Control of Wall-Separated Flow by Internal Acoustic Excitation

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This paper explores the control of wall-separated flow on a NACA 63_3-018 airfoil and a circular cylinder by using the internal acoustic excitation technique. Experimental study of the characteristics of the flow under internally emanating acoustic waves is performed in an open-type, suction wind tunnel. Tests are carried out at the Reynolds number ranging from 6.3×10^3 to 5.0×10^5 based on the relevant characteristic lengths, the airfoil chord, and the cylinder diameter. The control effectiveness is verified by the measurements of parameters such as the excitation frequency, the excitation level, and the forcing location. Data indicate that the excitation frequency and the forcing location are the key parameters for controlling the separated flow, and the forcing level is the least-effective parameter for the study. As long as the emanating acoustics is "locked in" to the separated shear-layer instability frequency and forcing is applied at the separation point, the separated flow is controlled most effectively. Moreover, the lift is increased and the drag reduced dramatically.

Nomenclature

C	= chord of airfoil
C_d	= section drag coefficient
C_d/C_{d0}	= drag coefficient ratio, defined as excited drag coefficient divided by unexcited drag coefficient
C_ℓ	= section lift coefficient
$C_\ell/C_{\ell 0}$	= lift coefficient ratio, defined as excited lift coefficient divided by unexcited lift coefficient
C_p	= pressure coefficient
D	= diameter of circular cylinder
f_e	= excitation frequency
f_s	= vortex shedding frequency
f_t	= shear layer instability frequency
P	= static pressure
P_t	= total pressure
$P_{t\infty}$	= total pressure in freestream
P_∞	= static pressure in freestream
Re_C	= Reynolds number based on chord, $Re_C = U_\infty C/\nu$
Re_D	= Reynolds number based on diameter, $Re_D = U_\infty D/\nu$
St	= Strouhal number, $(f_e C/U_\infty)$ or $(f_e D/U_\infty)$
U_∞	= freestream velocity
X	= coordinate along the freestream direction
Y	= coordinate normal to the freestream direction
ρ	= fluid density
ν	= kinematic viscosity
α	= sound emission angle of the circular cylinder
θ	= polar coordinate in the azimuthal direction

Introduction

THERE are many circumstances in practice where flow separation plays a major role in determining the flow characteristics. For instance, a wing-stall phenomenon may occur when an airplane flies at a high angle of attack. This is due to the occurrence of flow separation over a large portion of the wing surface. Likewise, a laminar flow passing a circular cylinder will also separate at about 82 deg with respect to the oncoming flow direction. The separated shear layer will roll up and merge into large vortices and shed downstream to the wake. It is of great interest to alleviate these problems by applying artificial means so as to suppress and control the flow separation, thus, subsequently improving the aerodynamic properties especially in the poststalled region.

In early 1948, Schubauer and Skramstad[1] observed that sound at particular frequencies and intensities could change the transition process of the boundary layer. The flowfield exhibited different characteristics, and the momentum exchange was enhanced due to the introduction of the acoustic waves. In accordance with this observation, the control of flow with vortical structures using acoustic excitation techniques has been studied extensively in recent years.

One of the techniques is called the external acoustic excitation, in which the sound is radiated onto the wall from a source outside the flow system. This technique has been applied by several researchers.[2-8] Collins[2,3] first observed that the application of the external acoustic excitation could change the lift on an airfoil. Ahuja and his coworkers[4,5] then conducted a series of studies of the excitation effects using the same technique. Further study of the interaction between the external acoustics and the separated flow was carried out by Zaman et al.[6] A beneficial effect of the external acoustic excitation on the separated boundary layer over the airfoil was presented in those papers. Regarding the application of this technique for flow passing over a circular cylinder, Peterka and Richardson[7] and Blevins[8] investigated its influence on the separated shear layers and the shedding vortices. They pointed out that the entrainment was enhanced due to the sound-induced velocity rather than the sound pressure. The same conclusion was also obtained by Zaman et al.[6] However, they found that the external excitation became effective only when the excitation frequency was close to the tunnel resonance frequency. Furthermore, the external excitation required high sound pressure levels in order to achieve satisfactory results (see Table 1). Hence, the external acoustic excitation appears impractical for actual applications.

To overcome those drawbacks, an internal excitation technique is used in which the sound is emanated from a narrow opening on the wall surface. The use of sound emission from small holes on the suction surface was mentioned by Collins[2] and Ahuja[5] and their coworkers, but no quantitative results were presented. The paper of Huang et al.[9] was probably the first that applied this technique for an airfoil performance study. Hsiao et al.[10] then employed a modified form of this

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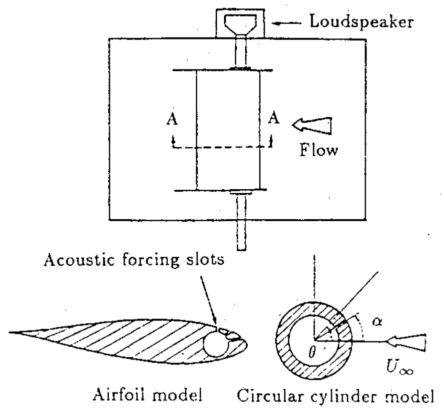


Fig. 1 Models and testing configurations.

same technique on a flow passing a circular cylinder. However, further studies on the effectiveness of the internal excitation and the sound-flow interaction mechanism are still needed.

In this paper emphasis is placed on the merits of the control for the wall-separated flow using the internal acoustic excitation technique. The flow passing a NACA 63_3-018 airfoil and a circular cylinder, respectively, is jointly investigated. The changes of the pressure distribution, lift, and drag with respect to the excitation level, the excitation frequency, and the excitation location are discussed in great detail.

Experimental Facilities

The investigations are conducted on a suction, open-type, subsonic wind tunnel with a 90- × 120-cm test section. A contraction section with contraction ratio 9 : 1 and five fine-meshed screens are used for managing the freestream turbulence intensity. The flow is driven by a 200 hp motor and its speed controlled by a set of rotary vanes. The freestream velocity in the test section ranges from 7 m/s to 35 m/s. The freestream turbulence intensity of the flow is less than 0.25%.

The flow visualization study is carried out in a smoke tunnel with a 20- ×40-cm test section. The contraction ratio for the smoke tunnel is also 9 : 1. Five interchangeable fine-meshed screens are installed in the settling chamber. A smoke wire technique is used to visualize the flow development over the model. Both photography and a video recording system are used to analyze the flow structures.

Two geometrically identical NACA 63_3-018 airfoil models and a circular cylinder model, both made of aluminum alloy, are employed in this study. Model A airfoil of a 30.5-cm chord is used in the subsonic wind tunnel for pressure measurements, and model B airfoil of a 12-cm chord is used for the flow visualization study. The pressure distributions are measured via 58 1-mm-diam pressure taps on model A. Three 1-mm wide slots, located at 1.25%, 6.25%, and 13.75% chord, are manufactured to emit acoustic waves respectively (see Fig. 1). The third model with circular cross section (see Fig. 1) is of 6 cm in outside diameter. It contains 71 pressure taps in the semispan region for the pressure measurements. The sound is emanated outward into the flowfield through a slot of 0.6 mm in width. Its emanating location is adjustable by simply rotating the cylinder. Throughout the experiments, the sound-pressure level is always kept at the value of 95 dB(A) measured at the slot exit.

A PDP 11/23$^+$ minicomputer is used as the central control unit. It is equipped with a 16-bit ANALOGIC ANDS 5400 data acquisition system. The frequency spectra are obtained by an Ono Sokki CF930 FFT analyzer. The Scanivalve system contains two 0.5-psid pressure transducers for measuring the static pressures. The scanning speed can be adjusted up to 20 ports/s by the computer. Information regarding the velocity fluctuations is acquired by a TSI 1050 constant-temperature, hot-wire anemometer. A probe with a single 5-μm-diam Pt wire is employed for the streamwise velocity measurements. The pressure fluctuations of the flow are monitored by a 0.125-in. B & K microphone.

For the drag measurements of the models, a 1.22-m-long wake rake is placed at 1.5 chords downstream of the trailing edge of the airfoil to investigate the momentum defect. The wake rake is made of 130 colinear total pressure tubes of 1.5 mm in diameter and three out of plane static pressure ports. The total pressure tubes are arranged at intervals of 4 mm.

Results and Discussion

Effects of Excitation on Pressure Distributions

The pressure distributions along the solid boundary of both the airfoil and the circular cylinder are measured to study the aerodynamic characteristics and the effect of the flow when they are excited internally by the acoustic waves. The pressure coefficient, normalized by the freestream properties, is defined as

$$C_p = \frac{P - P_\infty}{\frac{1}{2}\rho U_\infty^2}$$

and is used to describe the pressure distribution in the present study.

In the case of the flow passing a circular cylinder, it is found that the flow is very sensitive to the acoustic excitation especially when the forcing location is close to the separation point, about 82 deg. Figure 2 demonstrates an example of the dependence of the pressure distributions on the excitation frequency when the forcing location is fixed at 80 deg and $Re_D = 2 \times 10^4$. It is clear that a suction region is induced when the flow is excited at effective frequencies ranging from 100 to 400 Hz, and the change in pressure distribution is manifested vividly. Because of the streaming effect, as discussed by Williams and Amato,[11] the pressure on the upper surface is smaller in the neighborhood of the emission location than that on the lower surface. This results in a net lift, which becomes zero when the flow is unexcited.

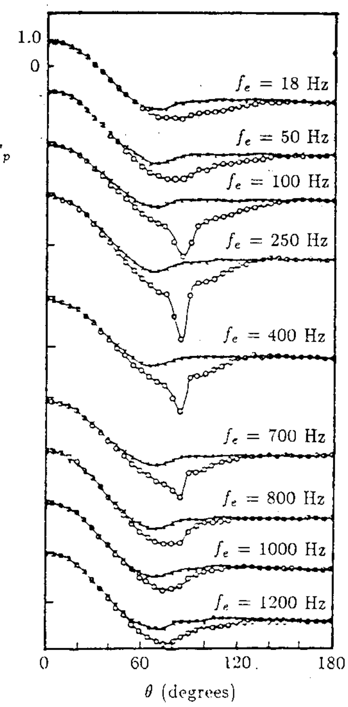


Fig. 2 Pressure distributions over a circular cylinder with acoustic excitation for $\alpha = 80$ deg and $Re_D = 2 \times 10^4$ (○: upper surface; ×: lower surface).

Table 1 Comparison of the effectiveness due to acoustic excitation

Ref.	Δ Cl%	Δ α stall deg	dB
Zaman[6] (external)	50	—	104 (on wall)
Ahuja[4] (external)	40	4	156 (on airfoil)
Ahuja[5] (external)	50	3	147 (on airfoil)
Collins[2] (external)	56	5	100 ~ 134 (on airfoil)
Collins[3] (external)	53	>5	100 ~ 134 (on airfoil)
Present (internal)	58	>6	95 (at slot exit)

In the study of the airfoil in the prestalled region, before a 12-deg angle of attack, there is no significant change in pressure distributions after the sound is introduced internally. However, for angles of attack higher than 16 deg in the poststalled region, a significant "suction peak" occurs near the leading edge on the upper surface. The larger suction peak area will result in a substantial contribution to the lift. Figure 3 depicts the comparison of the pressure distributions with and without excitation at particular frequencies. The solid curve represents the unexcited case, which illustrates an extraordinary flat pressure distribution on the upper surface. It corresponds to the occurrence of massive separation and thus would cause a severe leading-edge stall. When the flow is excited at the frequency of 50 Hz, a suction peak is formed. The pressure recovery steadily persists all the way downstream to the trailing edge on the upper surface. A steeper suction peak is formed when the excitation is increased to 100 Hz. A severe adverse pressure gradient causes earlier boundary-layer separation at about a 35% chord. The effect of excitation becomes degenerate with increasing excitation frequency. Forcing at the frequency of 700 Hz is a typical example.

Another set of experiments was also performed using the sound pressure level (SPL) as a parameter. It is found that the variation of the pressure distribution due to the SPL change is not so significant when compared to that of the frequency change as shown previously. This result is the same as control of mixing layer.[12] Throughout the experiment, the SPL is kept at 95 dB(A) measured at the slot exit. The effect is comparably good as seen in Table 1, although the level is relatively low.

It is concluded first from the pressure measurements that the internal acoustic excitation technique is highly feasible for separation control. That is, this technique can suppress and delay a severe leading-edge separation. The acoustic waves will cause a partially attached boundary layer as indicated in

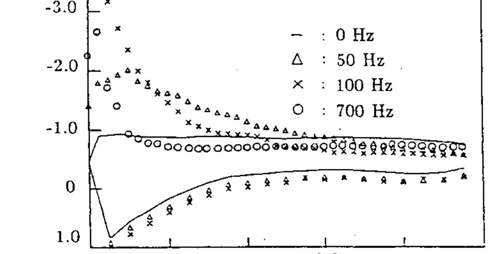


Fig. 3 Pressure distributions over an airfoil for an 18-deg angle of attack, $Re_C = 3.0 \times 10^5$ and forcing at a 1.25% chord.

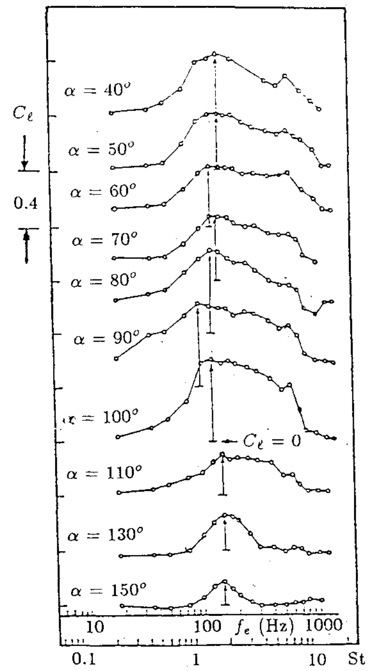


Fig. 4 Variation of lift coefficients over a circular cylinder with excitation frequency for different forcing locations and $Re_D = 2 \times 10^4$.

the pressure distribution, which would result in a lift increase. The explanation regarding how and why the flow reacts to the acoustic excitation will be discussed in the following sections.

Effects of Excitation Frequency on Lift

It has been known that the pressure distribution is sensitive to the disturbance frequency of the acoustics. The effective frequency for the present control technique will be discussed in this section.

The asymmetric pressure distributions of the circular cylinder as shown in Fig. 2 will certainly result in a nonzero lift. Correspondingly, Fig. 4 shows the variation of the lift coefficients with the excitation frequency at various forcing locations. It indicates that the increase in lift is most effective when the forcing location (α) is around in the postneighborhood of the separation point and at a Strouhal number (St) of 2.

Consider another case of the airfoil model in which the sound is emanated from the slot located at a 1.25% chord. Figure 5 shows the variations of the lift coefficient and the normalized lift coefficient ($C_\ell / C_{\ell 0}$) with the excitation frequency for the 18-deg angle-of-attack case, which is in the poststalled region. Results clearly indicate the sensitivity of the flow to the excitation frequency. The increase in lift is about 40% when the flow is excited within the frequency range from 50 to 250 Hz with peak frequency 110 Hz. The contribution from the excitation at other frequencies still exists, but it is not so significant. The collection of the normalized lift coefficients for various angles of attack is plotted in Fig. 6. The arrow symbol indicates the most prominent change in lift with respect to the unexcited case. The base of each arrow represents $C_\ell / C_{\ell 0} = 1$ respectively. It is clear that the maximum lift increase is also around the St number of 2. Moreover, the acoustic excitation affects the flow in a distinctly characteristic manner with respect to the angle of attack. In general, it can be divided into three regions: the prestalled region, stalled region, and the poststalled region.

When the angle of attack is less than 8 deg, the lift increase is negligible. It then becomes significant as the angle of attack increases. In the region just prior to stall, the lift can be improved by several appropriate excitation frequencies. It can also be deteriorated by several excitation frequencies. Such a situation is found especially in the frequency range from 500 to about 1500 Hz for the 14- and 16-deg angle-of-attack cases. Similar results using external excitation were also observed by Zaman et al.[6] with a different airfoil model.

In the poststalled region where the angle of attack is greater than 16 deg, the deterioration of the lift is no longer found. Instead, a substantial improvement in lift reappears. For an angle of attack from 18 to 22 deg, more than a 40% increase in lift is obtained at the excitation frequency near 100 Hz. At the angle of attack of 24 deg, an improvement of 20% in lift is still preserved. In this case the excitation effect is still beneficial although the lift improvement is not so significant.

To investigate the dependence of the internal excitation on the Reynolds number, another result is shown in Fig. 7 for a Reynolds number of 1.85×10^4, which is one order of magnitude less than the previous case. Here, it shows the same phenomenon as obtained in Fig. 6, and the effectiveness due to the excitation is still prominent. Although the effective

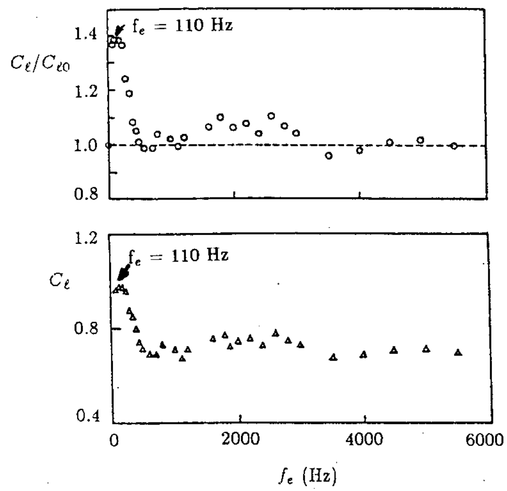


Fig. 5 Variation of lift coefficients over an airfoil with excitation frequency for an 18-deg angle of attack, $Re_C = 3.0 \times 10^5$ and forcing at a 1.25% chord.

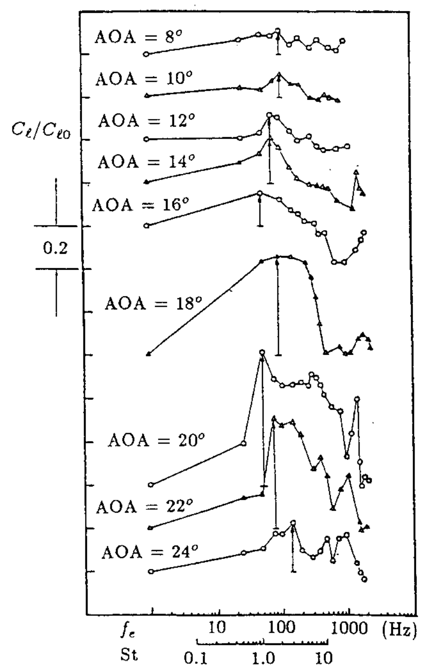


Fig. 6 Variation of normalized lift coefficients over an airfoil with excitation frequency at different angles of attack for $Re_C = 3.0 \times 10^5$ and forcing at a 1.25% chord.

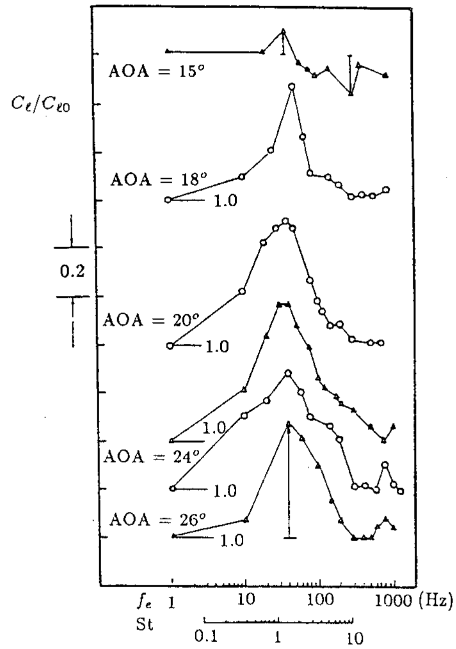


Fig. 7 Variation of normalized lift coefficients over an airfoil with excitation frequency at different angles of attack for $Re_C = 1.85 \times 10^4$ and forcing at a 1.25% chord.

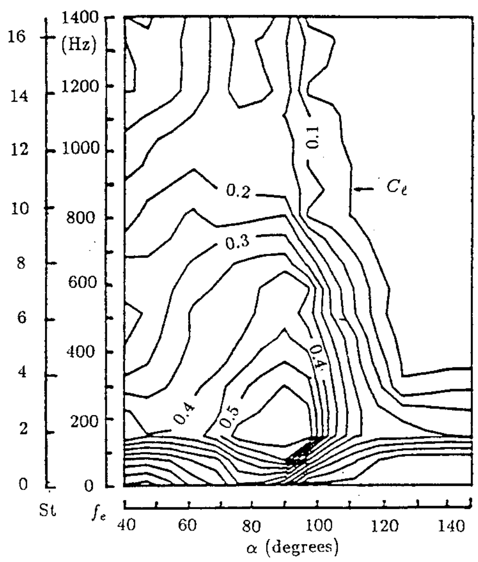


Fig. 8 Contour diagram of lift coefficient over a circular cylinder with excitation frequency and forcing location for $Re_D = 2 \times 10^4$.

excitation frequency is relatively lower than the frequency found at the Reynolds number of 3.0×10^5, the change of the effective *St* number range is not sensitive. That is, the peaked *St* number is centered at about 2.

Effects of Forcing Location on Lift

Figure 8 shows the contour of the lift for the circular cylinder under acoustic excitation. It clearly depicts a concentrated region where the change of lift is the most prominent. When the forcing location (α) is placed around, but preferably a little after, the separation point, a significant change in lift is observed. Meanwhile, this high gradient region corresponds to the *St* number in the order of 1. More precisely, the value ranges about from 1 to 3.

In the airfoil model study, three slots, located at a 1.25%, 6.25%, and 13.75% chord for the acoustic emission are investigated, respectively. Figure 9 depicts the dependence of

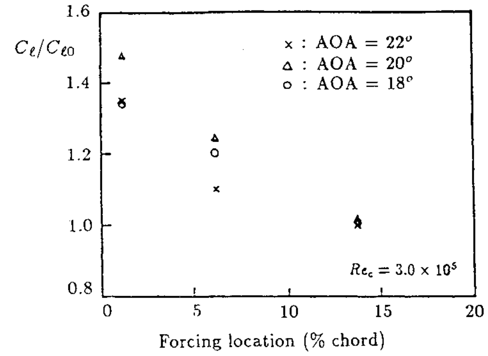


Fig. 9 Comparison of normalized lift coefficients over an airfoil at different forcing locations for $Re_C = 3.0 \times 10^5$, $f_e = 110$ Hz ($St = 2$).

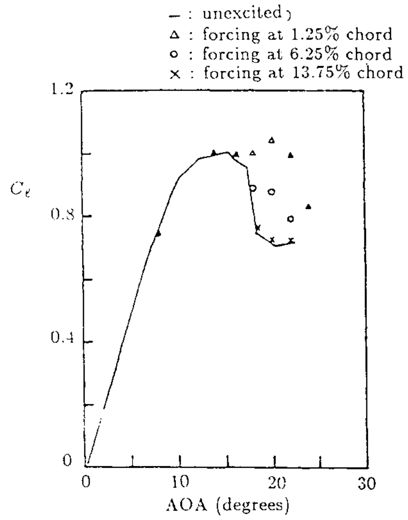


Fig. 10 Comparison of lift coefficient curves over an airfoil at different forcing locations for $Re_C = 3.0 \times 10^5$, $f_e = 110$ Hz ($St = 2$).

the lift on the forcing location at several angles of attack with f_t as a forcing frequency. It can be seen that forcing at a 1.25% chord has the most significant improvement in lift. In other words, the effectiveness of the boundary-layer control with internal excitation strongly depends on the forcing location, and forcing at a position close to the separation point is the most effective—especially in the poststalled region. In spite of forcing at different locations, the most prominent excitation frequency does not change at a fixed Reynolds number. When the forcing location gets further downstream, the effectiveness degenerates. From the results of both models, this is important evidence that the nature of local excitation control is due to hydrodynamical disturbances rather than acoustics. If it were acoustics, the forcing should not be sensitive to the location. The reason is because the length scale of the acoustic wave is much longer than the length scale of the models.

To summarize, the lift coefficient curve with an angle of attack is plotted in Fig. 10 for various forcing locations. The forcing frequency is fixed at f_t. When the flow is not excited, the airfoil is stalled permanently at the angle of attack higher than 16 deg, and a severe leading-edge separation occurs accordingly. When the flow is excited near the separation point at the favorable frequency, the flow separation is suppressed, and the maximum lift is increased. Moreover, in the poststalled region, not only the lift coefficient has been dramatically increased, but also the stall angle has been delayed by at least 6 deg. Table 1 illustrates the comparison of the results with the available data using the external acoustic excitation. It is clear that the current internal excitation technique bears more advantages as compared to the external excitation technique.

Flow-Acoustics Interaction

As mentioned earlier, the effectiveness of the boundary-layer control is highly sensitive to the excitation frequency, and the most effective frequency varies with the Reynolds number. How the effective excitation frequency correlates with the flow properties is a very important problem in the present investigation. The *St* number, defined as the excitation frequency scaled with the characteristic length and the freestream velocity, is found to be a governing nondimensional parameter in the present boundary-layer control technique. Figures 4 and 6–8 also depict the lift coefficient with respect to the *St* number. By examining those figures, it is found that the most effective *St* number always lies in the range from 1 to 3 provided that the chord of the airfoil and the diameter of the circular cylinder are considered as the length scale. Thus, it can be concluded that when the internal acoustic excitation technique is applied to control the flow separation, the most effective *St* number has to be in the order of 1 in the operating Reynolds number range from 6.3×10^3 to 5×10^5.

In order to investigate how the effective frequency is related to the flow characteristics, the measurements of the fluctuating velocities in the boundary layer are conducted at the operating Reynolds numbers. Figure 11 depicts the energy spectra of the velocity fluctuations at the Reynolds number of

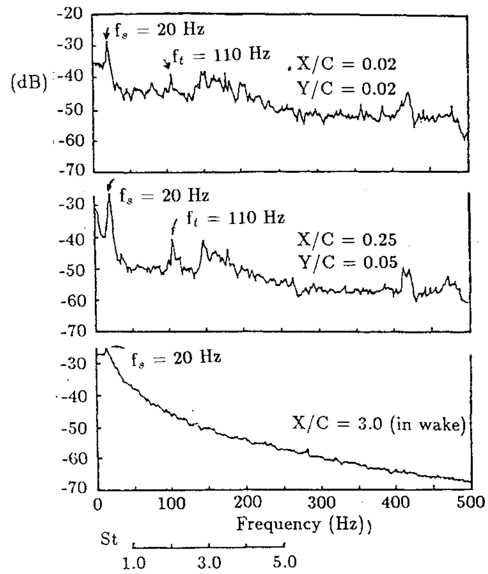


Fig. 11 Energy spectra of velocity fluctuations over an airfoil for a 24-deg angle of attack and $Re_C = 3.0 \times 10^5$.

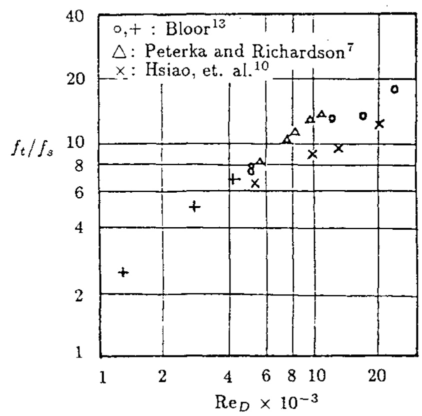
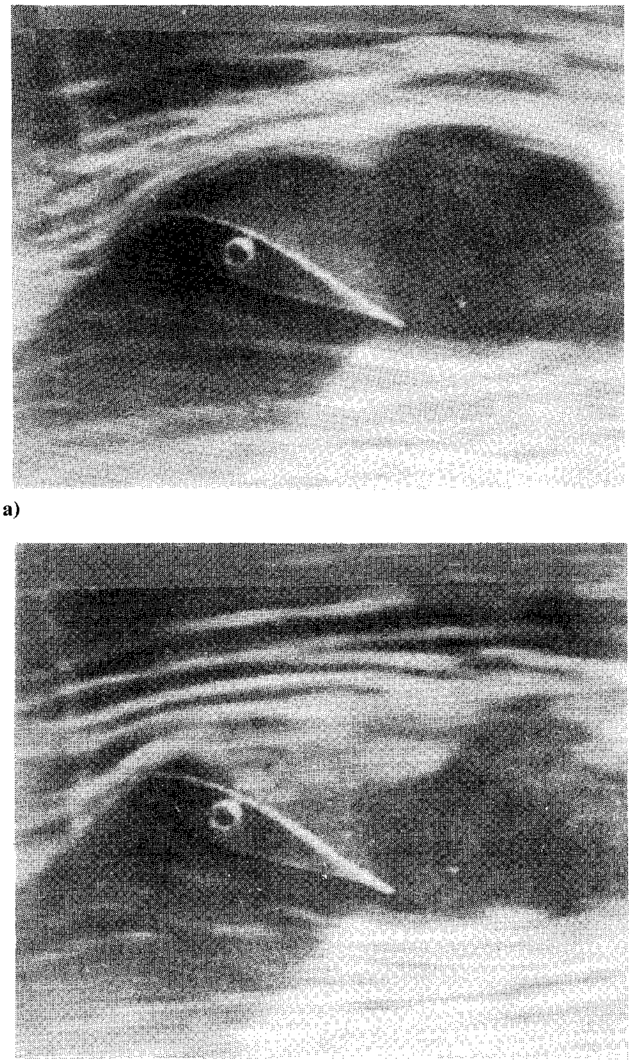


Fig. 12 Ratio of shear-layer instability frequency (f_t) and vortex-shedding frequency (f_s) with a Reynolds number for the flow passing a circular cylinder.

3.0×10^5 and a 24-deg angle of attack, where the flow is unexcited. The vortex shedding frequency (f_s) is clearly seen and measured to be 20 Hz, which is in good agreement with the formula given by Roshko[13] and Katz.[14] Another peak frequency of 110 Hz ($St = 2.1$) in the spectra matches the effective excitation frequency at the corresponding Reynolds number. This is also recognized to be the shear-layer instability waves (f_t) of the separated flow. The argument is further justified in the case of the flow passing a circular cylinder. The relationship between the frequencies of the shear-layer instabilities (f_t) and the shedding vortices (f_s) have been studied by Peterka and Richardson,[7] Hsiao et al.,[10] and Bloor and Gerrard.[15,16] Figure 12 shows the comparison of the results f_t/f_s with other data with respect to the Reynolds number. These frequencies were measured in the separated boundary layer by conducting velocity spectral analysis. It can be seen that the shear-layer instability frequency (f_t) is about an order of magnitude higher than the vortex shedding frequency (f_s). Recalling the results obtained using the internal acoustic excitation technique, the effective excitation frequency corresponds to the shear-layer instability frequency. Physically, this is a "locked-in" phenomenon, and it will amplify the instability waves such that the flow mixing and momentum transport are enhanced, and the separated boundary layer has the ability to be reattached to the boundary. The lift is thus recovered.



a)

b)

Fig. 13 Comparison of flow patterns over an airfoil: a) without excitation and b) with internal excitation at 30 Hz ($St = 1.6$) for a 24-deg angle of attack and $Re_C = 1.85 \times 10^4$.

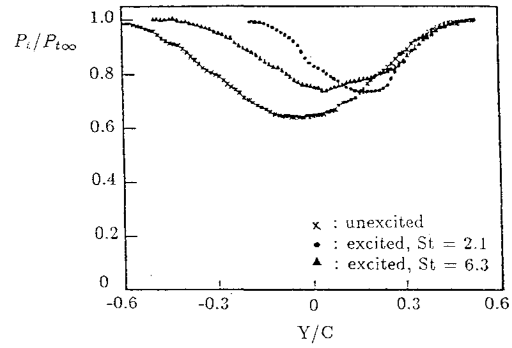


Fig. 14 Wake defect profiles in the wake of an airfoil with and without internal excitation for an 18-deg angle of attack and $Re_C = 3.0 \times 10^5$.

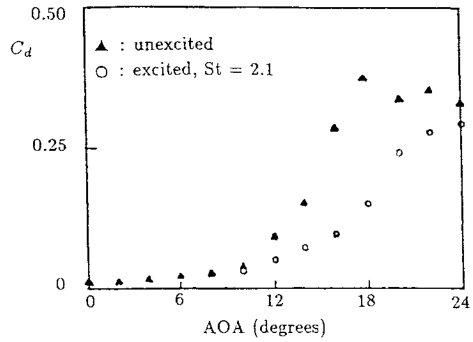


Fig. 15 Section drag coefficient against an angle of attack of an airfoil with and without internal excitation for $Re_C = 3.0 \times 10^5$.

The above quantitative results are further substantiated by carrying out a flow visualization study in the smoke tunnel. Figure 13 shows the typical flow patterns at the Reynolds number of 1.85×10^4 and a 24-deg angle of attack, which is in the poststalled region. The separated flow at the leading edge is clearly revealed when the flow is unexcited (see part a of Fig. 13). It would cause a severe deterioration in lift. After the flow is internally excited by the acoustic waves at the St number of 1.6, in which the excitation frequency equals the shear-layer instability frequency, the separated boundary layer is then reattached to the boundary of the airfoil (see part b of Fig. 13). The reattached boundary layer will certainly ensure the lift recovery. Also, since the wake region is narrowed due to the boundary-layer reattachment, the drag will be reduced accordingly. A typical result of the wake defect profiles excited at different St numbers is shown in Fig. 14 using the conventional wake defect method. A narrower wake with a smaller profile defect indicates a less momentum loss, which ensures a smaller drag coefficient. The collection of the drag coefficient measurements with the angle of attack is then depicted in Fig. 15 along with the unexcited case. It is clear that the drag is reduced significantly for higher angles of attack. We can conclude that, by applying the acoustics internally for the flow excitation at the shear layer instability frequency, the double advantages, that is higher lift and less drag, will assure the higher value of the lift-to-drag ratio. This, in turn, ensures the high performance of the present excitation method.

Concluding Remarks

The study of the internal acoustic excitation technique to control the flow separation passing an airfoil and a circular cylinder is conducted in a suction, open-type subsonic wind tunnel and a smoke wind tunnel. It is found that the technique is quite promising and effective in controlling the separation and subsequently providing a dramatic contribution to the aerodynamic performance—especially in the poststalled airfoil. The mechanism of the flow-acoustics interaction is the same for both models used in this internal excitation study.

The enhancement of the flow mixing and momentum transport, due to internal excitation, produces a suction peak at the leading edge of the upper-surface airfoil. The suction peak results in an increase of lift and a narrower wake with a smaller velocity defect. Consequently, the aerodynamic efficiency (lift-to-drag ratio) is substantially improved in a wide range of angle of attack. In the operating Reynolds number range, as long as the excitation frequency is locked-in to the shear-layer, instability frequency of the separated boundary layer and forcing around the natural separation point, the excitation is most effective, and the aerodynamic efficiency is strikingly improved. The most effective *St* number for excitation happens to be in the order of 1, or more precisely 1 to 3, in the operating Reynolds number range. This can be used as a quick way to estimate the effective forcing frequency.

In the present study, the sound pressure level is relatively lower than that employed by other researchers. Based on the results obtained, it can be concluded that the internal acoustic excitation technique is highly promising as compared to the external acoustic excitation method in flow separation control.

Acknowledgments

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References

[1]Schubauer, G. B., and Skramstad, H. K., "Laminar Boundary Layer Oscillation and Transition on a Flat Plate," NACA Rept. 909, 1948.

[2]Collins, F. G., and Zelenevitz, J., "Influence of Sound Upon Separated Flow Over Wings," *AIAA Journal*, Vol. 13, March 1975, pp. 408–410.

[3]Collins, F. G., "Boundary Layer Control on Wings Using Sound and Leading-Edge Serrations," *AIAA Journal*, Vol. 19, Feb. 1981, pp. 129–130.

[4]Ahuja, K. K., Whipkey, R. R. and Jones, G. S., "Control of Turbulent Boundary Layer Flows by Sound," AIAA Paper 83-0726, 1983.

[5]Ahuja, K. K., and Burrin, R. H., "Control of Flow Separation by Sound," AIAA Paper 84-2298, 1984.

[6]Zaman, K. B. M. Q., Bar-Sever, A., and Mangalam, S. M., "Effect of Acoustic Excitation on the Flow Over a Low-Re Airfoil," *Journal of Fluid Mechanics*, Vol. 182, Sept. 1987, pp. 127–148.

[7]Peterka, J. A., and Richardson, P. D., "Effects of Sound on Separated Flows," *Journal of Fluid Mechanics*, Vol. 37, June 1969, pp. 265–287.

[8]Blevins, R. D., "The Effect of Sound on Vortex Shedding from Cylinders," *Journal of Fluid Mechanics*, Vol. 161, Dec. 1985, pp. 217–237.

[9]Huang, L. S., Maestrello, L., and Bryant, T. D., "Separation Control Over an Airfoil at High Angles of Attack by Sound Emanating from the Surface," AIAA Paper 87-1261, 1987.

[10]Hsiao, F. B., Shyu, J. Y., Liu, C. F., and Shyu, R. N., "Experimental Study of an Acoustically Excited Flow Over a Circular Cylinder," *3rd International Symposium on Transport Phenomena in Thermal Control*, Taipei, R.O.C., Aug. 1988, (to be published by Hemisphere, Washington DC, pp. 261–270).

[11]Williams, D., and Amato, C., "Unsteady Pulsing of Cylinder Wakes," AIAA Paper 88-3532, 1988.

[12]Ho, C. M., and Huang, L. S., "Subharmonics and Vortex Merging in Mixing Layers," *Journal of Fluid Mechanics*, Vol. 119, June 1982, pp. 443–473.

[13]Roshko, A., "On Drag and Shedding Frequency of Two-Dimensional Bluff Bodies," NACA-TN-3169, 1954.

[14]Katz, J., "A Discrete Vortex Method for the Non-Steady Separated Flow Over an Airfoil," *Journal of Fluid Mechanics*, Vol. 102. Jan. 1981, pp. 315–328.

[15]Bloor, M. S., "The Transition to Turbulence in the Wake of a Circular Cylinder," *Journal of Fluid Mechanics*, Vol. 19, June 1964, pp. 290–304.

[16]Gerrard, J. H., "A Disturbance-Sensitive Reynolds Number Range of the Flow Past a Circular Cylinder," *Journal of Fluid Mechanics*, Vol. 22, May 1965, pp. 187–196.