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CALCULATION OF THE PLANAR SUPERCRITICAL FLOW OVER A NASA SUPERCRITICAL PROFILE (U)

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CALCULATION OF THE PLANAR SUPERCRITICAL FLOW OVER A NASA SUPERCRITICAL PROFILE

By H. Yoshihara and R. Magnus
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SUMMARY

An unsteady finite difference procedure is used to calculate the steady inviscid flow over a 11% thickness ratio NASA supercritical profile of LWP 505 at $M_\infty = 0.80$ and $\alpha = 0$. An attempt is made to include the viscous effects using a modified form of Head's entrainment method to calculate the turbulent boundary layer. The results of the inviscid calculations compared to the experimental results of Whitcomb indicate a significant influence of viscosity, though the comparison is partially obscured by the unknown wind tunnel wall interference effects. The effects of viscosity were manifested in three ways. Firstly, the experiments clearly indicated the presence of shock-induced separation at the base of the shock. The resultant flow displacement effect produced a significant upstream movement of the shock. Secondly, the displacement effect of the boundary layer, primarily on the aft upper surface, reduced the effective aft camber which in turn resulted in changes of the "plateau" pressures on both the upper and lower surfaces characteristic of a decreased flap effect. Thirdly, the severe adverse gradients on the aft portion of the lower surface resulted in a significant thickening or a local separation of the boundary layer that led to decreased overpressures.

The attempt to predict the viscous effects using the compressible form of Head's integral method with a modified auxiliary equation for the form factors was unsatisfactory. Though a reasonable separation bubble was established on the lower surface, a grossly exaggerated displacement effect resulted downstream of the shock on the upper surface. Here in these exploratory calculations the experimental pressure distribution was simply used to check the boundary layer equations. There clearly is substantial further effort required to evolve a satisfactory boundary layer procedure, which must then be coupled in a still unproven manner with the inviscid procedure.

INTRODUCTION

The calculation of the planar supercritical flow over airfoils is universally recognized as a horrendous task. Sources of the difficulties are manifold. At the high Reynolds numbers of pertinence the resulting boundary layer is turbulent, and is sufficiently thin that the classical boundary layer concept would be applicable were it not for the presence of the terminating shock wave. The presence of the shock wave can so drastically alter the "effective" shape of the profile through the displacement effects of the boundary layer that the inviscid solution no longer serves directly as a viable basis to obtain the pressure distribution. In short the inviscid and boundary layer flows are strongly coupled. Despite the strong coupling with proper ingenuity it would not be a hopeless task to compute the coupled flow if methods were available to calculate both the inviscid flow over a given shape, and the turbulent boundary layer flow, possibly separated, in the presence of a prescribed pressure distribution.

In the present study the objective will be to carry out the inviscid calculations for a prescribed airfoil, namely the 11% thickness ratio NASA supercritical profile of LWP 505, at $\alpha = 0^{\circ}$ and $M_{\infty} = 0.80$, and attempt to extend the results to include the viscous effects.

THE INVISCID CALCULATIONS

To compute the inviscid flow the unsteady finite difference procedure as described in Ref. 1 will be used. Here the coexistence of subsonic and supersonic flows with their highly differing mathematical characters in the steady state makes the unsteady approach more tractable than the steady from a calculational point of view, since in the latter representation the character of the equations remains hyperbolic in both the subsonic and supersonic flow regimes.

In Figure 1 is shown the resulting Mach wave pattern for the supercritical profile at $M_{\infty} = 0.80$ and $\alpha = 0^{\circ}$, whereas in Figure 2 the resulting chordwise distribution of the pressure is shown. Also shown in Fig. 2 are some experimental results as obtained by Whitcomb (Ref. 2) for several geometric values of α. The pressure distributions for the several values of α are shown here since the precise value of the angle of attack is not known because of the disturbing influence of the wind tunnel walls. If tests had been carried out additionally for a geometric α of approximately 2°, the plateau values of the pressure on both the upper and lower surfaces would agree closely with the calculated result. The immediate conclusion might then be that there is a minus 2° correction in α due to wall effects for this case. That this may not be a completely correct conclusion may be suggested by the lowness of the trailing edge pressure, and the fact the shock pressure rise falls significantly short of that for the normal shock. The latter clearly indicates the presence of a separation bubble immediately downstream of the shock which then leads to a significant boundary layer displacement effect in the vicinity of the upper surface trailing edge as manifested by the depressed trailing edge pressure. The added displacement effect of the boundary layer here gives rise to a lessening of the effective trailing edge angle; that is, a lessening of the aft camber. This in turn leads to increased plateau pressures on the upper side and decreased plateau pressures on the lower side as would result from a decreased flap effect. In short, the modifications of the pressure distribution due to viscous effects on the airfoil surface are of the same general character as that due to wall interference effects.

Aside from the viscous effects just described a more serious consequence of the separation bubble is that the shock wave at the surface is now oblique instead of normal as in the unseparated case, and the resultant shock pressure rise will then be significantly less than the normal shock pressure rise. As a result the shock wave will move upstream to a position sufficiently far upstream to allow a sufficient "run" of subsonic recompression to attain the required trailing edge pressure. Quite apart from the above effect there is a countering influence on the shock movement by the depression of the trailing edge pressure. Thus, for a depressed trailing edge pressure there would be less subsonic recompression required, and as a result the shock wave would not have to migrate as far upstream as for a higher trailing edge pressure.

Finally in Fig. 2 it is seen that there is a significant depression of the overpressures on the lower aft surface caused by a separation or a thickening of the boundary layer resulting from the severe adverse subcritical pressure gradient caused by the aft camber.

The above results clearly indicate that inviscid results must be supplemented by the addition of the viscous effects.

VISCOUS CALCULATIONS

There presently does not exist a suitable procedure to calculate a turbulent boundary layer in the presence of a prescribed external pressure distribution containing adverse pressure gradients sufficient to lead to separation. An attempt will be made however to extend an existing method for this purpose. Of the many procedures one of the more promising methods is due to Head (Ref. 3) and Green (Ref. 4), and this procedure will be adopted for our further consideration. Another widely used procedure is due to Nash and Macdonald (Ref. 5). The latter method is an extremely simple procedure and is based upon the use of the classical streamwise momentum integral equation which results in the following ordinary differential equation for the momentum thickness θ:

$$\frac{d\theta}{dx} = -\left(H + 2 - M_e^2\right)\theta \frac{d\,\ln u_e}{dx} + \frac{1}{2} C_f \qquad (1)$$

Here H is the usual form factor $H = \delta^*/\theta$, M_e and u_e are the Mach number and velocity in the external inviscid flow, and C_f is the wall skin friction coefficient. In the Nash-Macdonald method Eq. (1) is supplemented by two additional equations in order to obtain a fully determined system, assuming for the time being that $u_e(x)$ and $M_e(x)$ are prescribed. First of the required equations expresses C_f as a function of the Reynolds number based upon θ, and one or more of the other dependent variables appearing in (1). A suitable form is that given in Ref. 5, or the well known Ludwieg-Tillmann formula in one of its many variant forms. The second auxiliary relation suggested by Nash and Macdonald is based upon the assumption that the flow is locally an "equilibrium boundary layer;" that is one that fulfills the condition

$$G(\pi) = 6.1\,(\pi + 1.81)^{1/2} - 1.7 \qquad (2)$$

where

$$\pi \equiv -\frac{H}{\sqrt{C_f/2}}\,\theta\,\frac{d\,\ln u_e}{dx}, \qquad \text{and}$$

$$G \equiv (C_f/2)^{-1/2}\left(\frac{\bar{H} - 1}{\bar{H}}\right)$$

with

$$\bar{H} = \frac{H + 1}{1 + 0.178\,M_e^2} - 1$$

For a given $u_e(x)$ and $M_e(x)$ the differential equation (1) for θ can be integrated using (2) and the skin friction law, and one can obtain quite simply the chordwise distribution of $\theta(x)$, $C_f(x)$, as well as $\delta^*(x)$. For an unseparated boundary layer use of the above procedure has been found to yield a satisfactory distribution for $\theta(x)$, but is inadequate to yield an acceptable distribution for $\delta^*(x)$ (Ref. 5). (The general experience among most integral methods has been that all procedures appear to yield a reasonable distribution for θ, but the determination of δ^* is a much more delicate undertaking.) For the separated case it can be expected that the Nash-Macdonald procedure would simply be grossly inadequate to predict $\delta^*(x)$. (Note that the authors of Ref. 5 do not claim the applicability of their procedure to the separated case).

A more sophisticated approach is the use of the compressible form of Head's procedure (Ref. 3) given by Green (Ref. 4). The basic equations in this procedure in addition to Eq. (1) include the integral form of the continuity equation, namely,

$$\frac{d\Delta}{dx} = F_e + (M_e^2 - 1)\,\Delta\,\frac{d\,\ln u_e}{dx} \tag{3}$$

Here

$$\Delta \equiv \delta - \delta^* = \int_0^{\delta} \frac{\rho u}{\rho_e u_e}\,dy \text{ , and}$$

$$F_e \equiv -\frac{v_e}{u_e} + \frac{d\delta}{dx} \qquad \text{(Entrainment function)}$$

where ρ is the density, δ the boundary layer thickness, v the transverse velocity, and the subscript e denotes the local inviscid value at $y = \delta$.

In the Head-Green method the basic differential equations (1) and (3) must be supplemented by three additional equations. These are first a skin friction law, for which we use the Felsch modification of the Ludwieg-Tillmann law. (Ref. 1). The second is an expression for the entrainment function F_e, and the third is a relation of the form $H = H(H_1)$ where $H_1 \equiv \Delta/\theta$. It is in the last relation that an attempt was made to derive an expression valid for the separated case using available experimental data. For this purpose the experiments of Green (Ref. 4) and Seddon (Ref. 6) were used to derive the relation $H = H(H_1)$ despite the fact that in the latter experiments it can be clearly shown that there is serious departure from the planar condition due to the influence of the sidewall boundary layer displacement effects (see Ref. 1).

To test the resulting set of equations calculations were carried out where an existing experimental pressure distribution from LWP 505 was used. The results showed a reasonable δ^* upstream of the abrupt adverse gradients on both the upper and lower surfaces as would be expected. On the lower surface the abrupt recompression led to a local separation bubble that appeared to be reasonable, but downstream of the shock on the upper surface the resulting distribution of δ^* grew excessively towards the trailing edge.

This unreasonable growth of δ^* as the trailing edge was approached could be directly traced to the inadequacy of the auxiliary relation $H = H(H_1)$ used which yielded near-singular values of H for the values of H_1 that were being predicted.

Despite this shortcoming the resulting δ^*, with the unreasonable values arbitrarily modified, was used to obtain a modified airfoil shape, and the inviscid flow was recalculated in order to obtain a qualitative effect of the boundary layer displacement. The results indicated the change of the pressure distribution described earlier for the decreased aft camber.

These highly tentative attempts to incorporate the viscous effects clearly point out areas of future effort required to improve the procedure. With a reasonable amount of additional effort it is not excessively optimistic to expect that a viable boundary layer procedure can be eventually evolved to handle the specialized case of shock-induced separations on supercritical airfoil flows.

Convair Division of General Dynamics
San Diego, California, 18 September 1970

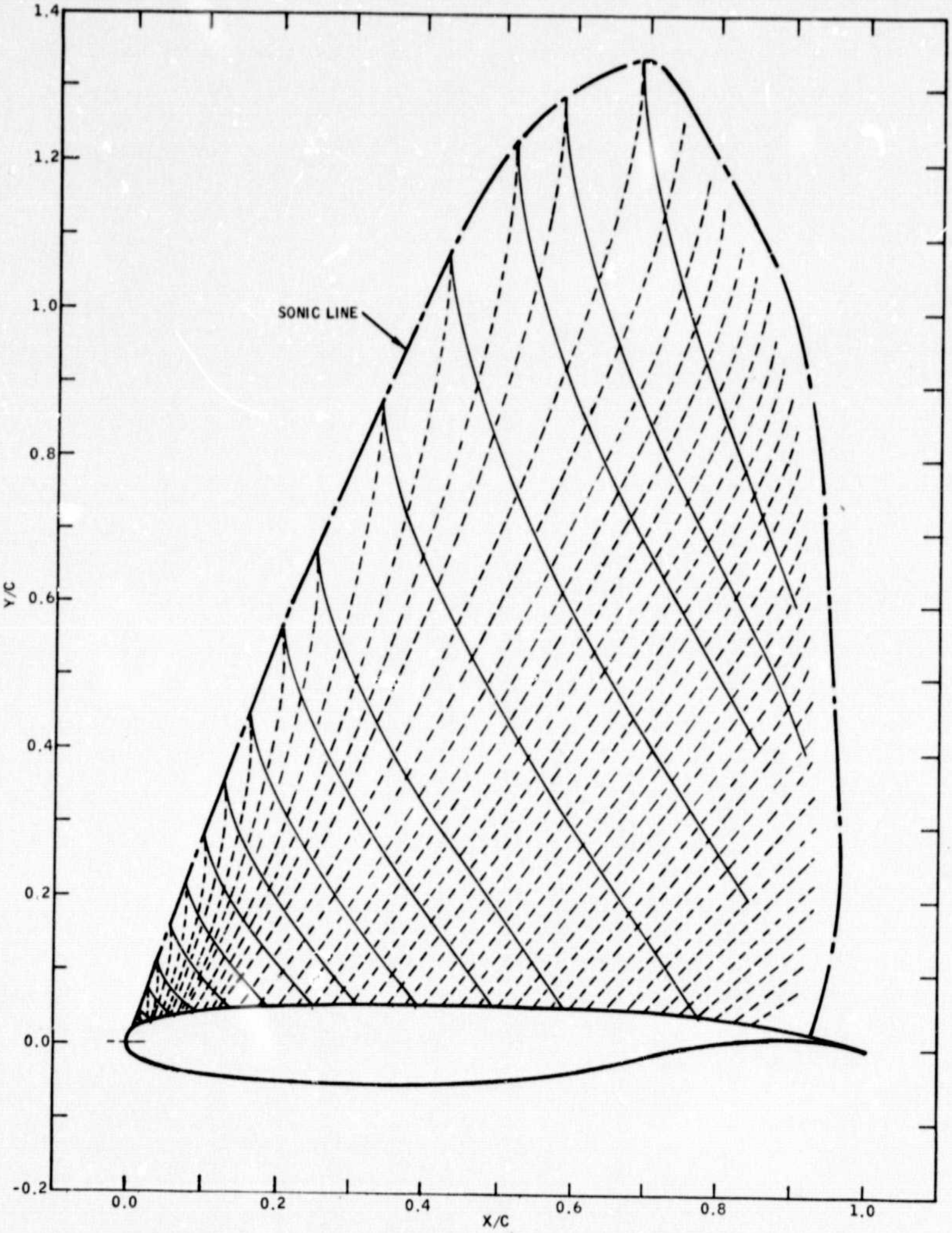

Figure 1. Mach Wave Pattern for the Basic NASA Profile at $M_{\infty} = 0.80$ and $\alpha = 0.0°$

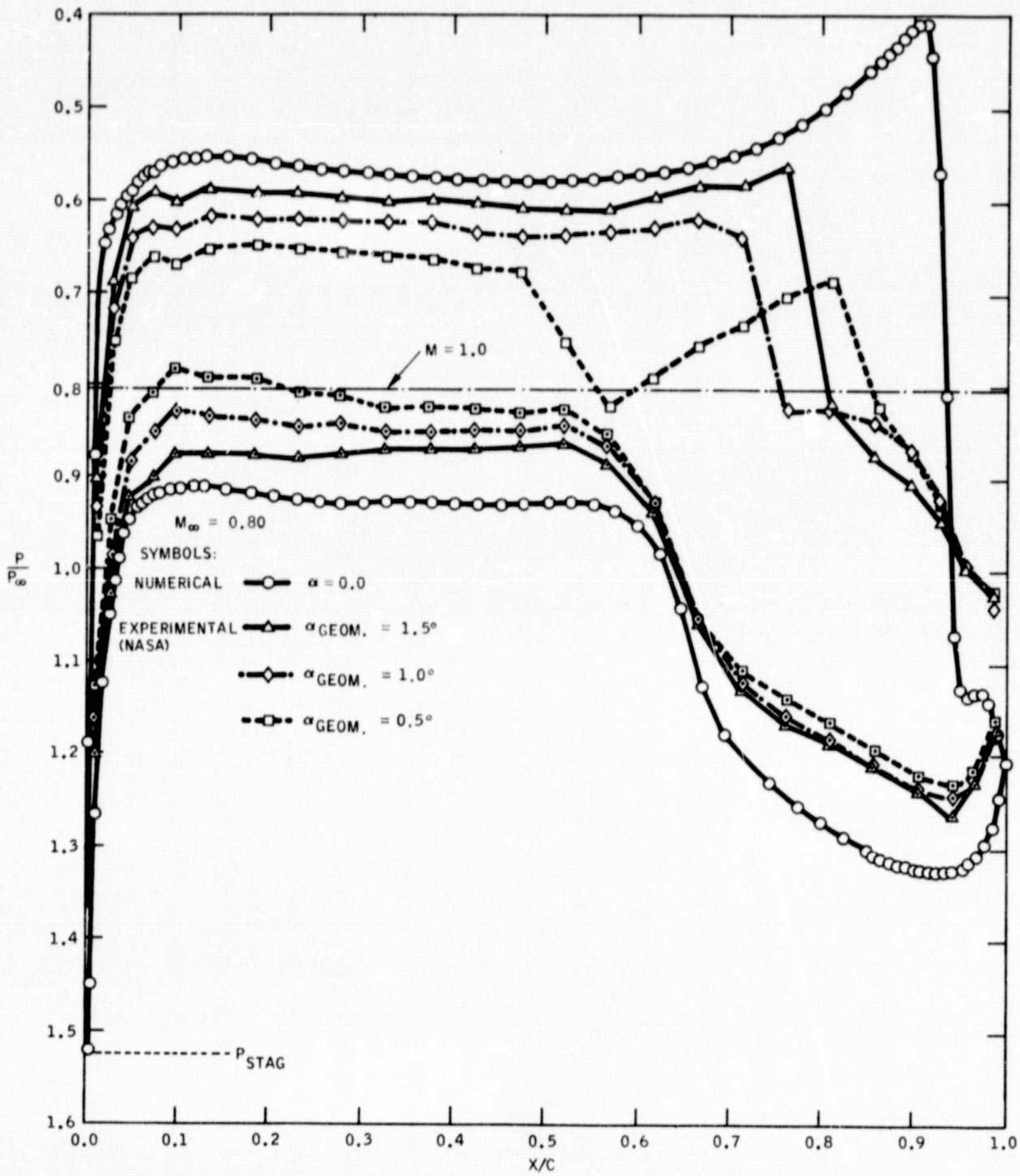

Figure 2. Calculated and Experimental Pressure Distributions for the Basic NASA Profile, $M_\infty = 0.80$

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Response to TD 77-376 October 28, 1976

As of the moment, statistics on the classified documents in this technology are as follows:

28 classified documents (27 conf., 1 sec)
7 are AD's (6 conf., 1 sec)
16 are TM-X's (14 LRC, 2 Ames)
5 are CR's (2 Lockheed, 2 Boeing and 1 Gen. Dyn.)

Amplifying data on the CR's are as follows:

Acc	Sec Class	Report No.	Contract No.	Pub Date	Responsible Center
X75-10263	Conf.	NASA-CR-132712	NAS 1-12325	00-00-75	Langley
X75-10057	Conf.	NASA-CR-132468	NAS 1-12325	00-00-74	Langley
X74-10002	Conf.	NASA-CR-2215	NAS 1-10824	09-00-73	Langley
X73-10303	Conf.	NASA-CR-2214	NAS 1-10824	04-00-73	Langley
X71-10591	Conf.	NASA-CR-111888	NAS 1-9308	00-00-70	Langley

A copy of this correspondence has been furnished to the Classified Supply Activity for use in detecting declassified supercritical wing technology documents. In addition a copy of this TD has been inserted in each case file of the above listed documents.

It should be noted that the 5 CR's were sponsored by Langley under 3 different contracts.

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