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Poststall Behavior of a Wing Under Externally Imposed Sound

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Nomenclature

$\mathcal{R}$	=	aspect ratio of wing
C_D	=	total drag coefficient of wing
C_L	=	lift coefficient of wing
C_l	=	section lift coefficient of wing
f	=	acoustic frequency
Re	=	Reynolds number of flow based on wing chord, wind-tunnel airstream velocity, density, and dynamic viscosity
Sr	=	Strouhal number based on wing chord, wind-tunnel freestream velocity, and acoustic excitation frequency
α	=	angle of attack or incidence
δ	=	Glauert coefficient

I. Introduction

LIFT force can generally be augmented by an increase in wing area, angle of incidence, camber, or artificial circulation for fixed air properties and freestream. There is however, a limit as to how far these features can be exploited without encountering what is known as wing stall. Wing stall not only decreases aerodynamic efficiency of a wing, but can pose severe hazard during flight. Poststall flow behavior on a wing, therefore, is an important area of research in aerodynamics and forms the basis of this study.

Improvement of stall behavior of a wing generally involves manipulation of boundary-layer flow on its surface. In the subsonic and transonic speeds, pressure gradients can be particularly strong, and a wing can only continue to generate lift successfully beyond stall incidence, if boundary-layer separation is either delayed or avoided. Various techniques such as suction or blowing or circulation control[1–3] have been used with varying degrees of success for a long time. The concept of controlling boundary-layer separation by acoustic excitation has occurred much later and remains least developed. It was probably the finding of Spangler and Wells[4] that sound has a significant effect on boundary-layer transition that has led to various attempts[5–8] to control or suppress laminar flow separation and induce turbulent flow without going through the unstable phase of transition flow. Most of these works have involved both internal as well as external excitation, but are mainly limited to investigation of two-dimensional flows such as those encountered on flat plates or airfoils. In the present study, an attempt has been made to look at the poststall behavior of a low aspect ratio wing subjected to externally excited sound energy.

II. Experiment

A 30-in.- (0.76-m-) diam open return, low-speed open test section of 0.2% turbulence intensity wind-tunnel of the Aerodynamics Laboratory of the University of New South Wales was used in the experiments. A schematic of the experimental setup and instrumentation is given in Fig. 1. The wind tunnel was powered by a 15-bhp compound wound dc variable speed electric motor driving a fan to give a freestream velocity range of 0–30 m/s. A manometer connected to a pitot static tube was used to record the wind speed.

The acoustic signals were generated using a sine wave generator and a speaker of 15-Ω impedence, which was powered by a 21-Ω rated load amplifier of 120-W capacity. The amplifier had no gain control, and power had to be varied by varying the input voltage.

A constant section symmetric NACA 0012 half-wing of effective aspect ratio of 4 was held on one end to a two-axis force balance platform while the other end was left free. The balance was equipped with two load cells of 100-N force transducers of ±0.1-N force resolution, one measuring side or lift force and the other in the direction of the flow or the drag force. Flow visualization was carried out using smoke illuminated by a laser light sheet that used a 10-mW He–Ne laser as a light source.

Experiments were performed at 16-, 17-, 18-, and 19-deg angle of incidence using externally excited sound frequencies that ranged between 100 Hz and 3 kHz and at three Reynolds number flows of 0.7×10^5, 1×10^5, and 2.6×10^5, respectively. Flow visualization, however, was restricted to the lowest Reynolds number of 0.7×10^5 to obtain clearer photographs.

III. Results and Discussion

A. Qualitative Results: Flow Visualization

Laser light sheet visualizations of flow over the wing were carried out at the 16-, 17-, and 18-deg angle of incidence for the unexcited and excited conditions, respectively, at the Reynolds number of 0.7×10^5. Figures 2 and 3 show two visualization obtained at 16- and 18-deg angle of incidences, respectively. In Figs. 2 and 3, flow separation on the wing from the leading edge under unexcited (Figs. 2a and 3a) and its subsequent suppression under externally imposed sound (Figs. 2b and 3b) are clearly visible. With increasing angle of attack, although the streamlines over the upper surface were observed to remain parallel to the contour of the top surface in the first-half or forebody of the wing, they appear to gradually diverge away in the second-half of the wing suggesting the onset of flow separation in this region.

B. Quantitative Results

1. C_L and C_D vs Acoustic Frequency

Figure 4 shows the variation of C_L and C_D with changes in acoustic frequency at the four incidences of 16-, 17-, 18-, and 19-deg at $Re = 0.7 \times 10^5$, 1×10^5, and 2.6×10^5, respectively. The solid lines without symbols represent the C_L and C_D values under the no excitation condition.

a. Results at $Re = 0.7 \times 10^5$. At this Reynolds number Re, externally excited imposed sound produces significant improvements on the C_L and C_D values at all of the four angles of incidence (Fig. 4a). However, there are some differences that can be noted. At the lower angles of incidence, that is, at $\alpha = 16$, 17, and 18 deg, the improvements can be observed over a wide frequency range, $200 < f < 2000$ Hz, or an equivalent Strouhal number range of $5 < Sr < 45$, with a 20–30% increase in C_L values and a 20–30% decrease in C_D values resulting in a near doubling of the aerodynamic efficiency over the unexcited values. At $\alpha = 19$ deg, the effect of sound is observed in a smaller frequency range of $200 < f < 1200$ Hz, and the improvements are less pronounced with up to 15% increase in lift and 10% decrease in drag being observed.

b. Results at $Re = 1 \times 10^5$. At $Re = 1 \times 10^5$ (Fig. 4b), there is a shift of the curves compared to those observed at $Re = 0.7 \times 10^5$,

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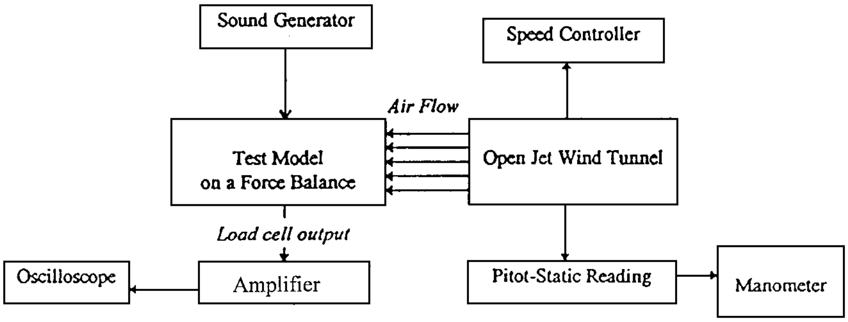

Fig. 1 Schematic of experimental setup and instrumentation.

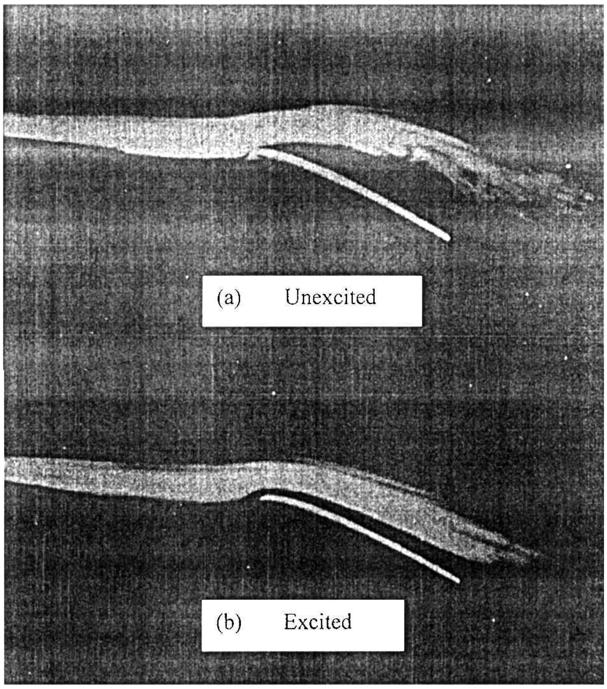

Fig. 2 Flow visualization: □ = 16 deg and $Re = 0.7 \times 10^5$.

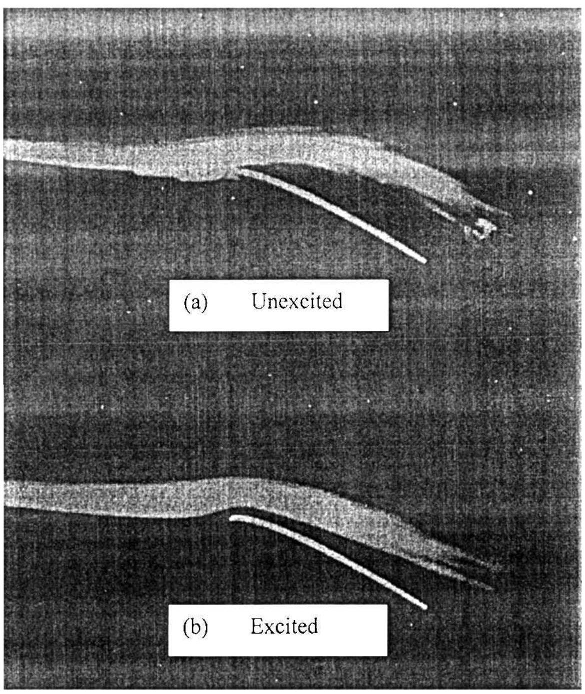

Fig. 3 Flow visualization: □ = 16 deg and $Re = 0.7 \times 10^5$.

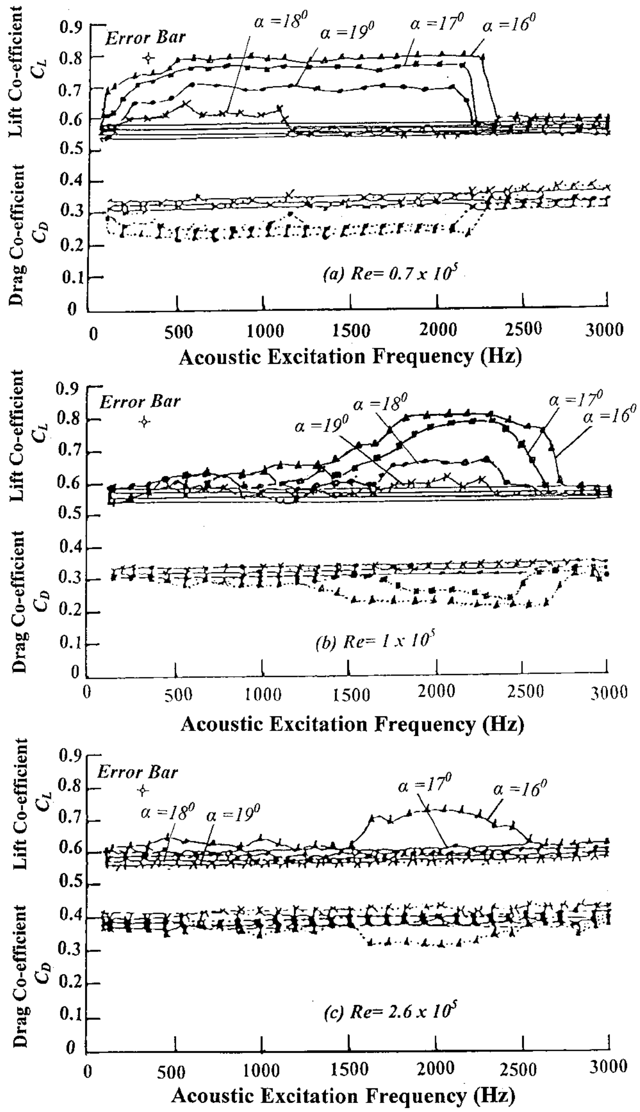

Fig. 4 Lift and drag coefficients vs acoustic excitation frequency: C_L and C_D are represented by solid and dotted lines, respectively, in the top and bottom halves of each graph; ▲, ■, ●, and × denote the results for $\alpha = 16$, 17, 18, and 19 deg, respectively.

and the improvements in C_L and C_D values seem to take place in the frequency range of $1400 < f < 2600$ Hz, or $22 < Sr < 40$. The improvements are observed in the $16 < \alpha < 18$ deg range to the order of up to 20–30% increase in C_L values and up to 30% decrease in C_D values over their corresponding unexcited values. This would again translate to an approximate doubling of the lift-to-drag ratio of the wing under acoustic excitation. Results at $\alpha = 19$ deg, do not show much improvements in C_L and C_D values over their corresponding unexcited values.

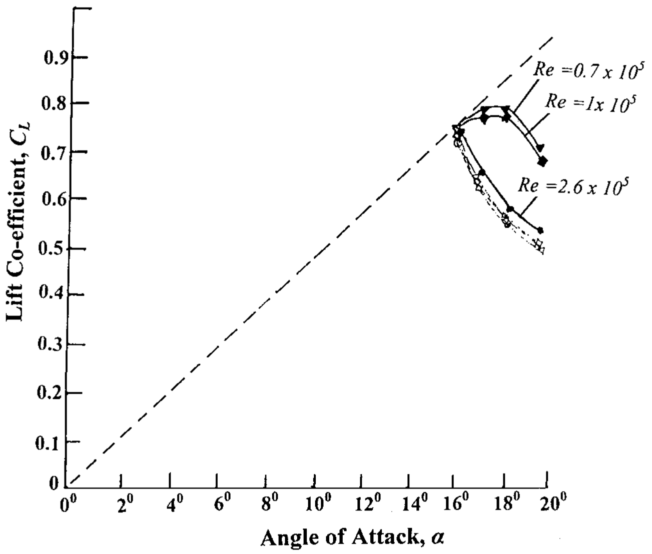

Fig. 5 Lift coefficient C_L vs angle of attack, □: acoustic excitation frequency = 2100 Hz.

c. Results at $Re = 2.6 \times 10^5$. At this Reynolds number, *Re*, the excited value for lift coefficient is found to have dropped when compared against the two earlier Reynolds numbers *Re* (Fig. 4c). The excited drag value also does not show much change. Improvement in lift and drag values were evident only at the lower angle of incidence of $\alpha = 16$ deg, where the lift and drag both improved by around 10% in the frequency range $1600 < f < 2500$ Hz, or $10 < Sr < 15$.

2. C_L vs α

To construct the C_L vs α curve, values at C_L at $\alpha = 0$, 15.5, 16, 17, 18, and 19 deg have been used (Fig. 5). For a NACA 0012 airfoil,[9] flow separation occurs at around $\alpha = 16$ deg, when the sectional or two-dimensional lift coefficient C_l drops rapidly. Noting that there is no reliable method of predicting C_L and C_D values on a wing once the flow has separated, we have made an approximate attempt to check the validity of our data just before flow separation, that is, for the case when $\alpha = 15.5$ deg using the following expression[10]:

$$C_L = 2\pi\alpha - 2C_l[(1+\delta)/AR]$$

where $0.05 \leq \delta \leq 0.25$ (Ref. 11).

For $\alpha = 15.5$ deg, $AR = 4$, and $C_l = 1.6$, with $\delta = 0.05$, the predicted $C_L \approx 0.86$; with $\delta = 0.15$, the predicted $C_L \approx 0.78$; and with $\delta = 0.25$, the predicted $C_L \approx 0.71$. At $Re \approx 0.7 \times 10^5$, the experimentally determined value for C_L at $\alpha = 15.5$ deg was found to be 0.74. Consequently, the value obtained in the experiment was considered to be of the right order for this low aspect ratio wing.

Although the C_L vs α curve (Fig. 5) for 2100-Hz acoustic excitation frequency show considerable improvement in the lift coefficient over their corresponding unexcited values, the linear relationship between the C_L and α curve is lost. The poststall drop in lift coefficient is less severe suggesting the occurrence of partial separation of flow on the wing.

IV. Conclusions

The main conclusion of this study is that acoustic excitation of boundary layer under appropriate frequencies has the potential to provide the extra energy required to modify the severe adverse pressure gradient at or near the stall. This would help the flow to remain attached to the wing and to increase the wing stall margin. In the present study, acoustic excitation on a NACA 0012 wing have shown suppression of leading-edge separation and improvement in the lift and drag coefficients over their corresponding unexcited values at $\alpha = 16$, 17, and 18 deg, that is, 3 deg beyond stall angle of the unexcited wing. This study also shows some dependence of the beneficial acoustic frequencies on Reynolds number, with higher frequencies required for higher Reynolds number. Our study, however, did not find significant improvements at $Re = 2.6 \times 10^5$, possibly because the maximum equivalent Strouhal number during the test was low. To confirm the presence of stall suppression at $Re = 2.6 \times 10^5$ at a *Sr* of around 40 as displayed for the other two Reynolds numbers would require an excitation frequency in excess of 6000 Hz, which was not available during this study.

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Lift and Drag Characteristics of a Supersonic Biplane Configuration

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Introduction

THE requirements of man's initial powered flight endeavours were ably met by the biplane configuration. However, subsequent structural and aerodynamic advances found the biplane falling into disfavor in the early 1930s. For a fixed wing span biplanes do possess aerodynamic efficiency advantages as compared to a monoplane. At a given lift coefficient and assuming elliptic loading, the vortex drag of a biplane tends to half that of a monoplane as the separation distance between the wings tends to infinity. The biplane captures a larger volume of air that is accelerated down to generate the lift impulse, so reducing the downwash velocity and hence the kinetic energy imbued to the accelerated fluid.

Biplanes have several interesting characteristics that are summarized below. Prandtl and Tietjens[1] has shown for unstaggered biplanes (i.e., neither wing extends in front of the other) the drag increments caused by the mutual influence of the wings are equal and are always additive. For positive stagger (the upper wing in front of the lower wing) the upper wing increases the downwash on the lower wing so increasing its drag; vice versa for the effect of the lower wing on the upper wing. Munk[2] has shown that the total mutual induced drag of a biplane for a fixed gap is independent of the amount of stagger (Munk's stagger wing theorem). This theorem is only valid if the two wing's lift distributions are

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