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Influence of Sound upon Separated Flow over Wings

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Introduction

A TECHNIQUE for increasing the lift coefficient and stall angle and decreasing the drag coefficient of a wing at moderate Reynolds numbers by radiating the airstream with sound is described. The sound waves cause premature boundary-layer transition, which reduces separated flow regions and suppresses stall. The technique is expected to be most useful at low Reynolds numbers where the separation of a laminar boundary layer normally would occur.

It has long been known that the turbulence level in a wind tunnel has a strong influence on the transition Reynolds number for the flow about a particular body, such as a sphere. Schubauer and Skramstad, in their classic work on boundary-layer transition on a flat plate,[1] were perhaps the first to recognize that sound of particular frequencies and intensities can also influence the transition process. Most succeeding sound studies have either involved an examination of the influence of sound on the transition process or the development of techniques to prevent sound from causing premature transition in laminar boundary layers on wings, thereby increasing the skin friction drag. Only recently have several investigators realized that sound can be used to control the flow about wings, resulting in an increase of the lift coefficient at high angles of attack. Preliminary measurements of the influence of external sound on wing properties at high angles of attack will be presented in this Note.

Previous Work on Audio Boundary Layer Control

Studies of the influence of sound on boundary-layer transition were performed by Schubauer and Skramstad,[1] Bergh,[2] Boltz, et al.,[3] Brown,[4,5] Jackson and Heckl,[6] Knapp and Roache,[7] and Spangler and Wells.[8] Pfenninger and Reed,[9] on the other hand, were concerned with determining the maximum sound levels which could be allowed and still maintain laminar flow on wings with suction boundary-layer control. Their studies were initiated in the early 1950's when it was noticed that sound propagating through the suction ducts would initiate boundary-layer transition.

These studies have led to a number of conclusions concerning the influence of sound on boundary layer transition. 1) Sound can initiate premature transition, causing the position of transition to move upstream.[1,3] This will happen when flows with or without a pressure gradient are radiated.[3–5] Sound is effective in causing transition up to $Re = 10^7$ (Ref. 9) and $M = 0.5$ (Ref. 3). 2) Only certain sound frequencies can initiate premature transition, the most effective ones being those on Branch II of the neutral stability curve.[1,3] 3) The amount of sound influence depends upon the sound intensity in a manner similar to the influence of the level of turbulence on transition.[1] 4) Both external sound and sound emitted internally through holes in the body surface will initiate transition.[1,9,10]

Recently several investigators have noted benefits which could result from the use of sound to control the boundary layer and, in particular, the stall or separation phenomena. Chang[11] noted that sound could be used to reduce airfoil drag by up to 20%, the power saving due to drag reduction being as much as 19 times greater than the amount of sound power required. The drag reduction had a smooth and continuous dependence upon Strouhal number. This work was performed at a rather low Reynolds number (8×10^4). Brown,[4,5] at Notre Dame, noticed that shear layer vortices could be controlled by sound within rather wide limits. He demonstrated that sound of proper frequency could be used to close the wake behind spheres,

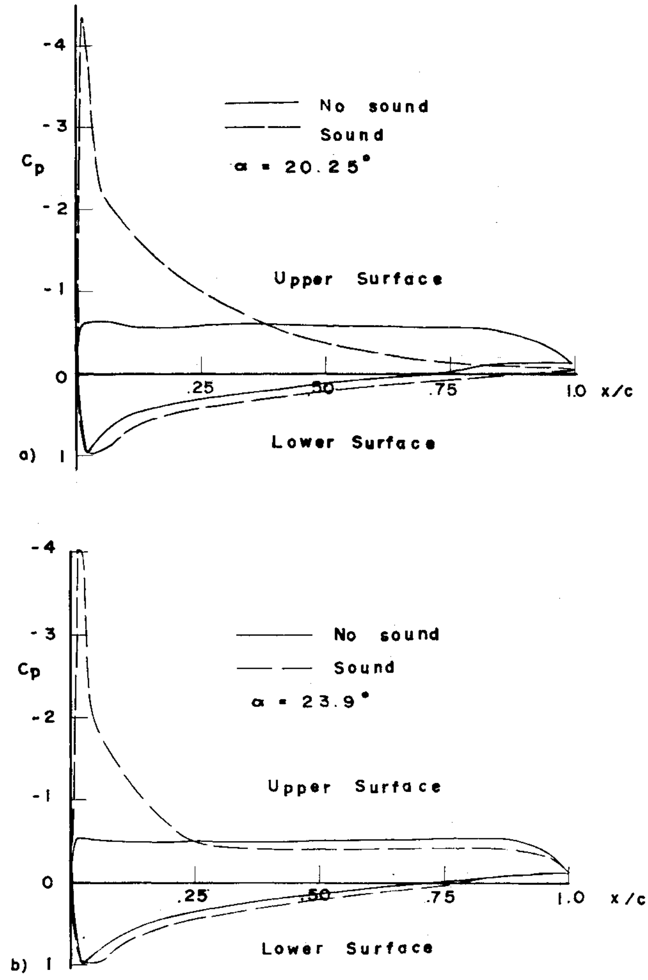

Fig. 1 Pressure distributions around airfoil with and without sound, $Re = 5.3 \times 10^5$. a) No sound properties: $C_L = 0.620$, $C_D = 0.247$, $C_M = -0.089$; sound properties: 7996 Hz, 88 db, $C_L = 0.844$, $C_D = 0.127$, $C_M = 0.042$. b) No sound properties: $C_L = 0.567$, $C_D = 0.283$, $C_M = -0.083$; sound properties: 2401 Hz, 134 db, $C_L = 0.832$, $C_D = 0.233$, $C_M = -0.042$.

Received July 10, 1974; revision received September 6, 1974. This work was supported by the National Science Foundation under Grant Gk-42133.

Index categories: Boundary-Layer Stability and Transition; Aircraft Aerodynamics.

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Table 1 Effect of increased sound pressure level on wing properties[a]

db	C_L	C_D	C_L/C_D	C_M	Power reduction ratio
100	0.820	0.125	6.56	0.012	3380.0
111	0.836	0.123	6.80	0.025	274.0
115	0.832	0.126	6.60	0.027	108.0
121	0.829	0.133	6.23	0.017	28.3
124	0.838	0.133	6.30	0.029	12.8

[a] $\alpha = 20.25°$, $f = 2000$ Hz, $U = 14.8$ msec (48.5 fps). No sound properties: $C_L = 0.572$, $C_D = 0.228$, $C_M = -0.073$.

greatly reducing the profile drag, and to eliminate airfoil stall at moderate Reynolds numbers. Unfortunately, most of his work is in the form of smoke visualization photographs and no details have been published.

Results

Presently an investigation is underway to obtain more quantitative information about the effect of external sound upon the flow about a static wing at high angles of attack. The wing has an NACA 2412 section, with a one foot chord, three foot span, and end plates. The static pressure was measured at 32 surface locations at mid span using an inclined manometer board. The surface pressures were numerically integrated to obtain the lift coefficient (C_L), the drag coefficient (C_D), and the quarter-chord moment coefficient (C_M). Tufts were also used to help assess the flow state.

The wing was placed in the open test section of the University of Texas subsonic wind tunnel. This tunnel has a test section measuring 55.9 cm × 91.4 cm (22 in. × 36 in.) and a turbulence level of $T = 0.005$, where T is defined as the rms streamwise fluctuating velocity component divided by the mean velocity. The sound speaker (Dukane, Model 5a 410, 30 w) was placed downstream and above the wing, with the horn pointed toward the center of the upper wing surface. Single frequency sound, produced by amplifying the signal from an audio oscillator, was used to excite the flowfield. The sound pressure level (SPL) at the wing surface was measured with a condensor microphone and the input power to the speaker was monitored. The actual incoming wind tunnel speed was measured with a hot wire anemometer, but no other tunnel corrections were made to the measured wing properties.

The wing was placed at two angles above the stall angle,

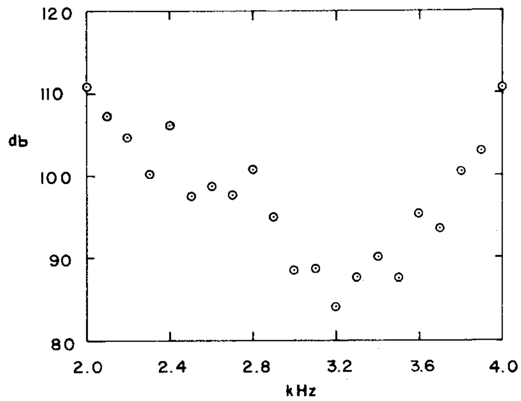

Fig. 2 Sound pressure level required to fully attach the flow over an NACA 2412 wing as a function of sound frequency; $R = 2.9 \times 10^5$, $\alpha = 20.25°$.

Table 2 Average results of sound experiments [Re (based upon chord) $= 2.5 \times 10^5$ to 5.3×10^5]

a) No sound properties:

α	C_L	C_D	C_L/C_D	C_M
20.25	0.615	0.248	2.48	−0.087
23.9	0.531	0.262	2.03	−0.074

b) Sound properties:

α	C_L	C_D	C_L/C_D	C_M	db	$\Delta U_{rms}/U$
20.25	0.831	0.126	6.57	0.034	84 to 123	5×10^{-5} to 5×10^{-3}
23.9	0.813	0.220	3.69	−0.033	115 to 134	2×10^{-3} to 1×10^{-2}

c) Comparison of sound and no sound properties:

α	$\frac{C_L \text{ (sound)}}{C_L \text{ (no sound)}}$	$\frac{C_D \text{ (sound)}}{C_D \text{ (no sound)}}$
20.25	1.35	0.511
23.9	1.53	0.840

Table 3 Power reduction possible with sound

Velocity (msec and fps)	Frequency Hz	db	C_L	C_D	Power reduction ratio
		a) $\alpha = 20.25°$			
12.6(41.5)	1000	109	0.784	0.113	79.3[a]
		119	0.787	0.113	6.2[b]
14.8(48.5)	3000	101	0.780	0.121	390.0[a]
		115	0.839	0.128	12.8[b]
19.4(63.6)	2839	106	0.753	0.122	1395.0[a]
22.2(72.8)	1033	117	0.848	0.140	77.1[a]
27.3(89.5)	7996	88	0.844	0.127	1778.0[a]
		b) $\alpha = 23.9°$			
12.6(41.5)	739	123	0.713	0.162	0.86[c]
14.6(48.0)	1208	115	0.752	0.187	1.99[c]
19.5(64.0)	2289	127	0.676	0.188	4.67[c]
22.4(73.5)	1494	119	0.665	0.197	5.24[c]
27.0(88.5)	2017	134	0.739	0.213	10.0 [c]

[a] Flow 2/3 attached. [b] Flow fully attached. [c] Flow 1/3 attached.

which is about 12° for this wing at the Reynolds numbers tested. For given flow conditions and sound frequency, the SPL was increased until suddenly partial attachment occurred over the wing. At $\alpha = 20.25°$ about 2/3 attachment occurred initially while at $\alpha = 23.9°$ only the front 1/3 of the wing was initially attached (see Fig. 1). Once partial attachment occurred the SPL could be greatly decreased, sometimes by as much as 10 db, and the wing properties (C_L, C_D, C_M) would not change. Also, increasing the db level to cause greater attachment changed the wing properties only negligibly. For the case illustrated in Table 1, 100 db was required for an initial 2/3 attachment whereas 124 db was required for full attachment. However, the lift and drag coefficients changed only slightly. In this table the power reduction ratio is the ratio of the power saved due to drag reduction divided by the total speaker power. This ratio obviously is very dependent upon the speaker efficiency but gives an indication of the benefits to be gained from the technique.

The minimum SPL required to fully attach the flow for one flow condition as a function of sound frequency is shown in Fig. 2. Although stall suppression is a continuous function of frequency, as seen by Chang,[11] suppression at certain frequencies is much easier than at others. It is believed that these are the frequencies that assist the linear boundary-layer instability whereas large amplitude sound waves can bypass the linear stability mechanism for transition. Similar results were reported by Pfenninger and Reed.[9]

A summary of the best results obtained thus far under various conditions is given in Table 2. ΔU_{rms} is the velocity fluctuation produced by the sound wave if it can be approximated by a plane wave.[3] The relative velocity fluctuation produced by the sound is generally lower than the tunnel turbulence level (5×10^{-3}). Much more intense sound is required to attach the flow at $\alpha = 23.9°$ than at 20.25°. Also, the power reduction ratio is greatly decreased (see Table 3 for typical values) at the higher angle.

At $\alpha = 20.25°$ the tunnel noise was at times sufficient to produce attachment at the higher Reynolds numbers. The tunnel noise was concentrated at 95 Hz and 120 Hz, with levels of 90 and 93 db, respectively, at 90 fps (105 db total SPL). However, it appeared that only 88 db was required to cause stall suppression (see Fig. 1a). As mentioned previously, single frequency sound is much more likely to cause boundary-layer transition than a continuous sound spectra.[1,3]

The SPL's described in this Note are very high in some cases. However, the tests performed at the Air Force Academy[10] indicate that much lower levels are required to cause boundary-layer attachment if internal sound is used. Future tests will include the use of internal sound.

Conclusion

External sound can cause partial reattachment of the flow about a stalled airfoil, greatly increasing the lift and decreasing the pressure drag. The amount of influence of the sound is dependent upon the sound frequency and sound pressure level, and a minimum SPL occurs for some frequencies.

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Interpretation of Merged Layer Behavior for Wedges

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PROCEEDING downstream from a leading edge there are idealized regimes referred to as kinetic, merged, strong and weak interaction, some of which may be of appreciable extent in a hypersonic flow. The over-all extremes are the well known free molecule flow at the leading edge and the boundary layer with an adjacent inviscid shock layer sufficiently far downstream.

The merged layer portion has received particular attention for sharp flat plates[1,2] in hypersonic flow and Shorenstein[3,4] has provided an extension to wedges and cones which shows rather good agreement with experimental data. He used a local similarity analysis to match both flow variables and their normal gradients at the "interface" between the adjacent shock and boundary-layer structures. The approach is essentially an extension of an earlier flat plate study[2] which correctly indicated departures from strong interaction behavior near a leading edge.

Implicit in Shorenstein's wedge analysis[3] is a generalization of the plate behavior such that a strong interaction always follows the merged region. The generalization proves to be somewhat misleading and the present purpose is a clarification of that point for the specific parameter ranges that were considered as well as qualitatively in general. In view of some important numerical inaccuracies, a secondary purpose relates to the need for caution when applying the summary correlations[3] that were suggested.

A tabulation of results was not included in either Refs. 3 or 4. The exact solutions for pressure and heat transfer were shown graphically for a range of wedge (half) angles, $2° \leq \theta_b \leq 20°$, and a single Mach number, surface temperature combination ($M_\infty = 20$, $T_b/T_o = 0.06$, T_o being the stream stagnation temperature) over the interval $40 \leq \bar{\chi}_\infty \leq 400$. The interaction parameter $\bar{\chi}_\infty \equiv M_\infty{}^3(C/Re)_\infty{}^{1/2}$ is defined here in terms of freestream conditions, the slant length along the surface in the Reynolds number, Re_∞, and the Chapman-Rubesin constant, $C_\infty = (\mu_b T_\infty)/(\mu_\infty T_b)$. In place of tables or additional graphs an empirical correlation was provided in summary of other numerical results. An accuracy of 5% was claimed within the intervals $20 \leq M_\infty \leq 25$, $0.06 \leq T_b/T_o \leq 0.15$, and $2° \leq \theta_b \leq 20°$.

Received July 15, 1974; revision received September 4, 1974. This work was sponsored by the Department of the Air Force.

Index categories: Supersonic and Hypersonic Flow; Boundary Layers and Convective Heat Transfer—Laminar.

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