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Improved Version of LTRAN2 for Unsteady Transonic Flow Computations

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Introduction

In order to study the aeroelastic characteristics of airplanes flying at transonic speeds, a reliable method of predicting unsteady airloads is required. For a slowly oscillating thin wing section such a prediction can be obtained at relatively low cost using the NASA Ames code LTRAN2.[1] This code solves the low frequency transonic small perturbation (TSP) equation for the velocity potential:

$$[1-M_\infty^2-(\gamma+1)M_\infty^2\phi_x]\phi_{xx}+\phi_{zz}-2M_\infty^2\phi_{xt}=0 \qquad (1)$$

derived at a condition $0[k]=0[\delta^{2/3}]=0\ [1-M_\infty^2]\ll 1$ for reduced frequency $k=\omega c/2U_\infty$, relative airfoil thickness δ, and freestream Mach number M_∞. For an airfoil $z=f(x)$ in unsteady motion $z=h(x,t)$, the low frequency airfoil boundary condition is:

$$\phi_z=f_x+h_x \qquad (2)$$

on the slit $z=0$, $0<x<1$. The condition in the wake ($z=0$, $x>1$) is given by $\Delta C_p=0$, with for C_p the low frequency expression:

$$C_p=-2\phi_x \qquad (3)$$

In applications of the original code,[1] however, Eq. (2) was replaced by the unsteady airfoil boundary condition:

$$\phi_z=f_x+h_x+h_t \qquad (4)$$

This is necessary to describe plunge motions.

In this Note it will be shown that the applicability of LTRAN2 is considerably improved at negligible cost if Eq. (3) is also replaced by the unsteady expression for C_p:

$$C_p=-2(\phi_x+\phi_t) \qquad (5)$$

The wake condition $\Delta C_p=0$ then correctly describes the downstream vorticity transport in the wake at finite (freestream) velocity.

Equations (4) and (5) and the TSP equation (1) with a slightly modified nonlinear term [the coefficient $(\gamma+1)M_\infty^2$ was replaced by $3-(2-\gamma)M_\infty^2$] have become the basis of a new version (LTRAN2-NLR[2]),which resembles a code developed simultaneously and independently at ONERA.[3] This version is intermediate between the original code[1] and the GTRAN2 code developed by Rizetta and Chin.[4] GTRAN2 is based on Eqs. (4) and (5) and the TSP equation (1) with the additional term $-M_\infty^2\phi_{tt}$, and therefore has no low frequency restriction. In LTRAN2-NLR this ϕ_{tt} term was not added because: 1) it is less effective at low reduced frequencies than the time-derivative terms in Eqs. (4) and (5); and 2) it requires a substantial modification of the original code, leading to a significant increase of computational costs.

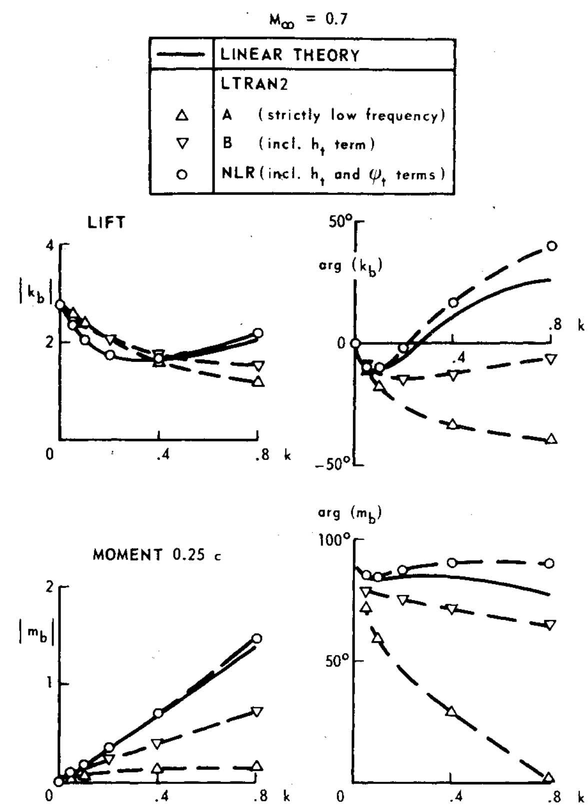

Fig. 1 Unsteady airloads on pitching flat plate showing effect of additional terms.

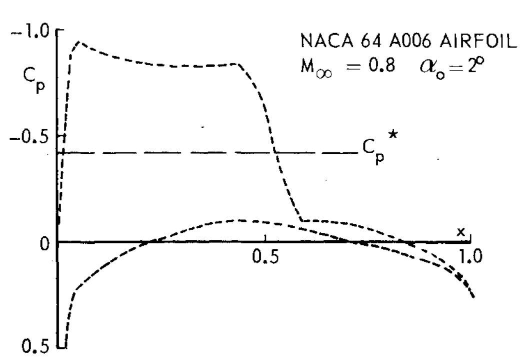

Fig. 2 Steady pressure distribution.

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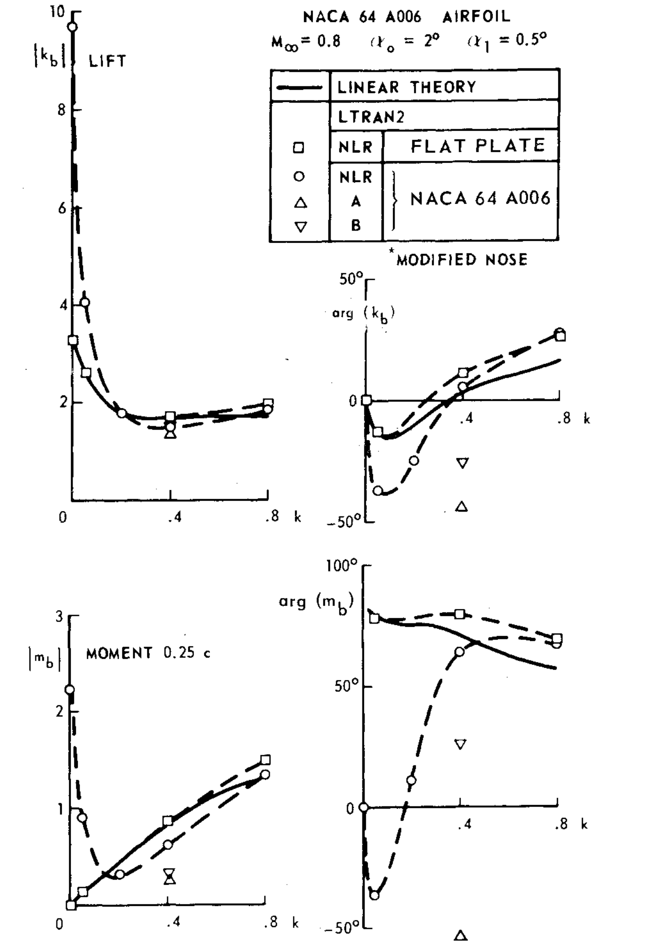

Fig. 3 Unsteady airloads on pitching transonic airfoil effects of 1) additional terms and 2) frequency and thickness.

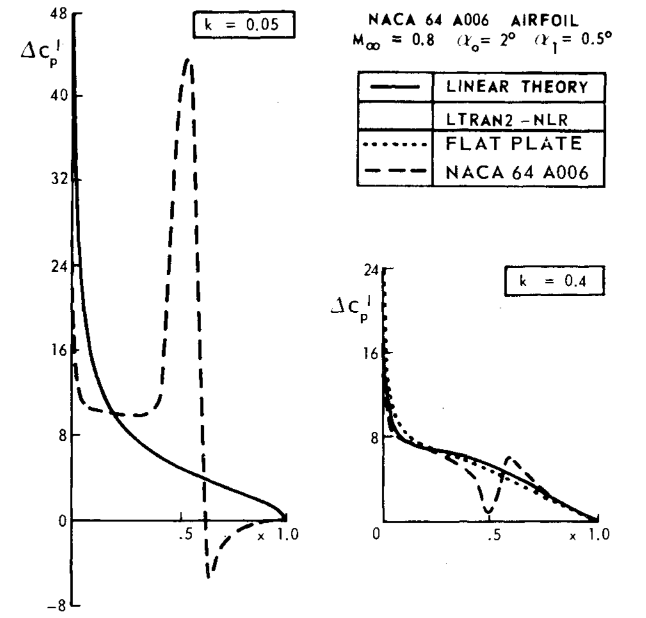

Fig. 4 Unsteady pressure distributions (real part) on pitching transonic airfoil showing effect of frequency and thickness.

The increased applicability of LTRAN2-NLR will be demonstrated for a subsonic and a transonic oscillating airfoil by comparison with results of the original code [Eqs. (1), (3), and (4), version B] and its strictly low frequency form [Eqs. (1), (2), and (3), version A].

Calculations

Unsteady airloads were calculated for a flat plate ($M_\infty = 0.7$ and 0.8, $\alpha_0 = 0$ deg) and a lifting transonic NACA64 A006 airfoil ($M_\infty = 0.8$, $\alpha_0 = 2$ deg, $C_{l_0} = 0.49$), oscillating in pitch about 0.25 chord at reduced frequencies $0 < k < 0.8$. Results at $k = 0$ were derived from steady results calculated at incidences $\alpha_0 \pm 0.1$ deg.

For $k \neq 0$, the unsteady flow was calculated for two cycles (flat plate) or four cycles (NACA64 A006) of a harmonic oscillation, using 120 time steps per cycle. The amplitudes were $\alpha_1 = 0.25$ deg for the flat plate and $\alpha_1 = 0.5$ deg for the NACA64 A006 airfoil. From the instantaneous pressure distributions $C_p(x,t)$, lift coefficients $C_l(t)$, and moment coefficients $C_m(t)$, the first harmonic components (subscript *1*) were determined over the last cycle. The overall coefficients were put into standard AGARD notation ($k_b = C_{l_1}/\pi\alpha_1, m_b = 2C_{m_1}/\pi\alpha_1$). It was verified that significant higher harmonics (indicating nonlinear behavior) occurred only in the transonic results at low reduced frequencies. Compared to the first harmonics, however, the higher harmonics were small, in particular in the lift coefficients.

Results and Discussion

In Fig. 1 the magnitude and phase angle of the coefficients k_b and m_b for the flat plate at $M_\infty = 0.7$, calculated by the various LTRAN2 versions, are compared with linear theory. For this case the nonlinear term in the TSP equation is negligible, hence such a comparison gives insight into the effect of the various unsteady terms. Clearly the NLR version gives the best agreement. The differences, appearing mainly in the phase angle, arise from the low frequency form of the TSP equation. As expected, version B shows greater differences, while version A is obviously valid only at very low frequencies. It can be concluded that the NLR version has enlarged the applicability of LTRAN2 from $k \leq 0.15$ to $k \leq 0.4$, approximately.

The mean steady pressure distribution on the NACA64 A006 airfoil is shown in Fig. 2. The flow exhibits a shock wave near midchord, indicating a strong effect of the nonlinear term in the TSP equation.

Figure 3 shows the magnitude and phase angle of the coefficients k_b and m_b on this airfoil. At $k = 0.4$, results of LTRAN2 versions A and B are compared with the NLR version, showing once again the effectiveness of the various unsteady terms. In Fig. 3 results of LTRAN2-NLR and linear theory for the flat plate are also shown. Compared with these results, the transonic results at low frequencies exhibit a large increase in magnitude and a more negative phase angle. The explanation is the shock wave, which by its motion causes large peaks in the real and imaginary parts of the pressure distribution. This is illustrated in Fig. 4, showing the real part $\Delta C_p' = Re(\Delta C_{p1}/\alpha_1)$ of the pressure difference across the airfoil (the imaginary part behaves in a similar way[2]).

At increasing frequency the amplitude of the shock wave motion decreases, leading to better agreement between results for the flat plate and the transonic airfoil. Simultaneously, the error in the LTRAN2-NLR results increases due to the missing ϕ_{tt} term in the TSP equation. Hence, for approximately $k \geq 0.4$ the use of the low frequency TSP equation is not advisable, and a theory without frequency limitation[4] should be used. In the present case, at $k = 0.8$ even linear theory would be acceptable.

Conclusions

Of the versions of LTRAN2 investigated, the new version including both h_t and ϕ_t terms (LTRAN2-NLR) has the largest low frequency range of applicability (typically $k \leq 0.4$) for unsteady transonic flow computations. Therefore in the low frequency regime the use of this version, which does not require additional computational cost, is recommended. Its improved applicability is of practical importance because the transonic phenomena are strongest at low reduced frequencies and decrease rapidly if the frequency is increased.

Acknowledgements

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J80-192

Review of the Influence of Cooled Walls on Boundary-Layer Transition

20004

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Nomenclature

M	= Mach number
Re	= Reynolds number based on wetted length
$\bar{R}e$	$= Re_{\delta t}/Re_{\delta ta}$
Taw	= surface temperature for an insulated wall, i.e., adiabatic recovery temperature
Tw	= surface temperature
$\bar{T}w$	$= Tw/Taw$
$(U/\nu)_\delta$	= unit Reynolds number at edge of boundary layer

Subscripts

a	= adiabatic wall conditions
s	= complete boundary-layer stabilization
t	= boundary-layer transition
δ	= edge of boundary layer conditions
∞	= freestream conditions

THIS Note presents a review of the influence of cooled walls on boundary-layer transition. The purpose is to clarify a somewhat confused picture which has evolved because the relevant data have been collected piecemeal for roughly three decades and are characterized by some gaps in coverage as well as disagreements. Although a completely coherent view is not developed here, it is believed that some improvement in understanding is gained by looking at the data in as general a framework as possible.

Figure 1 is the basis for most of the following discussion. It displays the available experimental data on the influence of cooled walls on the Reynolds number of boundary-layer transition under supersonic flow conditions.[1-8] Only results which include the condition $Tw = Taw$ are presented because of a need to normalize the data. In selecting data to be presented, it was necessary to accept sources where different methods of determining $Re_{\delta t}$ were used, and presumably different combinations of flow disturbances were present in the various wind tunnels. Only data for zero pressure gradient are represented, but both sharp cones and sharp plates or wedges are included. At a given M_δ, data for a constant $(U/\nu)_\delta$ were utilized, but different $(U/\nu)_\delta$ had to be accepted at different M_δ. There is evidence that the influence of wall cooling varies somewhat with unit Reynolds number at a given Mach number (cf. Refs. 5 and 7). Freestream temperature also generally varied with Mach number. A significant step toward eliminating the effects of such variables was taken when the ratio $\bar{R}e$ was adopted. The use of this ratio also makes possible a more simple and orderly display of the various data included in Fig. 1.

Unfortunately, the adiabatic wall case $\bar{T}w = 1$ often is not included in investigations of higher Mach number flows, which prevents inclusion of such data in Fig. 1. Those cases are discussed later. In regard to Ref. 5, the data do not extend to $\bar{T}w = 1$, but extrapolation to obtain $Re_{\delta ta}$ seemed safe enough to justify that being done. In Ref. 8, only a few points are given and a claim could be made that no $\bar{T}w$ effect is evident.

In addition to the Ref. 4 data shown in Fig. 1, there are other experimental results which apparently exhibit no $\bar{T}w$ effect. Because $Re_{\delta ta}$ is not available in those cases, they are not represented in Fig. 1, but they must be recognized at this point. References 9-11 are the principal ones that the writer is aware of. Sanator et al.[9] covered the range $0.08 \leq \bar{T}w \leq 0.4$ at $M_\delta = 8.8$ and found no significant change in $Re_{\delta t}$ on a sharp cone. The adiabatic wall condition could not be included. Deem and Murphy[10] also report no effect for $0.2 \leq \bar{T}w \leq 0.8$ at $M_\delta = 10.2$, and Everhart and Hamilton[11] found no effect for $0.4 \leq \bar{T}w \leq 0.6$ at $M_\delta = 8.9$.

In contrast to the findings of the last three sets of investigations, there are references showing marked $\bar{T}w$ influence at hypersonic Mach numbers. Stetson and Rushton[12] reported a pronounced decrease of $Re_{\delta t}$ on a cone as $\bar{T}w$ decreased in the region $0.2 \leq \bar{T}w \leq 0.6$ at $M_\delta = 4.8$. This rather strongly suggests that a transition reversal would have been found somewhat above $\bar{T}w = 0.6$ if it had been feasible to explore higher $\bar{T}w$. Mateer[13] has reported a similar decrease of $Re_{\delta t}$ with decreasing $\bar{T}w$ for $0.2 \leq \bar{T}w \leq 0.4$ and an indication of a reversal at $\bar{T}w \approx 0.2$, so that $Re_{\delta t}$ began increasing again as $\bar{T}w$ was lowered below 0.2. His data

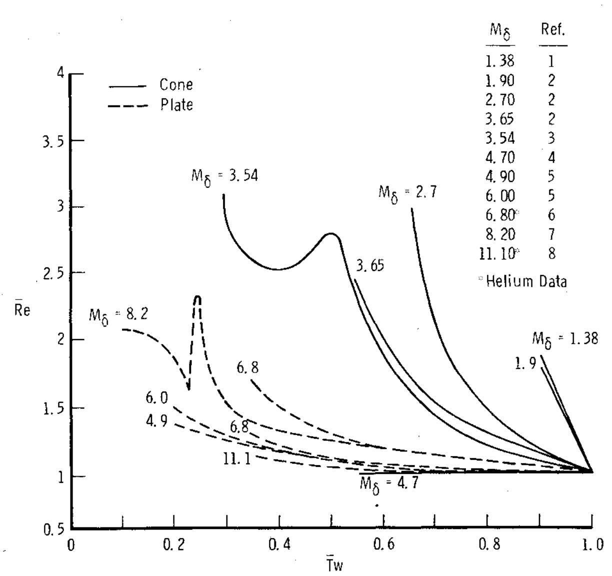

Fig. 1 Experimental effect of wall cooling on transition Reynolds numbers at various Mach numbers.

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Index category: Boundary-Layer Stability and Transition.

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