Effect of Edge-Tone Noise on Supercritical Airfoil Data

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AN investigation has recently been carried out in the National Aeronautical Establishment (NAE) 0.38×1.5 m two-dimensional test section[1] to determine:

1) The possibility of edge-tone noise suppression by overlaying the perforated floor and ceiling with a fine gauze or screen.

2) The change in wall interference characteristics due to the application of point 1.

3) The effect of edge-tone noise suppression on two-dimensional airfoil test data.

The investigation was performed with a 0.254 m chord model of the BGK No. 1 airfoil and covered the Mach number range of 0.3-0.8 at chord Reynolds numbers of $10\text{-}21 \times 10^6$. A full account of the investigation is given in Ref. 2. This Note deals primarily with selected data related to point 3. However, first some brief comments on points 1 and 2.

The fine gauze technique, first demonstrated by Vaucheret,[3] was found to be both a practical and an effective means for edge-tone suppression, as demonstrated in Figs. 1 and 2, respectively. It was also found that the presence of the gauze, although reducing the geometric porosity from 20.5 to 8%, had a very minor effect on the wall interference characteristics. Measurements on the BGK No. 1 airfoil comprised balance, wake, and surface pressure. Only the latter two are discussed here.

The data depicted in Fig. 2 were obtained by a sidewall-mounted differential pressure transducer, with the reference side detuned and connected to the plenum chamber. The frequency response of the active side was flat up to 16 kHz. The data are presented in the form of the root mean square of the fluctuating wall static pressure coefficient, $C_{P_{RMS}}$, referenced to the mean value of the freestream dynamic pressure. The significant reduction in noise level due to the edge-tone suppression is clearly evident. In fact, at the higher Mach number the measured noise level is reduced to a level comparable to that inherent in the wall boundary layer itself. For a well-developed turbulent boundary layer, the wall $C_{P_{RMS}}$ (referenced to the dynamic pressure at the edge of the boundary layer) is known to be about 0.6-0.7%.

To what degree then are model test data influenced by this improvement in flow quality? The obvious thought that comes to mind is that the edge tones could excite the Tollmien-Schlichting waves and thus cause early transition. A cursory investigation reveals that, at the Reynolds numbers here of interest, the "dangerous" frequencies are an order of magnitude higher than the edge-tone frequencies. Thus the edge tones are not expected to influence the boundary-layer transition. However, we may speculate that they could interfere with, for instance, a turbulent boundary layer close to separation, although the mechanism in such a case would be far from clear.

We shall first examine drag data obtained from wake pressure measurements. In Fig. 3 we have plotted the drag coefficient vs freestream Mach number, both corrected for wall interference effects, for two values of the normal force coefficient C_N, 0.3 and 0.6. The data points are obtained by a

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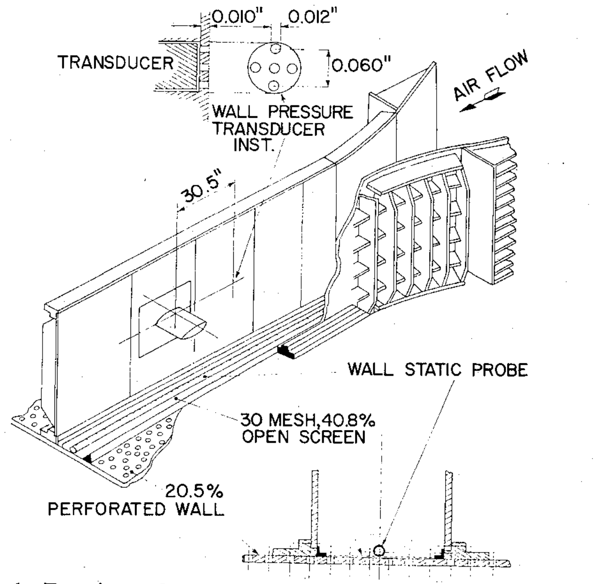

Fig. 1 Experimental arrangements in NAE two-dimensional test section.

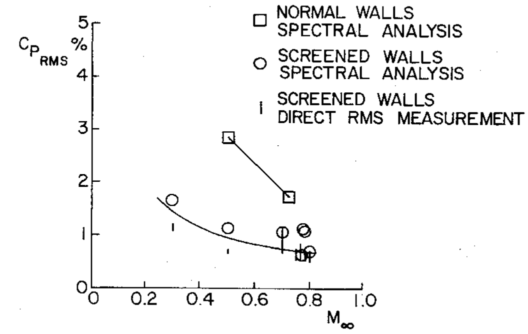

Fig. 2 Overall noise level.

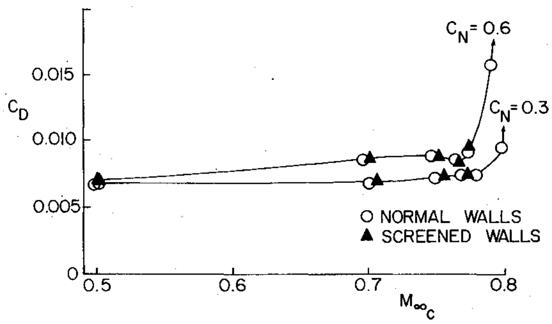

Fig. 3 Wake drag data for BGK No. 1 airfoil.

small degree of interpolation from drag polar curves. There is clearly no measurable effect caused by the presence of edge tones. This then must be taken as indirect evidence that the presence of edge-tone noise has little, if any, effect on boundary-layer transition at these Reynolds numbers, corroborating the above reasoning.

This being the case, we can not expect to find significant differences in pressure distributions. Some difficulties are encountered in comparing pressure distribution because of inevitable small differences in Mach number and angle of attack. Nevertheless, the results show that the edge-tone noise has no measurable effect in subcritical flow. In supercritical flow with a shock wave, however, there is a consistent small downstream shift in the upper surface shock location when the edge tones are suppressed. This effect is present when the

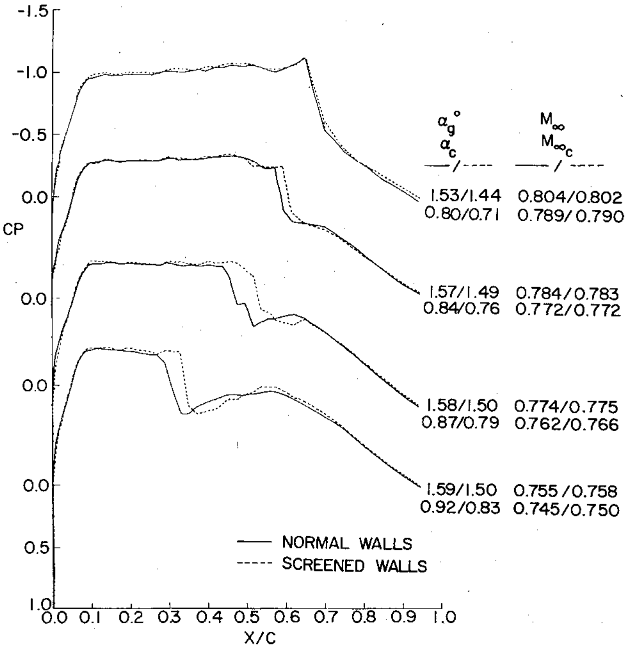

Fig. 4 Upper surface pressure distributions for BGK No. 1 airfoil.

shock is located over the "flat" portion of the airfoil, say 20-65% chord. Sample results of the upper surface pressure distribution for a fixed nominal incidence of 1.5 deg at various Mach numbers are given in Fig. 4. The wind-tunnel freestream Mach number M_∞, the geometric incidence α_g, and their values M_{∞_c} and α_c corrected for wall interference following Mokry,[4] are listed in the figures. The differences in M_{∞_c} for the two lower cases are in the direction that would cause the shock wave to move further downstream for the screened wall case. In the one case where the two M_{∞_c} are identical, there is still a noticeable shift in shock location. At the highest Mach number the shocks coalesce. In all cases the differences in the angle of attack, whether corrected or not, are in the direction that would suggest a reversal in the observed shock positions.

Computational results, using the SCW II method,[5] have been obtained to determine the sensitivity of shock position to small changes in Mach number and angle of attack. These may or may not be all that meaningful because of an inexact treatment of the shock/boundary-layer interaction. In any case, they show in general that the observed differences in shock location cannot be explained by the small differences in Mach number and angle of attack. The computational results are presented and discussed in more detail in Ref. 2. Thus we are led to conclude that at some supercritical conditions the shock position can be affected by the presence of edge-tone noise. The mechanism of such an effect is far from clear. One may surmise that the edge-tone noise interacts in some way with the disturbed boundary layer in the shock/boundary-layer interaction zone, so as to cause the observed effect.

As a final comment, it is worth relating the present findings to those of Dougherty and Steinle.[6] Using the so-called AEDC 10 deg cone to determine transition Reynolds number in various wind tunnels, they found that by covering the perforated walls of the AEDC 16T wind tunnel with tape, thus eliminating the edge-tone noise, the transition Reynolds number increased compared to that for the open perforated walls. This appears to be in conflict with the present finding.

However, one can reason that they are not necessarily in conflict. In their case the unit Reynolds number was 6.6×10^6/m, which is an order to magnitude smaller than for the present NAE test ($33\text{-}69\times10^6$/m). This lower unit Reynolds number implies a correspondingly lower "dangerous" frequency for the Tollmien-Schlichting (T-S) waves. But, the edge-tone center frequency in the 16T wind tunnel is also much lower than in the NAE wind tunnel (600 and 7400 Hz, respectively, at $M=0.75$). In neither case are these frequencies close to the estimated T-S dangerous frequencies, which are an order of magnitude higher. It is still possible that higher harmonics of the edge tones may excite the T-S waves. The increase in transition Reynolds number attributed to edge-tone suppression in the 16T wind tunnel is only 10-15% in $M=0.7\text{-}0.8$ range. Such an increase would hardly be detectable in the NAE two-dimensional drag data. Transition location at the high Reynolds number NAE investigation is known to be close to the leading edge. A shift in transition from 10 to 11% chord, say, would result in less than 0.0001 decrease in the drag coefficient, which is within normal data scatter. Thus, the results from the two investigations may not be conflicting.

Furthermore, they both show, in the author's opinion, that the edge-tone noise, in spite of its very significant contribution for the overall noise levels in the two wind tunnels, has a surprisingly small effect on aerodynamic measurements, and demonstrably so only at high subsonic speeds.

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Correlation of Hypersonic Stagnation Point Heat Transfer at Low Reynolds Numbers

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Nomenclature

a	= sound velocity
C^*	$=\mu^* T_\infty / \mu_\infty T^*$
CH	= Stanton number, $\dot{q}/\rho_\infty u_\infty (H-H_w)$
H	= altitude or total enthalpy
Kn	= Knudsen number
Kr^2	= Cheng's parameter of Eq. (7)
$\dot{q}$	= heat-transfer rate
Re	= Reynolds number

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