Control of trailing-edge noise from airfoil using a plasma actuator

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Abstract
Control of noise generation at an airfoil trailing-edge was conducted using a plasma actuator for an NACA0012 airfoil with an angle of attack of -2° at a chord Reynolds number $Re = 1.6 \times 10^5$, where the boundary-layer instability on the pressure side was responsible for the generation of the tonal trailing-edge noise. A thin electrode was installed uniformly in the spanwise direction at the maximum wing thickness location. The actuator was operated in pulsed (burst) mode to excite linear disturbances other than the naturally growing one artificially, and the responses of flow and tonal sound were examined. The naturally radiated tonal trailing-edge noise was found to be replaced by the weaker tonal sound at the bursting frequency when the bursting frequency was near that of the natural tone. It was also demonstrated that when the actuation frequency was far from that of original (natural) sound, the boundary-layer transition was dominated by naturally unstable broadband disturbances, leading to complete suppression of the tonal trailing-edge noise.

Keywords: Trailing-edge noise, Aerodynamic sound, Instability wave, Flow control, Plasma actuator

1. Introduction
It is known that the tonal sound radiation from an airfoil trailing-edge occurs at low and middle Reynolds number up to $Re=10^6$ for small angle of attack (Paterson 1973). Such tonal noise generation is addressed by the feed-back loop mechanism between the excitation of the instability wave (the Tollmien-Schlichting (T-S) wave) in the boundary layer and sound radiation at the trailing-edge (Tam 1974; Arbey and Bataille 1984; Fink 1975; Nash et al. 1999; Desquesnes et al. 2007). So far, the acoustic feedback leading to the tonal trailing-edge noise is known to be suppressed either by tripping the boundary-layer far upstream of the trailing edge (Longhouse 1977; Akishita 1986) or by changing the boundary layer instability (Inasawa et al. 2013) although the former method brings additional friction and form drag. In the present study, we shall expand the latter approach and demonstrate another control methodology, in which the frequency and intensity of the tonal trailing-edge noise are controlled by introducing less unstable disturbances using a plasma actuator.

Plasma actuators (PAs) (Corke et al. 2010) have been used to control the flow in a wide range of applications, e.g. separation (Post and Corke 2004; Huang and Corke 2006), wake (Thomas et al. 2008; Kozlov and Thomas 2011), boundary layer transition (Grundmann and Tropea 2008; Grundmann and Tropea 2009) and so on. Among these, attempts to stabilize boundary layers using PAs have been relatively limited. Grundmann and Tropea (2008, 2009) first demonstrated the ability of PAs to delay boundary-layer transitions. In their experiments, the artificially excited T-S waves in the flat-plate boundary layer under adverse pressure gradient were attenuated using downstream installed plasma actuators. They examined both continuous and pulsed-mode operations and showed that the boundary-layer transition could be delayed even by pulsed operation, which has better energy efficiency. Stabilization of the boundary layer in the natural transition over an airfoil, with the aim of tonal noise suppression, was examined by Inasawa et al. (2013), who employed a flush-mounted configuration of actuator electrode in order to minimize the possible interference with
Inherent instability waves and demonstrated that the weak surface flow induced by the plasma actuator slightly modified the mean velocity profile of the downstream boundary layer, stabilizing the boundary layer significantly to suppress the vortex roll-up near the trailing edge. As a result, complete suppression of trailing-edge noise was achieved.

In order to explore a more efficient control methodology of tonal trailing-edge-noise generation, we are focusing on unsteady (time-periodic) operation of actuators, in which PAs may excite less unstable instability waves in the boundary layer. If such disturbances can affect the boundary-layer transition contributing to the acoustic feedback, manipulation of both the frequency and intensity of the tonal trailing-edge noise would be possible. Besides, unsteady operations of the actuator may improve energy efficiency from a control viewpoint. Thus, in the present study, the response of tonal sound generation at the airfoil trailing-edge to artificial disturbances excited in the upstream boundary layer is examined experimentally using a plasma actuator in pulsed-mode (burst) operation.

2. Experimental setup and procedure

The experiment was conducted in an open-jet type wind tunnel with an exit cross-section of 600 mm × 600 mm. The turbulence intensity at the tunnel exit was 0.1 % of the free-stream velocity in terms of the root-mean-square (r.m.s.) value of the streamwise velocity fluctuation. Two Plexiglas sidewalls maintained the two-dimensionality of the main stream in the test section although the upper and lower areas were opened. An NACA0012 wing model made of resin whose chord (c) and span (s) were 200 mm and 598 mm, respectively, was set between the side walls (Fig. 1). To measure the trailing-edge noise with a dipole nature (Desquesnes et al. 2007), a microphone was installed 900 mm above the wing trailing-edge at mid-span. The output signal of the microphone was acquired by a PC through a 16-bit analog-to-digital conversion. The coordinates x, y and z represent the chordwise distance from the leading edge, the vertical distance from the wing surface, and the spanwise direction, respectively.

A plasma actuator (PA) of Single Dielectric Barrier Discharge (SDBD) type was used to control the flow. A plasma actuator consisting of copper tape (0.07mm thick) and polyimide film (0.05mm thick), as shown in Fig. 2(a), was installed uniformly in the spanwise direction on the pressure (upper) side at x_{PA}/c = 0.3, at the maximum thickness of the airfoil. In order to minimize actuator projection from the surface, no overlap was employed between the electrodes. A sinusoidal wave of f_{PA} = 20 kHz and E_{PA} = 800 V, in terms of r.m.s value, was supplied to the actuator in bursting manner (see Fig. 2b). The duty ratio of the bursting was fixed at DR (=T_{ON}/T_1) = 0.25, and thus, net input power was constant irrespective of the bursting frequency f_1 = 1/T_1. In the present experiment the bursting frequency was varied between f_1 = 238Hz and f_1 = 714Hz.

Velocity profiles in the boundary layer were measured by using hot-wire anemometers with single I-type probe.
Throughout the experiment the uniform flow velocity was fixed at $U_\infty = 12$ m/s. The Reynolds number based on the wing chord, $Re (=U_\infty c/v)$ was $Re = 1.6 \times 10^5$. Here, $v$ is the kinetic viscosity. The angle of attack was fixed at $\alpha = -2$deg. At this angle of attack, the pressure side boundary layer dominated the tonal sound generation at the Reynolds number $7 \times 10^4 \leq Re \leq 8 \times 10^5$ (Lowson et al. 1994).

The sensitive length of hot-wire sensor, a tungsten wire of 5μm in diameter, was 1mm. A particle image velocimetry (PIV) system (Dantec) consisting of a double-pulsed Nd:Yag laser and a CCD camera of $2048 \times 2048$ pixels was used to obtain instantaneous velocity and spanwise vorticity fields near the wing trailing-edge. In PIV measurements, an adaptive correlation algorithm with interrogation area of 32×16 pixels (in the $x$-$y$ plane) and 32×32 pixels (in the $x$-$z$ plane) was employed. The overlap of the interrogation area was 50%. The spatial resolution was $(\Delta x, \Delta y) = (1.2$mm, 0.6mm) in the $x$-$y$ plane and $(\Delta x, \Delta z) = (1.3$mm, 1.3mm) in the $x$-$z$ plane. Smoke particles were injected from the inlet of the wind tunnel.

Throughout the experiment the uniform flow velocity was fixed at $U_\infty = 12$ m/s. The Reynolds number based on the wing chord, $Re (=U_\infty c/v)$ was $Re = 1.6 \times 10^5$. Here, $v$ is the kinetic viscosity. The angle of attack was fixed at $\alpha = -2$deg. At this angle of attack, the pressure side boundary layer dominated the tonal sound generation at the Reynolds number $7 \times 10^4 \leq Re \leq 8 \times 10^5$ (Lowson et al. 1994).
3. Results and discussion

First, the characteristics of radiated trailing-edge noise from a single airfoil and the stability of the boundary-layer on the pressure side were examined. Figure 3 illustrates the power spectra of sound pressure levels (SPL) in the natural condition at $\alpha = -2^\circ$ at $U_{\infty}=12$ m/s ($Re=1.6 \times 10^5$). Here, the plasma actuator was not operated. We observed a distinct discrete tone at $f_0 = 365$ Hz whose magnitude was $SPL_0=78$ dB, about 25 dB larger than the background noise level. Figure 4 demonstrates the instantaneous (a) spanwise vorticity and (b) transverse velocity fluctuation at $z=0$, and (c) streamwise velocity in the $x$-$z$ plane. One of the 100 snapshots is displayed in the figure. The figure shows that two-dimensional vortices are formed and shed at and around the wing trailing-edge in the pressure-side (upper-side) boundary layer (Figs. 4a and 4c). These vortices produce a strong pressure fluctuation, i.e., the trailing-edge acoustic dipole by diffraction of vortex-induced transverse velocity fluctuation at the trailing-edge, as shown in Fig. 4(b). On the suction side (lower side) boundary layer, on the other hand, no certain patterns corresponding to coherent vortices were observed (Fig. 4a) and strong velocity fluctuation $(v/U_{\infty}>0.1)$ was found at $x/c=0.85$ (Fig. 4b), implying that the boundary layer had nearly undergone a transition to turbulence at the trailing edge, and thus, little contribute to the tonal sound generation there.

![Figure 5](image1.png)

**Fig. 5.** (a)The $y$-distributions of the mean streamwise velocity, and (b) the amplification based on linear stability theory between $x/c=0.3$ and 0.9. Symbols and lines in (a) represent hot-wire measurements and the Falkner-Skan velocity profile, respectively.

![Figure 6](image2.png)

**Fig. 6.** Effect of actuator operation on SPL. (a) $f_1=294$Hz, (b) $f_1=417$Hz, and (c) $f_1=714$Hz
Figure 5(a) depicts the $y$-distribution of the mean streamwise velocity $U$ of the boundary-layer on the pressure (upper) side at $x/c$ values of 0.2, 0.4, 0.6 and 0.8, measured by the hot-wire, showing that the laminar boundary layer grows downstream. Then, the linear stability at each $x$-station was examined. The Falkner-Skan velocity profile under non-zero pressure gradients was used to model the basic flow on the pressure side (illustrated by the solid lines in Fig. 5a), and the Orr-Sommerfeld equation with the Chebyshev collocation method was used for the analyses. Figure 5(b) illustrates the amplification factor, calculated by integrating spatial growth rates at $x$-stations between $x/c=0.3$ and 0.9 at interval of 0.1. The figure shows that the frequency of natural tonal trailing-edge noise, $f_0$, is nearly equal to that with maximum amplification, confirming that the growth of two-dimensional linear disturbance plays an important role in the tonal noise generation at the trailing edge.

We then next applied flow control on the pressure-side boundary-layer, which is responsible for the tonal noise generation at the trailing-edge. Figure 6 demonstrates the power spectra of SPL when the actuator was operated near the natural frequency $f_0$, at $f_1=294$ Hz (Fig. 6a) and $f_1=417$Hz (Fig. 6b). In both cases in which the actuator was

![Figure 5(a)](image1)

![Figure 5(b)](image2)

Fig. 7. Instantaneous snapshot of flow near the trailing-edge when the actuator was operated at $f_1=417$Hz. (a) spanwise vorticity, (b) transverse velocity fluctuation at $z=0$, and (c) streamwise velocity in the $x-z$ plane 2.5mm above the pressure side (upper side) trailing edge. Note that the (a) and (b) are simultaneous, and (c) was taken at different moment.

![Figure 5(c)](image3)

![Figure 5(d)](image4)

Fig. 8. Bursting frequency $f_1$ vs. sound pressure level, compared with the amplification based on linear stability theory as shown in Fig. 5(b).
deployed, the naturally radiated tonal sound was below the background level, and the tonal sound with \(f=f_1\) newly appeared although its intensity was weaker than that of natural tone, i.e., the intensity of the excited tonal sound was \(SPL_1=65\) dB for \(f_1=294\) Hz and \(SPL_1=63\) dB for \(f_1=417\) Hz, both of which were about 10 dB lower than \(SPL_0=78\) dB. As the bursting frequency of the actuator operation increased further, both the \(f_0\) and \(f_1\) components disappeared as shown in Fig. 6(c). Here, it should be noted that sound radiated from the actuator was at most 38 dB for \(238\) Hz \(\leq f_1 \leq 714\) Hz, below the background noise. Figure 7 illustrates instantaneous flow fields when the actuator was operated at \(f_1=417\) Hz. On the upper (pressure) side, vortex roll-up and corresponding transverse velocity fluctuation was observed at \(x/c=0.95\) in Figs. 7(a) and 7(b), and as shown in Fig. 7(c) the two-dimensionality of vortices was weaker compared to the natural case (see Fig. 4c), implying that the tonal sound became weaker due to the loss of spanwise coherency of vortices at the wing trailing-edge although the flow feature on the lower (suction) side was nearly the same as in Fig. 4(b). The intensity of tonal sound at \(f=f_1\), \(SPL_1\), is plotted and compared with the amplification curve in Fig. 8. We see that the variation of \(SPL_1\) is well correlated with the amplification of the linear waves, suggesting that the growth of linear disturbance excited by the unsteady (bursting) operation of the plasma actuator affected the sound generation at the trailing edge. In order to clarify this point, development of disturbance in the boundary layer was examined in detail using hot-wire. Figures 9(a) and 9(b) respectively depict the mean velocity profile and waveform of streamwise velocity fluctuation inside the boundary layer at \(x/c=0.5\), 0.2c downstream of the actuator, when the
Fig. 11. (a) The $y$-distributions of r.m.s. amplitude of streamwise velocity fluctuation and (b) power spectra at the $y$ location where the overall amplitude reaches its maximum ($f_1=417 \text{Hz}$). (a,b) $x/c=0.8$, (c,d) $x/c=0.9$.

Fig. 12. The $y$-distributions of r.m.s. amplitude of streamwise velocity fluctuation and power spectra at the $y$ location where the overall amplitude reaches its maximum ($f_1=714 \text{Hz}$). (a,b) $x/c=0.8$, (c,d) $x/c=0.9$. 

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actuator was operated at \( f_1 = 417 \text{ Hz} \). We see that the sinusoidal velocity fluctuation with forcing frequency \( f_1 \) develops in the boundary layer (Fig. 9b), although the mean velocity profile is almost the same (Fig. 9a). Note that the waveform of velocity fluctuation differs from the input signal to the actuator, confirming that the electrical noise radiated from the actuator did not affect the current measurements. Figure 10 illustrates the \( y \)-distribution of the amplitude of streamwise velocity fluctuation at \( x/c = 0.5 \) when the actuator was operated at \( f_1 = 417 \text{ Hz} \) (Fig. 10a) and \( f_1 = 714 \text{ Hz} \) (Fig. 10b). The amplitude of velocity fluctuation with \( f = f_1, u', \) was extracted using fast-Fourier-transform analyses, and the value was normalized by the maximum, \( u'_{1,m} \). The amplitude curve based on the linear stability analyses at each frequency is also plotted for comparison purposes. The agreement between experiments and the linear theory is quite good in both cases, indicating that the linear disturbances were surely excited by the plasma actuator with unsteady (pulsed-mode or “burst”) operation. Here, it should be noted that the maximum r.m.s. amplitude of excited disturbances was about 0.2\% of \( U_\infty \) at \( x/c = 0.5 \). Figures 11 and 12 depict the \( y \)-distributions of the r.m.s. amplitude of the streamwise velocity fluctuation and power spectra near the trailing-edge at \( x/c = 0.8 \) and \( x/c = 0.9 \) for \( f_1 = 417 \text{ Hz} \) (Fig. 11) and \( f_1 = 714 \text{ Hz} \) (Fig. 12). In the case of \( f_1 = 417 \text{ Hz} \), where the tonal sound radiation at \( f_1 \) was observed, the boundary-layer transition was initiated by the growth of excited wave as seen from the fact that most of disturbance was composed of the \( f = f_1 \) component whose magnitude was at most 3\% of \( U_\infty \) at \( x/c = 0.8 \) as shown in Figs. 11(a) and 11(b), indicating that the vortex roll-up was dominated by the growth of excited wave there. At \( x/c = 0.9 \), however, broadband disturbances were pronounced in addition to the \( f = f_1 \) component, as in Figs. 11(c) and 11(d), indicating that the excited wave generating vortices soon became three-dimensional (less spanwise coherency) at the trailing edge, and thus, the intensity of tonal sound became weaker than that of the natural one. In the case of \( f_1 = 714 \text{ Hz} \), on the other hand, frequency components between \( f = 200 \text{ Hz} \) and \( 500 \text{ Hz} \), being exhibited naturally unstable nature (see Fig. 5b), were dominant at \( x/c = 0.8 \) (Figs. 12a and 12b), and continuously distributing broadband spectrum was found at \( x/c = 0.9 \) (Figs. 12c and 12d). Thus, when the actuator was operated at frequency far from the naturally unstable one, the boundary-layer had undergone transition to turbulence just upstream the trailing edge, leading to complete suppression of tonal noise generation at the trailing edge.

4. Concluding remarks

Control of tonal trailing-edge noise was carried out by using a plasma actuator for an NACA0012 airfoil at an angle of attack of \(-2^\circ\) and a chord Reynolds number of \( 1.6 \times 10^5 \), where the boundary-layer instability on the pressure side was responsible for the generation of tonal trailing-edge noise. A 0.12mm-thick plasma actuator was installed on the pressure side at the 30\% chord position, beyond which the flow was exposed in the adverse pressure gradient. The actuator was operated in an unsteady (burst) manner, with the aim of exciting less amplified disturbance in the boundary layer downstream. The sound due to actuator operation was below the background noise level, and thus, its effects were negligible.

In the natural condition without actuator operation, prominent tonal sound radiation, whose amplitude was about 25dB larger than the background noise level, was observed at the frequency \( f_0 \) close to the most amplified disturbance predicted by the linear stability analyses. In this case, spanwise uniformly two-dimensional vortices generated by the instability wave were formed and shed at the wing trailing-edge.

When the actuator was operated at a frequency near that of the natural tone at \( f_1 \), the tonal sound at \( f = f_0 \) was replaced by a tone with exciting frequency \( f_1 = f_0 \) but with a smaller intensity than that of natural one, where the spanwise coherency of shed vortices became weaker at the trailing edge. The intensity of tonal sound excited by the actuator operation was well correlated with the amplification of the instability wave in the natural boundary layer; that is, the intensity became smaller as the bursting frequency \( f_1 \) deviated from \( f_0 \). When the actuator was operated far from \( f_0 \), on the other hand, the naturally unstable broadband disturbances dominated the boundary layer transition, and the boundary layer underwent transition to turbulence just upstream of the wing trailing-edge, leading to complete suppression of tonal trailing-edge noise generation.

Therefore, through the present flow control experiments, it was demonstrated that both the frequency and intensity of tonal sound generated at the wing trailing-edge could be well controlled using a plasma actuator with unsteady (pulsed-mode) operation.
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