = \{1 + [(\gamma - 1)/2] M_\infty^2 (1 - U^2 - V^2)\}, \quad (2)$$

where U and V are the velocity components in x and y directions, ϕ is the potential function, M_∞ , the free stream Mach number, a is the local speed of sound. Without loss of generality, U_∞ is taken to be unity.

Consider a profile to be modified to make it shock-free at a particular M_∞ and incidence α . Then analyse this profile using the potential code with the modification that when the total velocity q in the flow field exceeds a^* , the critical speed of sound, replace (2) by a fictitious gas law,

$$\rho/\rho^* = (a^*/q)^p; \quad p < 1 \dots \text{for } q > a^*, \quad (3)$$

which makes the differential equation elliptic even in the supersonic region $q > a^*$. For instance, when $p = 0$, $\rho = \rho^*$ and this incompressible flow in the region $q > a^*$. Therefore as shown in figure 1, the flow pattern over the airfoil profile will have no shocks and the flow in the region between the sonic line and the airfoil boundary is fictitious. But the flow on the sonic boundary and everywhere else outside it is correct.

In order to make the flow in this region also correct corresponding to real gases, the flow angle and the sonic velocity on the sonic boundary are taken as initial conditions, and this region is solved for a real gas using the characteristic method. This solution to the real flow field would generate a new surface on the airfoil between the sonic points so as to be compatible with the real gas flow and would also generate the real pressure distribution on this modified contour. The modified airfoil is re-analysed using the potential code, with real gas law everywhere, to confirm shock-free flow. If wiggles in the pressure distribution are noticed near the sonic points, then the contours are locally smoothed in those regions to remove the wiggles. Complete details of this method are given by Sobieczky *et al* (1979). As pointed out by him, there is no guarantee that the characteristic solution does not end up with limit lines. However, it has been found in practice that with a suitable choice of p , this problem can be overcome in most of the cases.

To design a symmetric lifting supercritical airfoil, we take a baseline airfoil which is symmetric. We choose an M_∞ and α such that there is moderate shock on the top surface and no shock on the lower surface. Using the method described above, the top surface is modified to make it shock-free. Then the bottom surface is made symmetrical with the top surface to get a symmetrical airfoil. This airfoil is analysed to confirm that the shock-free characteristic is preserved. Thus a lifting symmetric supercritical airfoil is obtained.

Figure 1. Principle of shock-free airfoil design by fictitious gas method. (a) Flow over basic airfoil with real gas flow; (b) flow over basic airfoil at same M and α but with fictitious gas law in supersonic region; (c) modification to contour to maintain shock-free flow.

The basis of our thinking that the shock-free characteristic of the flow on the upper surface would be maintained even after the bottom surface was modified to make it symmetrical was the following. There was evidence that when supercritical airfoils for transport aircraft were modified as shown in figure 2 to provide increased lift through aft camber, the pressure distribution on the upper surface was hardly affected. Our experience has also shown that shock-free characteristic on the upper surface was maintained even after the bottom surface was modified to make it symmetrical with the top surface.

Figure 2. Example illustrating modification to lower contour without affecting shock-free flow on top surface.

3. Theoretical characteristics of NAL-114-36-00 airfoil

This airfoil has been obtained by modifying the NACA 0012 airfoil. It is 11.4% thick, with maximum thickness at 36% chord; and zero camber. To arrive at this airfoil, the flow past NACA 0012 airfoil at $M_\infty = 0.752$ and incidence equal to 1.1 degrees was considered. The pressure distribution obtained using the BGKJ code is shown in figure 3. Reasonably strong shock can be noted. As described in the previous section, the top surface of this airfoil was modified to make it shock-free and the bottom surface was made symmetrical with the top one to make the airfoil as a whole symmetrical. Comparison of the contour of NAL-114-36-00 with that of NACA 0012 is shown in figure 4. The pressure distribution obtained on the NAL 114-36-00 airfoil at the same M_∞ and α condition is shown in figure 5. The shock-free nature of the pressure distribution may be noted. The corresponding pressure distribution on NACA 0012 is also shown for comparison. Figure 6 shows the pressure distributions under slightly off design incidence conditions, whereas figure 7 shows the pressure distributions under slightly off design Mach number conditions. From these figures, it may be seen that no abrupt changes take place