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ADAPTIVE AIRFOILS AND WINGS FOR EFFICIENT TRANSONIC FLIGHT

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SUMMARY

A simple design method for two- and three-dimensional shock-free configurations is used for systematic airfoil modification to maintain shock-free flow at varying operating conditions. A mechanical realization is proposed since only minor and local changes of the contour are required.

INTRODUCTION

High speed aircraft design has become one of the most challenging fields of the aeronautical sciences. With availability of large computers new tools for design and analysis of aircraft components became available within the last decade, which encouraged the introduction of new aerodynamic concepts to increase fuel efficiency which is proportional to the ratio of lift over drag, multiplied by the flight Mach number. Rapidly increasing fuel -costs within the last years underlined the urgent call for techniques to improve efficiency of the next generation transport aircraft.

A possibility for increasing efficiency by drag reduction is to avoid the occurrence of shock waves which requires a complicated iterative process of aerodynamic shaping carried out using engineering experience, computational facilities and wind tunnels. The resulting wing shapes for the flight regime just below the speed of sound have become known as "supercritical wings", they are designed to be completely or nearly free of recompression shocks at certain operating conditions, while wings with conventional sections have strong shocks and, therefore, additional drag. Theoretically isolated within flow fields containing shocks if the operating conditions are slightly changed [1], such shock-free flows have been considered of not much practical value for some years, but pioneering experiments [2,3] also stimulated the development of computational methods to obtain practically interesting shock-free airfoil shapes [4,5].

These design methods are restricted to two-dimensional flow, they work in the hodograph plane and are, therefore, relatively complicated. A similar method [6] allowed an extension of the approach into physical space [7]. The ability to solve transonic design problems was then coupled to the development of reliable flow analysis algorithms by this approach. It led to efficient design methods which became known as "Elliptic Continuation" or "Fictitious Gas" methods. They form computational tools for the aerodynamic concept of adaptive aircraft geometry for adjusting contours to obtain optimal efficiency even at variable operating conditions.

DESIGN PROCEDURE FOR SHOCK-FREE FLOWS

The purpose of this paper is to illustrate some recent results obtained with a systematic computational procedure for supercritical airfoils and wings which are shock-free at prescribed operating conditions. Since the design method may be developed by extension of any reliable analysis algorithm, we give a short description of the concept with a physical interpretation in order to allow for an implementation of the idea into new and more sophisticated analysis methods becoming operational now and in future.

A local supersonic domain embedded into a subsonic flow field is enclosed in general by a surface consisting of the sonic isotach and a recompression shock. If the flow is shock-free the sonic surface forms a smooth convex bubble situated on the body surface. In this latter case the structure of the flow is qualitatively similar to a subsonic flow: isotachs of velocity higher than velocity at infinity form also bubbles with smooth transition of the flow properties. This relationship of subsonic and shock-free transonic flows gave rise to the following idea to calculate examples of shock-free flow (see Fig. 1):

In a first step we solve a partly fictitious problem by altering the governing isentropic density - velocity, $\rho_{is}(q)$ relation in the domain of supersonic velocities. An artificial compressibility relation $\rho_f(q) > a^*/q$ where $q > a^*$ (a^* the speed of sound) defines a fictitious supersonic flow with subsonic flow quality. The basic differential equation of the complete flow is now of elliptic type, locally describing physically realistic subsonic flows and fictitious supersonic flow. Such a flow will have no recompression shock, the sonic line will qualitatively resemble one of a physically realistic shock-free transonic flow. Examples of such flows may be obtained with use of numerical elliptic solver routines, we observe that only the local supersonic domain is physically not real, the surrounding flow field is a locally correct solution. We ask now for a possibility to use the subsonic part to construct a complete real shock-free flow.

The second step of the procedure consists of an integration of the real supersonic differential equations, with restored density $\rho_{is}(q)$. Initial conditions of this hyperbolic type problem are prescribed along the given sonic surface with velocity directions resulting from the previous solution of the fictitious problem. This ensures a smooth connection between the two physically real parts of the solution. Numerical marching procedures based on the method of characteristics allow an integration of the potential equation, starting at the sonic surface and proceeding toward the body surface. The latter was part of the first step elliptic boundary value problem but the resulting body stream surface from the hyperbolic initial value problem (initial values at the sonic surface) will be different from the given body where wetted by supersonic flow. The body will be flattened providing more space for the real flow than for the fictitious flow to pass because of $\rho_{is} < \rho_f$.

The analytical background[7,8] as well as the numerical aspects[9,10] of this method are described elsewhere, this paper is intended to present some illustrative results in the light of an application to advanced technology computational aircraft design tools.

SELECTION OF ANALYSIS ALGORITHMS, FICTITIOUS GAS MODELS AND BASELINE CONFIGURATIONS

Our design procedure requires in its first step a reliable analysis algorithm for elliptic partial differential equations to solve the subsonic part of the flow and provide flow properties along the sonic surface. Many computational codes are operational for inviscid flow past airfoils. We prefer solvers for the basic equations in conservation form. A finite difference relaxation code was extended to be a design tool[11]. A boundary layer method and - for the analysis version - a method to treat shock - boundary layer interaction was added[12]. Another computer code[13] based on the same analysis algorithm treats viscous interaction between boundary layer and wake[14]. Results obtained with these computer programs will be illustrated in the following.

Wing design codes based on the outlined method have been developed, too, but an implementation of 3D viscous effects still needs to be done. Both non-conservative finite difference and fully conservative finite volume codes have been extended to be shock-free wing design programs[15,16]. With rapid progress in numerical methods more efficient codes will become operational, examples given her are intended to stimulate the engineer to introduce the idea into new computer programs for transonic flow problems.

Given an analysis algorithm for transonic flow we have to introduce the design option by providing an alternate formula for the isentropic flow density

$$\varrho_{is}/\varrho^* = ((\gamma+1)/2 - (\gamma-1)/2 \cdot (q/a^*)^2)^{1/(\gamma-1)}$$

ensuring elliptic partial differential equations. The formula

$$\varrho_f/\varrho^* = c^p \cdot (q/a^* + c - 1)^{-p}$$

allows a 2-parametric variation of fictitious gas properties and elliptic equations if $p < 1$, $c \geq p$. A continuous slope at sonic conditions $q = a^*$, where ϱ_{is} is switched to ϱ_f, is obtained if $c = p$, but useful results with smooth body surface modifications may also be obtained for $c \neq p$. The value $c = 1$ gives

$$\varrho_f/\varrho^* = (q/a^*)^{-p} ,$$

results of this gas model have been studied extensively. Gas properties are defined by p and a result is illustrated in Fig. 2 for different values of p to demonstrate the influence of this parameter on the resulting new surface shape. A conventional NACA 0012 airfoil is flattened by the design procedure, we observe that a long flat sonic bubble on the airfoil is obtained by low values of p, here $p = 0$, which describes an incompressible fictitious gas. Surface changes between 2 and 47 percent chord are required, the maximum deviation of the new contour is 0.0054 percent chord.

For higher values of p the surface deformations are smaller and more local, but surface curvature changes become substantial if $p \longrightarrow 1$. This example illustrates the fact that shock-free modification of a given (initial-) configuration for prescribed lift coefficient and flight Mach number does not result in a unique new shape. A variety of shape changes within certain limits is possible and the criterion of choice of the fictitious gas model is the desired resulting pressure distribution on the airfoil. All types between "peaky" and "roof-top" c_p - distributions are possible and selection is at the designer's disposal. Off-design properties of an airfoil or wing are dominated by the occurrence of shock-waves and complicated by viscous interaction, but the design pressure distribution is crucial for prediction of these effects. This leads us to the selection of baseline configurations. Extensive experimental work was performed to arrive at the widely used and well documented NACA airfoils. One of the first results of this method was a series of shock-free modifications of a NACA 64A410 airfoil. The results are illustrated in Fig. 3, in a Mach-c_L-diagram. We see the amount of thickness reduction and the limits for shock-free redesign of this airfoil and chosen gas parameter ($p = 0$).

A thickness reduction usually tends to shift the occurrence of shock-waves and drag rise toward higher Mach numbers. So a shock-free modification requiring thickness reduction seems not very surprising. A shock-free modification without reduction of the maximum thickness seems important for practical design requirements. Fig. 4 illustrates another result, the verification of a known shock-free inviscid flow (KORN airfoil 75-06-12) with our method. A local surface thickness bump had to be added to the upper surface, a careful variation of its shape and the gas parameter p finally resulted in equal thickness addition and subsequent design thickness reduction so that the original KORN airfoil and its pressure distribution was verified.

These inviscid test results illustrated above lead us to the conclusion that we have computational tools to

- modify conventional configurations to be shock-free at transonic operating conditions,
- specify the type of shock-free flow by a selection of fictitious gas model and initial configuration geometry changes,

- obtain a whole series of neighboring shock-free flow solutions for variable operating conditions.

It is this third capacity of the method we will investigate in the following.

SHOCK-FREE AIRFOIL SERIES: CONCEPT OF ADAPTIVE CONFIGURATION

Aerodynamic efficiency of a wing is defined by the ratio of lift over drag, multiplied by the flight Mach number. With drag rising sharply if the Mach number approaches unity, efficiency drops and it is therefore a principal goal of high subsonic speed aircraft design to delay drag-rise to higher Mach numbers for prescribed lift. This is usually achieved by delaying the occurrence of shock waves to higher Mach numbers through a careful variation of wing shapes, many analysis computations and very costly wind tunnel experiments.

Our design method seems to be a useful tool to obtain better airfoils and wings for transonic flight. Moreover, the computational definition of surface modifications for varying free stream conditions gives an idea about possible mechanical adjustments of the configuration in order to maintain efficient operation even at different flight conditions.

We choose a design example for illustration of the required surface modifications at varying flight Mach number at constant lift coefficient, Fig. 5. A given airfoil A is designed to be shock-free at Mach = 0.73, c_L = 0.55. We ask for its performance at Mach = 0.75 and c_L = 0.6. Analysis including viscous interaction gives a result with a recompression shock. A bump, added to the upper surface gave an initial configuration 0.3 percent thicker than airfoil A, original thickness was obtained from the subsequent design computation. The new airfoil B is investigated by the analysis version of the code to confirm the design result. Fig. 6 shows off design analysis results, we see that an increase of 0.01 of the drag rise Mach number has been achieved.

Geometry modifications which led from airfoil A to B are depicted in Fig. 7. Addition of a bump (a) which extends from 0 to 85 percent chord and subtraction of a design bump (b) within the supersonic region from 2 to 68 percent chord leaves two small bumps to be added to the original airfoil. These bumps are only 0.0021 and 0.0013 percent chord high. At this point we might think about a technical realization of such a bump addition in order to have both airfoils available for operation. Experiments with a possible use of elastic or pneumatic devices should be carried out. Another concept is a controlled distribution of suction and blowing as already investigated for laminar flow control and similar efforts to influence flow quality. Boundary layer displacement of the flow past airfoil B at design conditions is drawn in Fig. 7 (curve c) for comparison with the required surface modifications.

Another example to investigate sensivity of the calculated shock-free design is shown in Fig. 8. NACA 0012 airfoil was modified to be shock-free using incompressible fictitious gas ($p = 0$). We are interested now in an approximate representation of the calculated surface modifications by a smooth analytical curve. For simplicity we choose a spline function with few supports which is the mathematical model of an elastic beam deformed by single loads. Analysis results are compared with design pressure distribution and to our pleasant surprise we find that this airfoil with an elastic section is practically shock-free, too, even though the pressure distribution and sonic line are different from the original design. The reason for this is obviously related to the multiplicity of possible shock-free designs with different fictitious gas parameters, Fig. 2.

Having proved that desirable flow quality could be achieved by shape changes generated by mechanical devices we go one step further and propose [17] a system for automatically controlling the flow quality, Fig. 9. In the system shown, a flow qualitiy sensor F determines the operating conditions and surface pressure at selected stations and is interrogated by a microcomputer M that determines the proper changes of the effective contour necessary for shock-wave reduction. Our experience with the presented design method enables us to set up the programming of the micro-computer which energizes a servo system S which appropriately alters the effective shape. This is accomplished by servo motors for mechanically adjusting sections of the wing surface, and other mechanical devices on the structure for the opening or closing of apertures on the wing surface to bleed (or add) various amounts of air from (to) the upper surface of the wing. Any combination of the above may also be used. As seen from the illustrated examples, the surface area that needs to be changed is limited and the amount of change required is small.

ADAPTIVE SUPERCRITICAL WINGS

We have outlined a concept of transonic design and illustrated some cases of airfoil flow. At this stage experiments need to be performed to prove both new design results and some realization of adaptive airfoil technology in the wind tunnel.

Parallel to experimental verification an implementation of the idea into new and reliable 3D wing and wing-body configuration analysis codes is necessary. Our experience with wing design is limited to date, also because of a lack of 3D boundary layer and viscous interaction methods. Design studies of inviscid shock-free wings are presently carried out to refine the 3D marching procedure and determine the structure of 3D local supersonic shock-free flow fields. Fig. 10 shows a result obtained by extension of a finite difference analysis code[16] to a design tool. A simple wing based on NACA 64A410 section is modified to be shock-free.

Viscous effects may be accounted for by adding estimated displacement thickness to the initial configuration. Results of a finite volume analysis code[18] design extension for a shock-free supercritical wing with added displacement thickness is shown in Fig. 11. Extent of the supersonic domain on this "flying wing" without body defines the area of possible adaptive surface changes. A thick span loader flying wing seems to be a suitable test bed for experiments with 3D adaptive devices.

CONCLUSION

We have applied the elliptic continuation shock-free flow design method to some illustrative test examples to form a theoretical base for the concept of adaptive wing geometry at variable operating conditions. A system for automatic shape variations of wings based on experience with systematic computational design is proposed. Both special designs and the adaptive shape control system need to be tested experimentally, possibly in combination with new aerodynamic concepts for higher efficiency of transonic aircraft like variable geometry and boundary layer control investigations.

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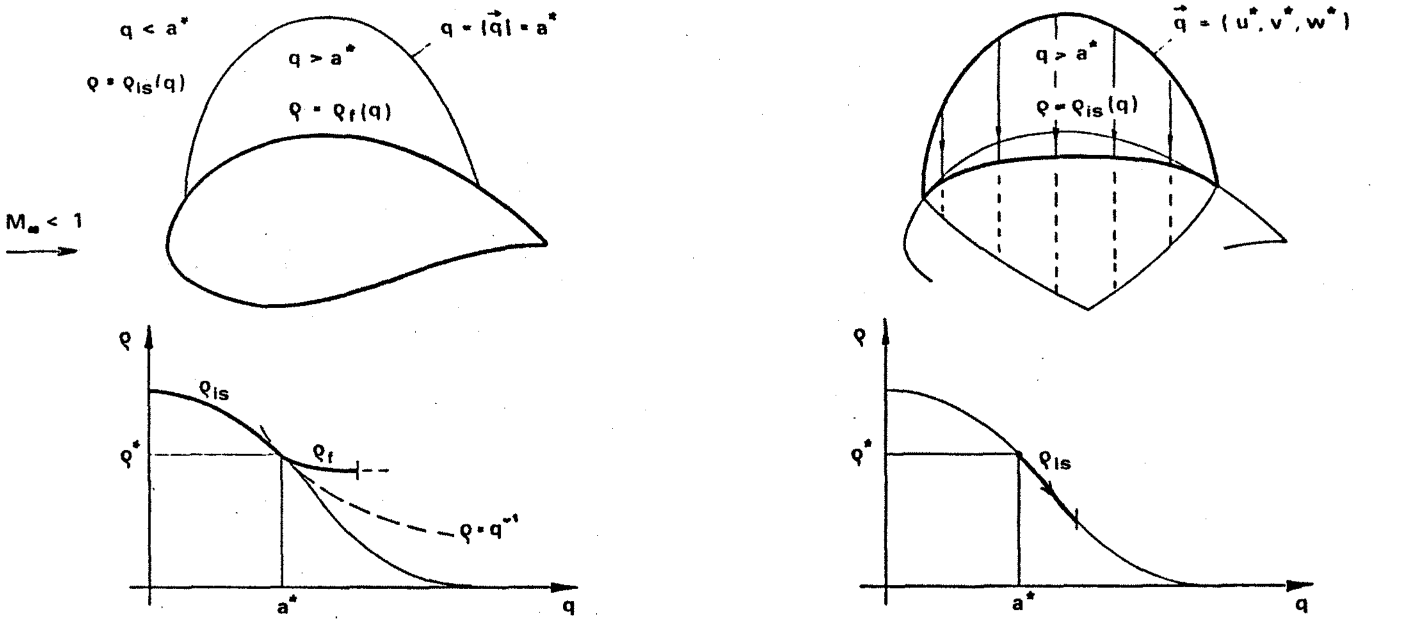

FICTITIOUS COMPRESSIBILITY $\rho_f(q > a^*)$,
ELLIPTIC EQUATIONS FOR $\rho_f > q^{-1}$,
BOUNDARY VALUES, ELLIPTIC SOLVER.

ISENTROPIC COMPRESSIBILITY $\rho_{is}(q > a^*)$,
HYPERBOLIC EQUATIONS FOR $q > a^*$,
INITIAL VALUES, MARCHING PROCEDURE.

Fig. 1 Elliptic continuation shock-free design.
a) First step: Fictitious gas flow analysis
b) Second step: Supersonic domain integration

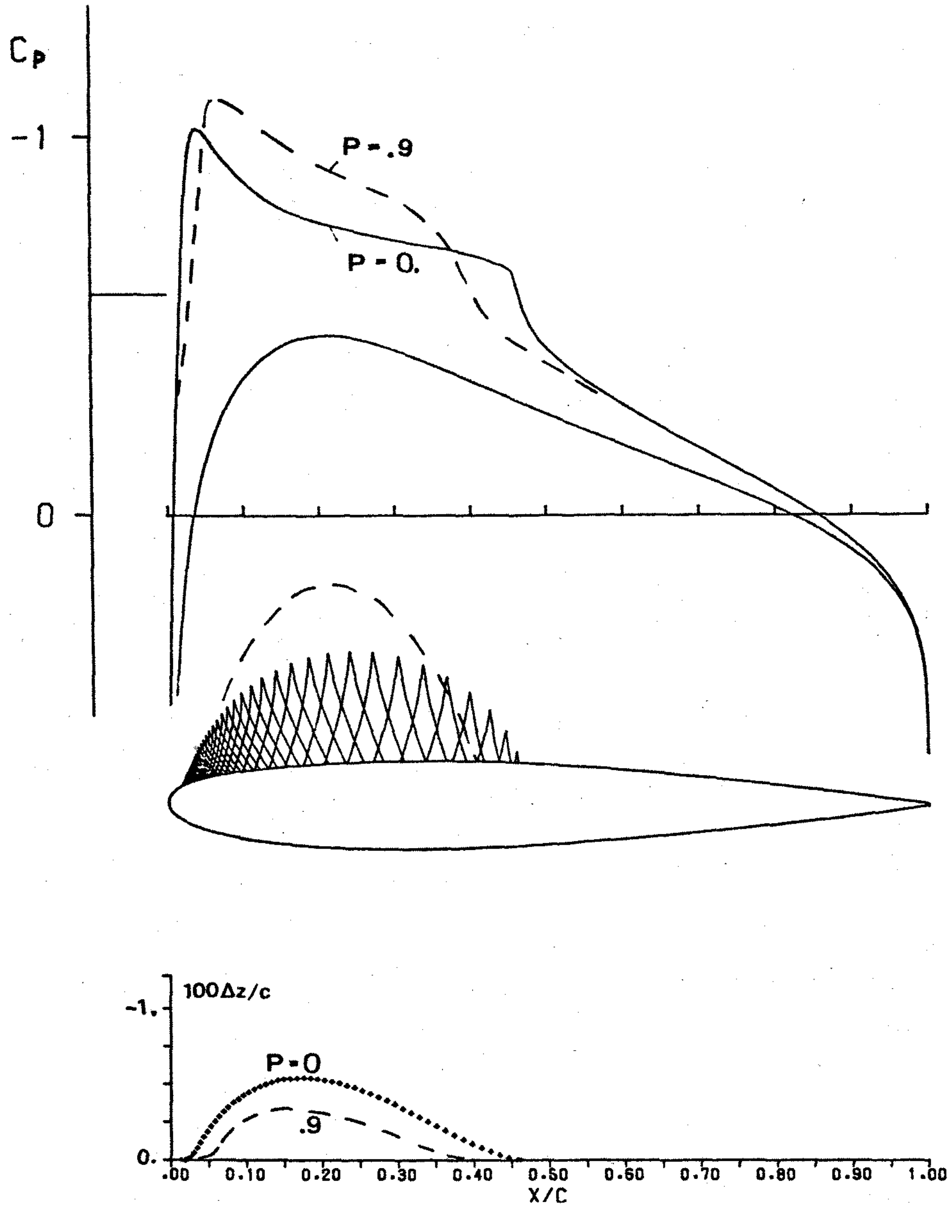

Fig. 2 Surface modifications and structure of local supersonic flow field for different fictitious gas parameters

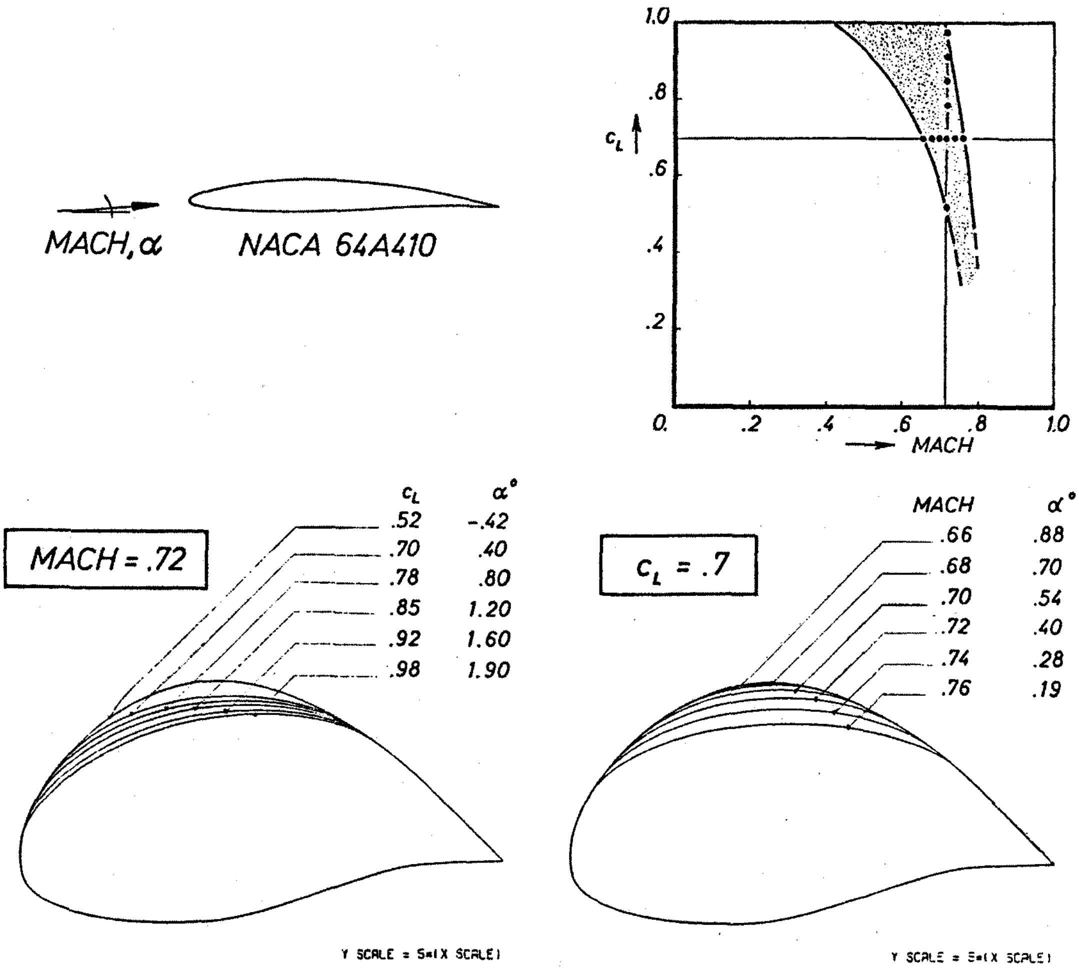

Fig. 3 Inviscid design example: Shock-free flow modifications of NACA 64A410 airfoil.

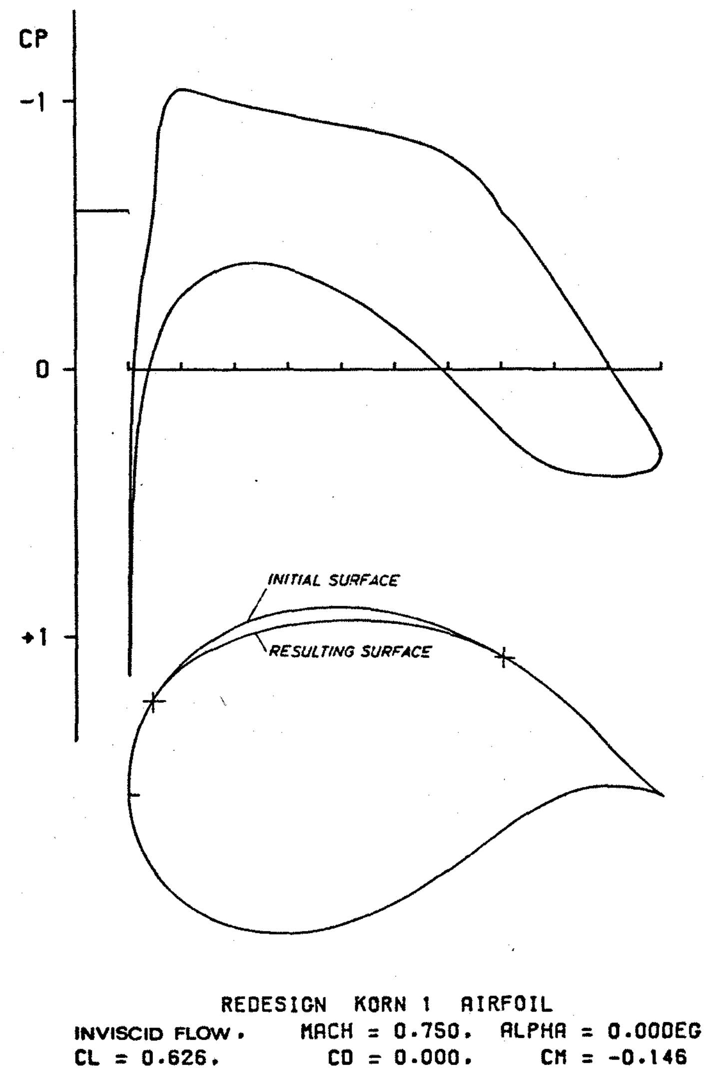

Fig. 4 Design verification of an inviscid shock-free flow: Korn 1 (75-06-12) airfoil.

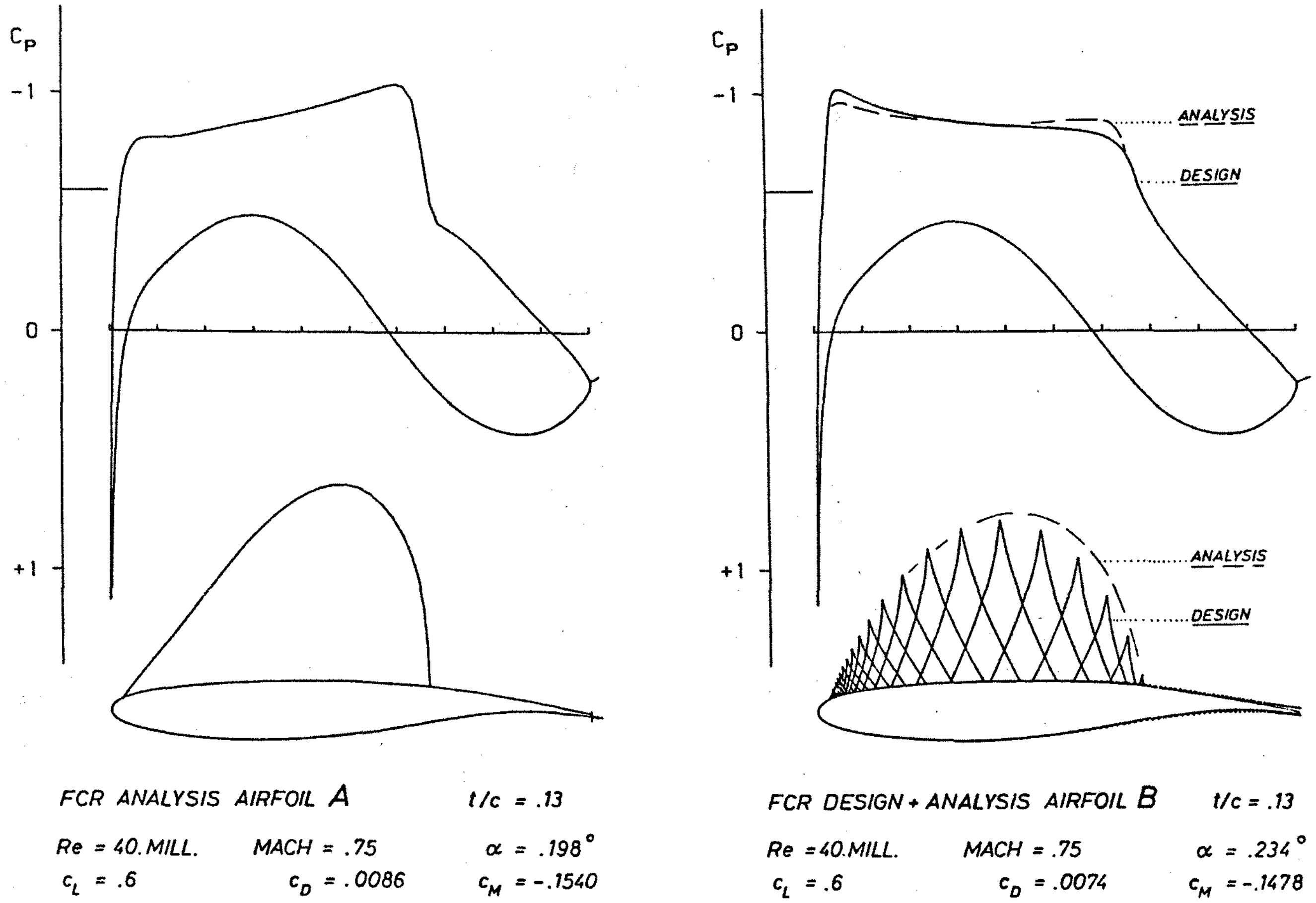

Fig. 5 Shock-free redesign with constant thickness for prescribed Mach number and lift coefficient, including viscous effects.

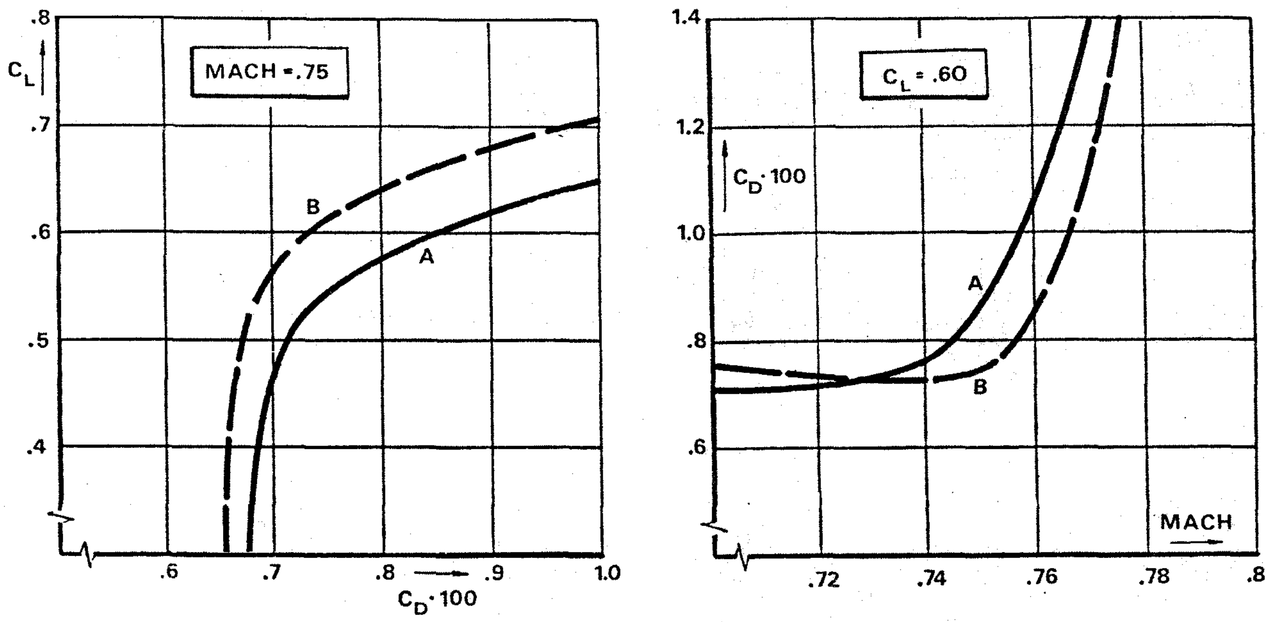

Fig. 6 Comparison of original and redesigned airfoil: lift, drag, Mach number.

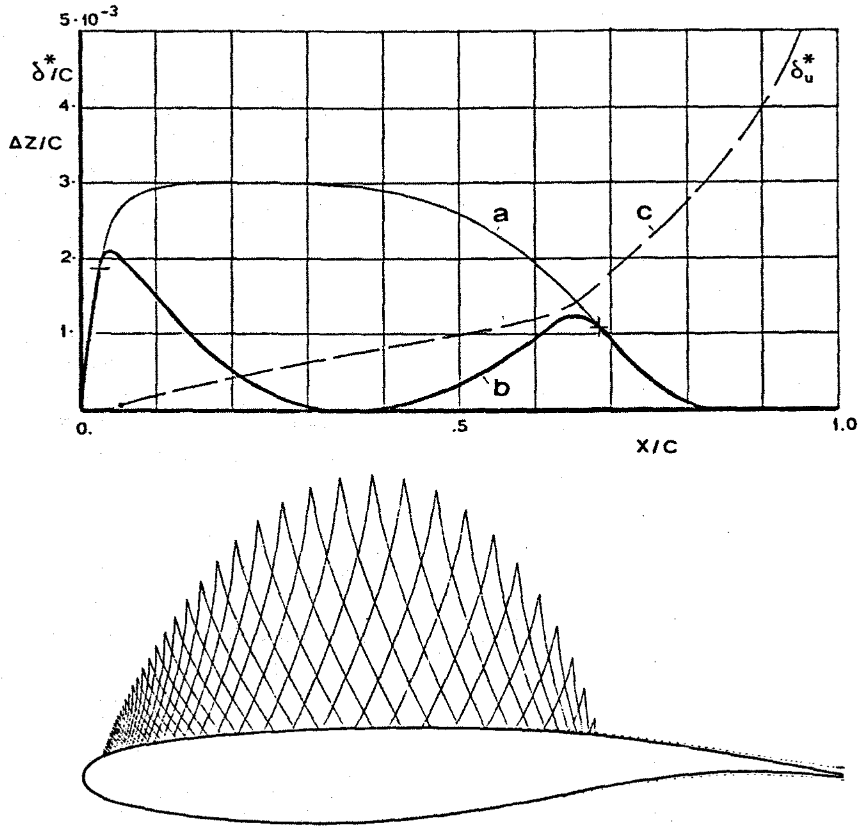

Fig. 7 Surface modifications:
(a) addition for fictitious gas analysis
(b) resulting addition after redesign;
(c) for comparison: upper surface boundary layer displacement thickness.

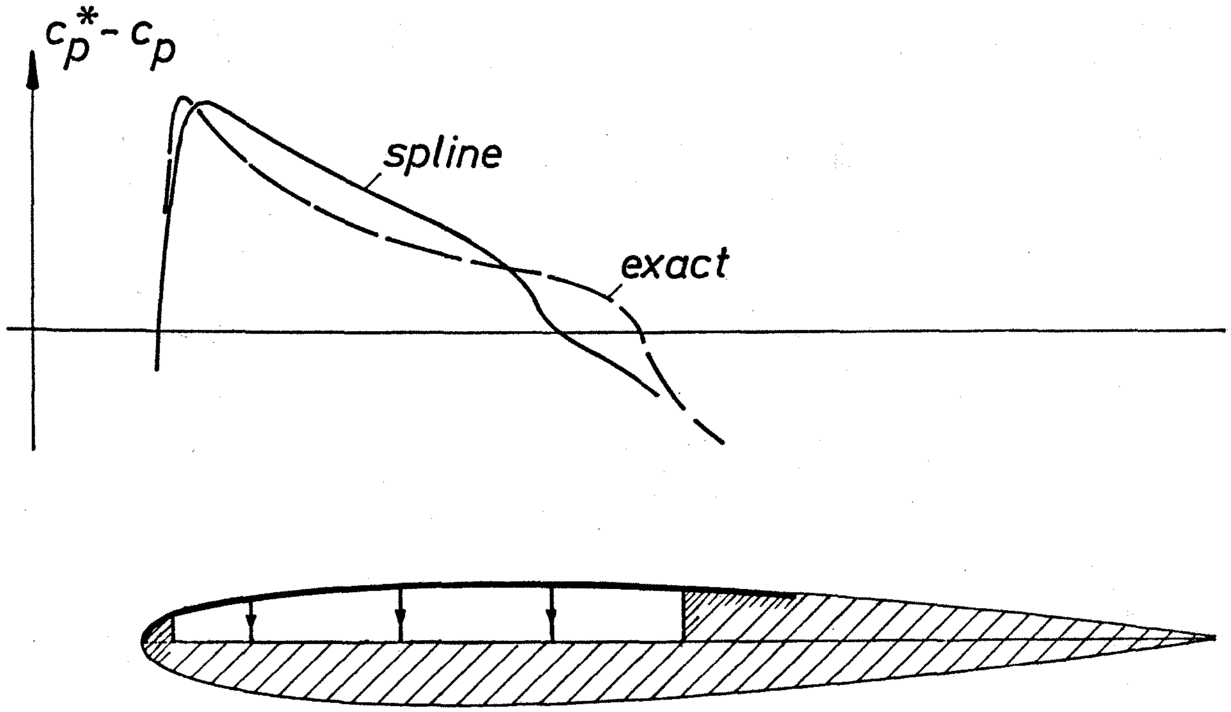

Fig. 8   Shock-free modification of NACA 0012 airfoil, elastic section approximation.

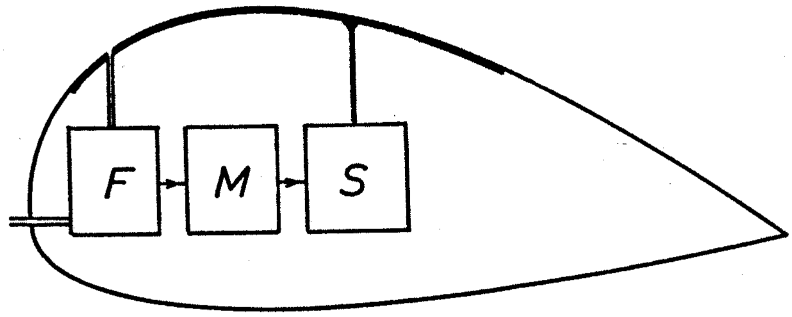

Fig. 9 Adaptive shape control system.

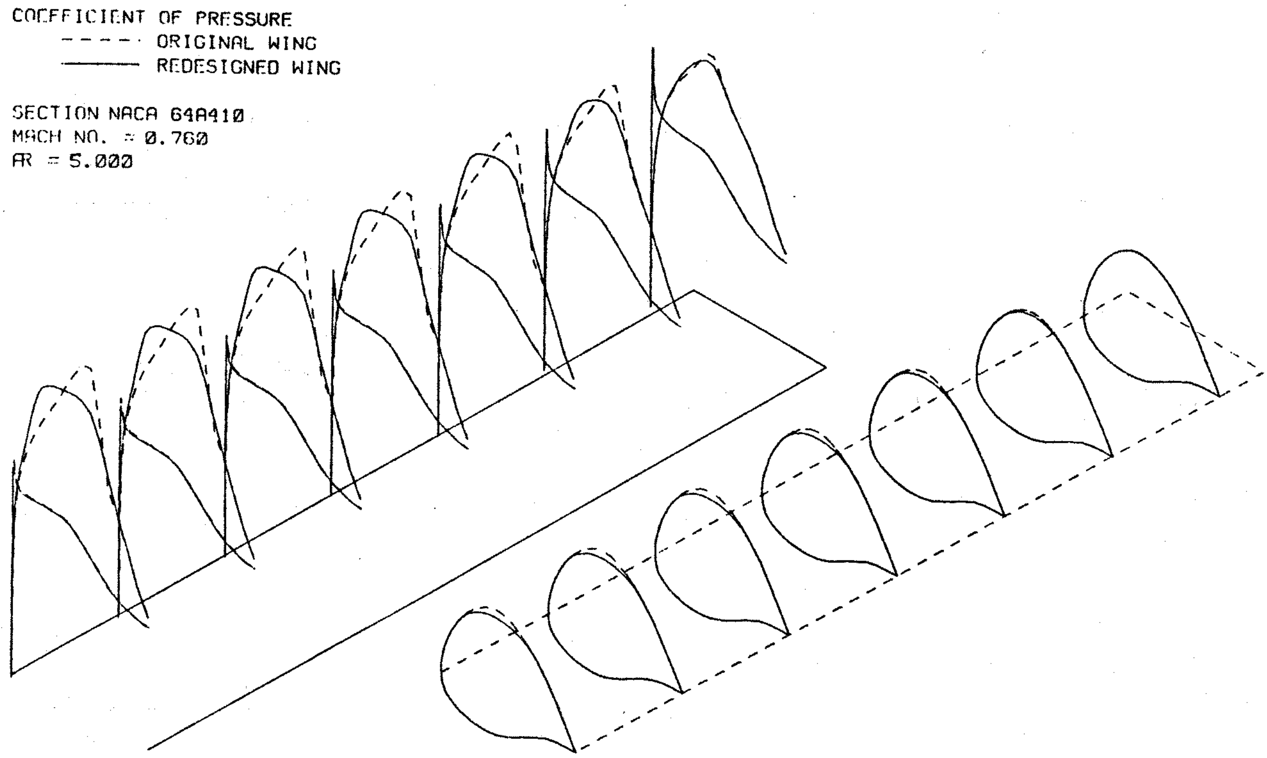

Fig. 10 Shock-free redesign of a rectangular wing with NACA 64A410 section.

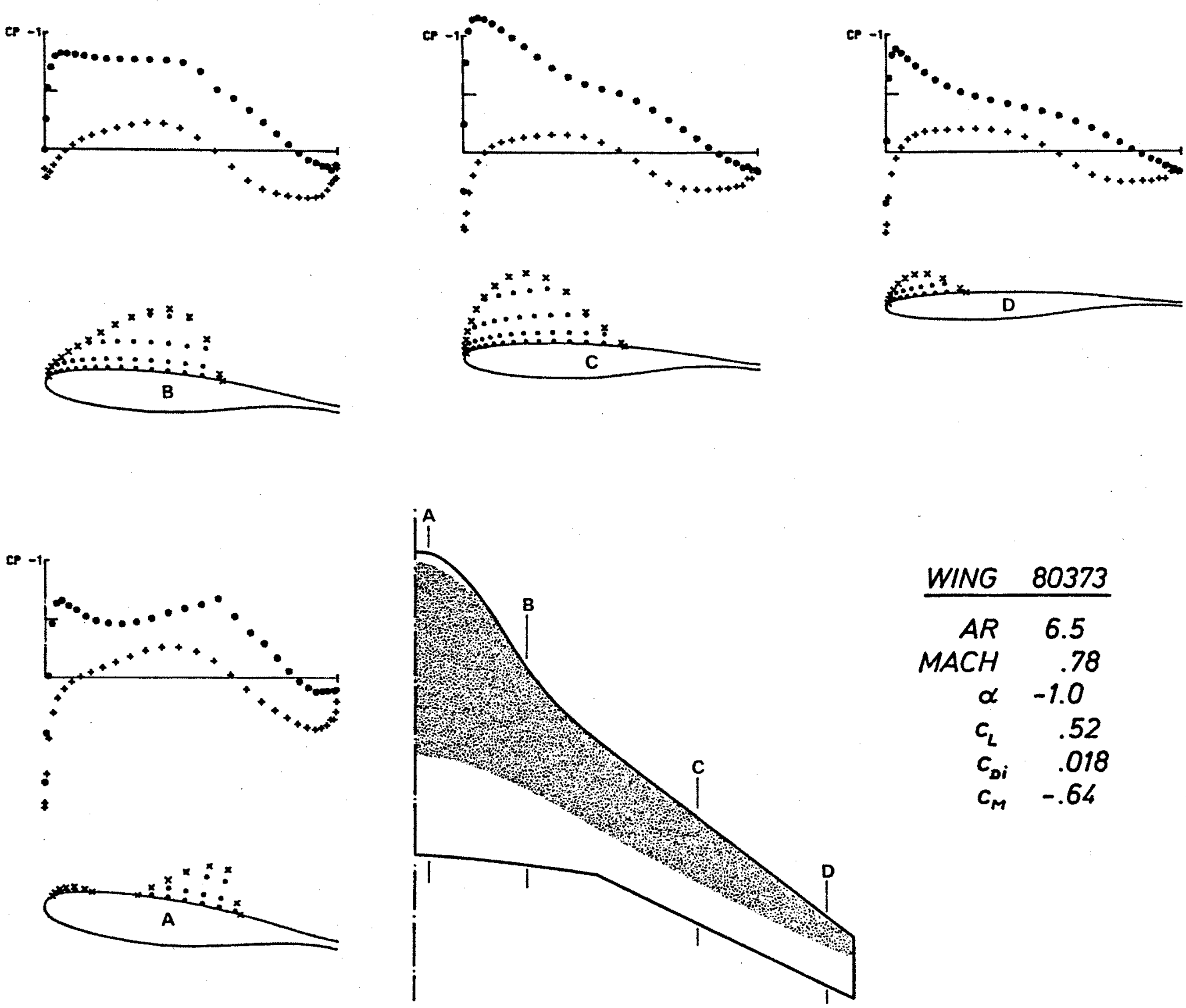

Fig. 11 Shock-free design of a swept wing.
Viscous diplacement model added.