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Transonic airfoil thickness variation requirements for maintaining shock-free flow

Helmut Sobieczky

Deutsche Forschungs und Fersuchanstalt, Theoretische Stroemungsmechanik
D-3400 Goettingen, F. R. Germany
and

George S. Dulikravich

The Pennsylvania State University, Department of Aerospace Engineering
233 Hammond Building, University Park, PA 16802, USA

ABSTRACT

The volume of certain types of smart materials proposed for coating of aerodynamic surfaces can be instantaneously affected if subjected to an electric or a magnetic field, thus creating dramatic changes in the entire flow field. For example, a transonic airfoil that was designed to be shock-free at a given angle of attack and a given flight Mach number will start developing shock waves immediately after either the angle of attack or the flight Mach number are perturbed. In order to maintain the same aerodynamic performance of an airfoil (that is, maintain the identical distribution of surface pressures on the airfoil) over a range of flight speeds, the airfoil shape must readjust continuously. Using our highly accurate and proven transonic airfoil design code we have performed a detailed evaluation of the required local thickness alterations necessary to maintain the shock-free flow field at different flight Mach numbers. A recommendation as to the required volumetric change of the smart skin materials to be used for the continuous shape adjustments of the aerodynamic configurations was established.

1. INTRODUCTION

In the field of transonic aerodynamics, because of the highly non-linear nature of the problem, seemingly infinitesimal alterations in the shape of a flying object can have a dramatic influence on the entire flow field. Specifically, a airfoil that has been shaped to operate without any aerodynamic shock waves[1-2] at a given free stream angle of attack and a given transonic flight Mach number will start developing shock waves immediately when either angle of attack or the flight Mach number are perturbed. Thus, the main drawback of the present-day transonic shock-free inverse designs is that all these designs are of the single-point type rather than of a multi-point type. It has also been demonstrated that in order to maintain the same performance of an airfoil (for example, to maintain an identical distribution of surface pressures[3] on the airfoil) over a range of flight Mach numbers, the airfoil shape would have to readjust continuously and instantaneously. Although similar flow field alterations could be achieved by selectively heating or cooling the aerodynamic surfaces[4], the response time and the auxiliary thermal units render such concepts unsuitable for most aerodynamic applications. In the case of actual shape alteration, it has been observed that the modifications in the airfoil thickness and camber distribution are extremely small. Certain types of smart materials have a property that their volume can be instantaneously affected if subjected to a steady electric field[5-7] or a magnetic field[8 9]. If an aerodynamic surface (for example, a wing of an airplane, a rotor blade of a helicopter or an axial flow compressor, etc.) is coated with such a smart material and instrumented with pressure sensors, then, by applying adequate electric or magnetic fields locally to the coating the actual shape of the coated surface should change instantaneously in order to maintain a desired aerodynamic pressure distribution on the surface. This feature of an aerodynamic smart structure can have a significant impact on effectively maintaining desired steady aerodynamic loads and thus widening the operating range of such vehicles. Moreover, this concept could lead to effective elimination of unsteady aerodynamic loads and an active suppression of dangerous aeroelastic phenomena. A thorough evaluation of the quantitative measure of magnitude of the required shape alterations has not been reported in the open literature. Thus, it is not yet known if existing smart materials will have sufficient volume alteration capability to be utilized as smart aerodynamic coatings. This paper attempts to address this issue.

2. AERODYNAMIC FLOW FIELD ANALYSIS

We have utilized our computer code[2] to analyze transonic inviscid irrotational (potential) homoentropic fluid flow around isolated airfoils and in two-dimensional linear cascades. The code integrates iteratively a transonic full potential equation (FPE) given as

$$(1 - (\phi_x/a)^2)\, \phi_{xx} - 2\, \phi_x \phi_y \phi_{xy}/a^2 + (1 - (\phi_y/a)^2)\, \phi_{yy} = 0 \tag{1}$$

where a is the local speed of sound and ϕ is the velocity potential function so that ϕ_x and ϕ_y are the velocity vector components in the x and y direction, respectively. The equation is of a mixed elliptic/hyperbolic type thus accepting discontinuous solutions which are usually called shock waves. Since the basic assumption in deriving this equation is that the flow field has constant entropy, such shocks are only an approximation of the actual physical aerodynamic shock waves that generate entropy and vorticity in the flow downstream of the shock. But, if the flow field is free of shock waves, this model gives results that compare very well with experiments especially when combined with the influence of the viscous boundary layer[10] on the surface of the object analyzed. Our flow field analysis code uses a finite-area technique and successive line over-relaxation on a sequence of successively refined boundary conforming geometrically periodic O-type computational grids. For demonstration purposes alone, we have computed a flow field through a non-staggered cascade of symmetric NACA0012 airfoils having gap/chord ratio of 3.6, zero angle of attack and flight Mach number M = 0.8. The computed surface pressure coefficient (Fig. 1) distribution indicates the existence of a large supersonic region terminating with a strong shock wave that would in an actual situation cause unsteady boundary layer separation, high aerodynamic drag, and high noise level.

3. SHOCK-FREE INVERSE DESIGN

To eliminate all the shock waves while still moving at the same flight speed, one should change the shape of the basic airfoil appropriately. We have accomplished this redesign automatically by utilizing Sobieczky's fictitious gas concept as an inverse design tool. Since the shocks occur only where the flow speed locally changes from supersonic (locally hyperbolic FPE) to subsonic (locally elliptic FPE), the main idea is to make the governing equation (1) fictitiously elliptic throughout the entire flow field so that the shock waves cannot occur. This can be accomplished by utilizing a certain novel formulation for the dependency of the local density and the local speed of sound on the local value of the Mach number. This formulation is different from the known physical isentropic relations, thus the name "fictitious gas" technique. There are certain constraints that apply when creating such a new analytical model[2]. We used the following fictitious relation for the local density ρ normalized with the critical density ρ_*

$$\rho/\rho_* = 1 + (1 - A)/B \tag{2}$$

and the corresponding local speed of sound normalized with the critical speed of sound a_*

$$(a/a_*)^2 = M_* \; A\,(1 + A/B) \tag{3}$$

where

$$A = (1 + 2\,B\,(M_* - 1))^{1/2} \tag{4}$$

Here, M_* is the local critical Mach number and B is the user specified input parameter that can make the fictitious gas more or less compressible. Thus, when iteratively solving the FPE, isentropic relations should be used at every point of the flow field where the local speed is subsonic. But, at the points where the local speed is supersonic, the fictitious gas relations for the density and the speed of sound should be used. Consequently, the converged flow field solution is correct at all subsonic points, but it is incorrect (fictitious) at all supersonic points. Nevertheless, flow parameters computed on sonic lines ($M = 1$) separating regions of supersonic and subsonic flow are correct since at the sonic condition ($M = 1$) both physical and fictitious relations give the same results. Using these computed flow parameters along the sonic line as an initial data, a method of characteristics was used to numerically integrate physically corrects (isentropic) FPE inside the previously fictitious supersonic region. The only reason of utilizing the fictitious gas concept was to determine the shape of a sonic line that is compatible with a shock-free flow field. Once the physical isentropic supersonic flow inside the supersonic domain attached to the aerodynamic surface has been numerically determined, the new shock-free surface that is compatible with such a flow field was found from the condition that the value of a streamline on the entire aerodynamic surface should be constant.

4. COMPUTATIONAL RESULTS

Detailed and highly accurate evaluation of local thickness alterations necessary to maintain the shock-free flow field through the NACA0012 airfoil cascade at the given Mach numbers was performed with the objective of generating a definitive recommendation as to the required volumetric change of the candidate smart materials to be used for the continuous shape adjustments of the aerodynamic lifting surfaces. Using our highly accurate and proven inviscid transonic airfoil aerodynamic inverse design code[7] with $B = 1.5$ we have redesigned the cascade of NACA0012 airfoils and made it shockless at the free stream of $M = 0.8$. The computed sonic lines that are bounding the supersonic shock-free domains are symmetrically positioned (Fig. 2) on the upper and the lower airfoil surface. The corresponding computed surface coefficient of pressure distribution (Fig. 3) is clearly shock-free and significantly different from the shocked[7] original NACA0012 airfoil. Since the new shock-free NACA0012 airfoil shape is almost indistinguishable from the original NACA0012 shape, the two airfoil contours have been superimposed (Fig. 4) so that their vertical y-coordinates were multiplied by the factor of 10. The shape alteration that changed the flow field so dramatically is very small and confined to the region of the airfoil mainly covered by the supersonic flow. The required change in the airfoil surface y-coordinate can be then quantified by taking the difference between the original and the redesigned airfoil surface y-coordinate. The result (Fig. 5) was magnified by the factor of 100 and displayed on a graph (Fig. 5) having a significantly enlarged vertical axis in order to give a clear picture of the magnitude and the smooth variation of the required surface height alteration. This allows us to quantify the volumetric change requirements of the candidate smart skin materials for the aerodynamic applications. For example, helicopter rotor blades are build of airfoils that have approximately 10% maximum relative thickness and are almost symmetric. Let us say that such a blade on a heavy lifting rotor has a chord length of 500 mm. This means that its maximum actual semi-thickness is about 50 mm. Then, Figure 5 suggests that this maximum semi-thickness should be reduced by about 2.7%. That is, the redesigned blade should shrink in its maximum thickness by about 1.35 mm on both its upper and its lower surface. The entire skin of such a blade is typically not thicker than half an inch. Thus, if the entire skin of the helicopter rotor blade is made of a smart material, then its volumetric reduction under the influence of an electric or a magnetic field would be about 10% of its original volume. If such material is to be used as a coating only, the volumetric reduction requirement could easily reach 50% of the original volume or more.

5. CONCLUSIONS

Volumetric change requirements for candidate smart materials intended for instantaneous aerodynamic flow field control have been found to be quite severe amounting to a minimum of 10% reduction in volume. In case of the aerodynamic surfaces that have only a coating of such a material, the volumetric shrinkage requirements can be up to 50% and even higher.

6. REFERENCES

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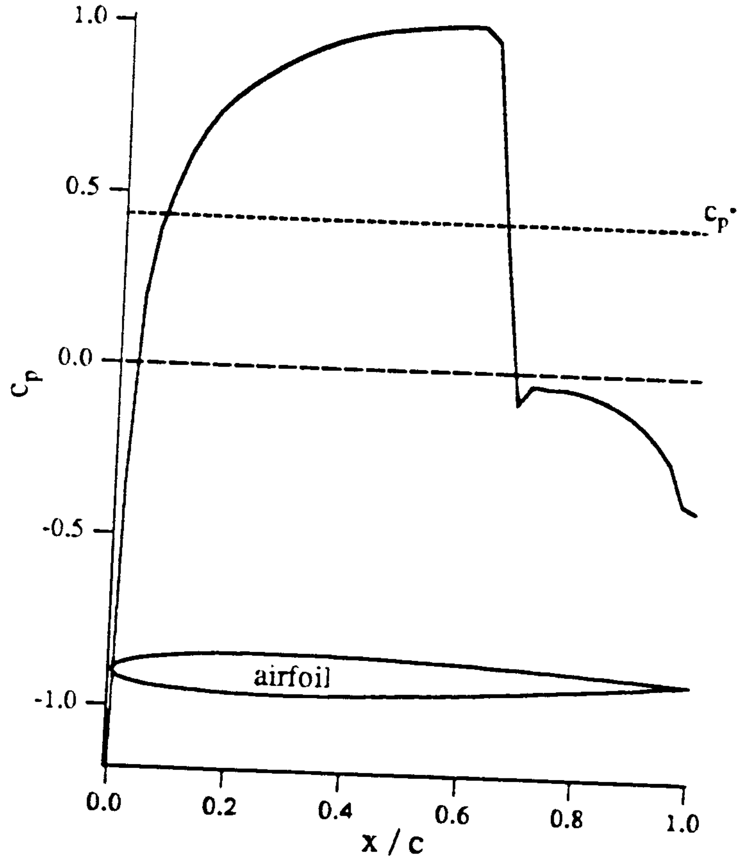

Figure 1. Surface coefficient of pressure distribution on the original NACA0012 airfoil in a cascade having gap/chord ratio of 3.6 and an oncoming free stream at M = 0.8.

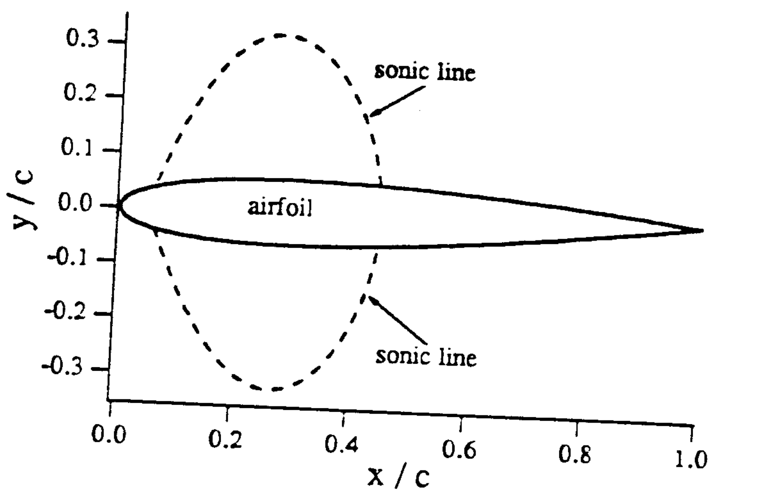

Figure 2. Computed sonic lines on the redesigned (shock-free) NACA0012 airfoil