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Frequency Selection Mechanism of Airfoil Trailing-Edge Noise

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An experimental investigation was conducted with the aim to reveal the frequency-selection mechanism of the acoustic noise emanated from the trailing edge of two-dimensional airfoils, and a single splitter plate was proposed for suppressing trailing-edge noise. It was observed that broadband disturbances caused by Tollmien–Schlichting instabilities in the transitional boundary layer increase if trailing-edge noise is suppressed with the proposed device. In addition, by the introduction of artificial acoustic feedback, for which the initial disturbances are obtained from a sensor for unsteady pressure near the trailing edge, a particular discrete Tollmien–Schlichting wave is selected from among the broadband disturbances caused by Tollmien–Schlichting instabilities. This result demonstrates that the process of frequency selection is clearly ascribable to a positive feedback loop between instabilities in the boundary layer and trailing-edge noise. Also, the application of artificial acoustic feedback shows that the airfoil boundary layer favors a particular frequency, which results in steplike structures as a function of the distance between the loudspeaker and the trailing edge that are similarly observed in conventional environments with natural trailing-edge noise emission.

Nomenclature

Cp	=	pressure coefficient
c	=	chordwise length of the model
f	=	frequency
l	=	distance between the trailing edge of the model and the microphone or loudspeaker
s	=	length of the splitter plate
U	=	mean velocity in streamwise direction
u	=	fluctuating velocity in streamwise direction
x	=	distance in streamwise direction
y	=	distance in the direction normal to the model surface
z	=	distance in spanwise direction
α	=	angle of attack
β	=	angle between wing centerline and microphone or loudspeaker
δ^*	=	displacement thickness of the boundary layer

Subscripts

f	=	component at the discrete frequency
0	=	fundamental
1	=	first harmonic
2	=	second harmonic
∞	=	quantity at infinity
$'$	=	root mean square value of a fluctuating quantity

I. Introduction

IT WAS reported [1,2] at the beginning of the 1970s that discrete tones are emitted from isolated airfoils or helicopter rotors in specific flow conditions. It has been known that such tonal noises are generated at the trailing edge (TE) of two-dimensional airfoils with a certain configuration in a moderate Reynolds number range of 10^4 to 2×10^6, based on the airfoil chordwise length. A similar phenomenon has been observed to occur on wind turbines, gliders, cooling fans of office appliances, and so on. Therefore, this topic, which is practically important, has attracted many engineers and researchers regarding noise reduction. Although the basic flow is so simple, the generation mechanism of discrete tones is quite complex because of the acoustic interaction between the tonal noise and the transitional boundary layer developing on the airfoils. Thus, the generation mechanism still remains open.

The fact that noise at the TE of an airfoil is discrete is conventionally explained as the result of the presence of an acoustic feedback loop between the Tollmien–Schlichting (T–S) instability waves and the acoustic emission at the TE of the airfoil. In other words, a single frequency is selected in such a positive feedback loop, even though naturally growing T–S waves consist of broadband disturbances. To the authors' knowledge, there have been no attempts to experimentally examine the scenario in which a particular fre-k loop. The present paper focuses on evidence that the acoustic feedback loop in the generation of TE noise in airfoils is in fact formed, and a particular frequency is selected from among naturally growing disturbances. Subsequently, we focus on the phenomenon that, although the general trend of the frequency of acoustic noise follows the freestream velocity U raised to the power of 1.5, the local trend is a function of U to the power of 0.8, resulting in the formation of a steplike structure in the trend.

To elucidate the existence of an acoustic feedback loop in the process of the generation of TE noise, it is necessary to completely suppress acoustic emission from the airfoil. Since acoustic noise is emitted in the process of vortex shedding at the TE, vortex motion should be prevented in order to successfully suppress acoustic noise. Our previous paper [3] successfully removed TE noise emission with the help of a row of roughness elements near the TE on the pressure side of a NACA0015 airfoil, which facilitates the transitional boundary layer into a turbulent state. Nevertheless, this method is not adequate for the present purpose because of the alteration of the basic flow. In the present paper, a sophisticated method is proposed that is based on the use of a thin flat plate at the TE of the airfoil. In suppressing acoustic noise, a new attempt was made to search for the origin of the ladderlike structure in the general acoustic trend. Since feedback occurs between T–S waves at the boundary layer of the

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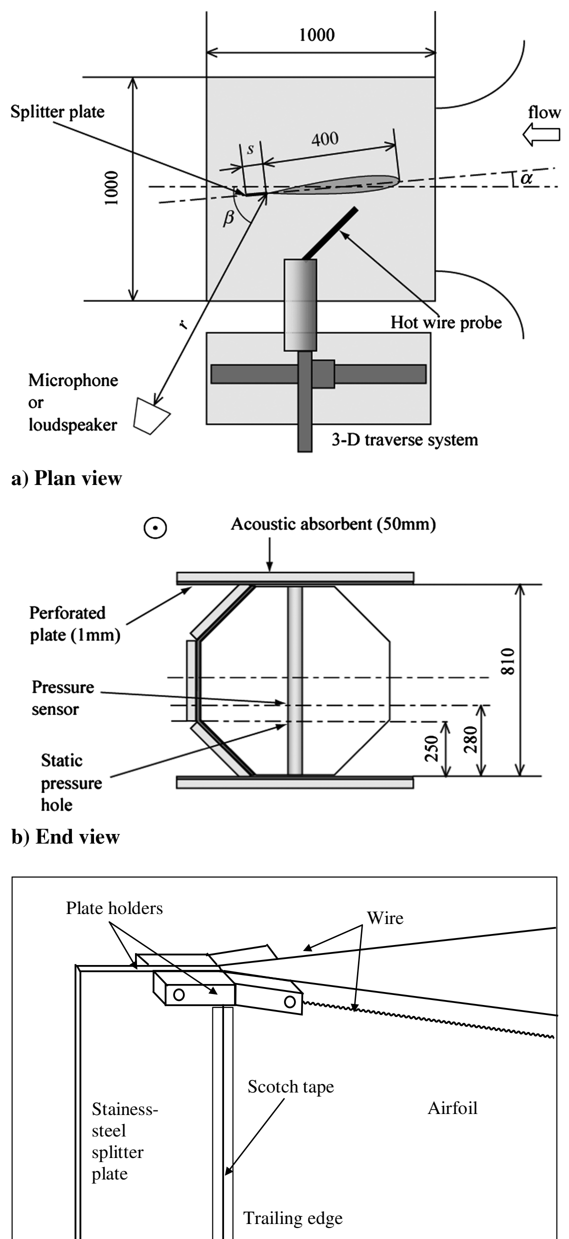
airfoil and the TE noise, naturally growing unstable disturbances in the boundary layer of the airfoil are collected and acoustically emitted via a loudspeaker outside of the test section. It is expected that this synthetic loop favors a particular frequency among the broadband disturbances caused by the T–S instabilities and ensues the formation of the aforementioned ladderlike structure in association with sound receptivity.

II. Experimental Setup

A. Wind Tunnel

The experiment was conducted in a low-turbulence wind tunnel at the Institute of Fluid Science, Tohoku University, Japan. Further details regarding the wind tunnel are provided by Kohama et al. [4]. Figure 1 illustrates the schematics of the test section. To avoid the reflection of sound at the hard wall, the experiment was conducted under semiopen conditions, with an octagonal test section with a streamwise length of 1000 mm, which was placed at a distance of 810 mm from the opposite side. The ceiling and the floor comprised a sandwich structure, of an acoustic absorbent with a thickness of 50 mm, placed on a perforated stainless-steel plate with a thickness of 1 mm, for which the porosity was 58%. The sidewall on the suction side of the airfoil was also made from a perforated plate, for which the inside surface was covered with Japanese paper used for a sliding door, while the other sidewall was left open. The surface of the collector was covered with a layer of acoustic absorbent and pile fabrics downstream of the test section in order to make it anechoic. The effects of this setup on sound absorption are described by Nishimura et al. [5] The test section table was also covered with a layer of acoustic absorbent with a thickness of 20 mm. Regarding the coordinate system, x is the distance measured streamwise from the leading edge, y is the normal-to-wall distance, and z is the spanwise distance. The overall experimental configuration is shown in Fig. 1.



a) Plan view

b) End view

c) Close-up view of the splitter plate attached to the trailing edge of airfoil

Fig. 1 Experimental setup of airfoil model in the test section.

B. Wing Model

The airfoil model, which was made of stainless steel, had a NACA0012 section for which the chord length and spanwise length were 40 and 810 mm, respectively. The model was placed on a turntable and spanned vertically across the test section, where its leading edge was located 200 mm downstream of the outlet of the nozzle of the wind tunnel. Forty-six static pressure ports were tapped and 22 sensors for unsteady pressure (Kulite XCS-062) were flush-mounted on both sides of the wing in a streamwise direction, as shown in Fig. 1b. The Helmholtz resonance frequency of the sensors for unsteady pressure was approximately 10 KHz, which was much higher than the target frequency of the TE noise. More details are described in [6].

C. Splitter Plate for Noise Suppression

For the purpose of suppressing TE noise, four stainless-steel splitter plates with chordwise lengths of 20, 40, 60, and 80 mm and a thickness of 1 mm were prepared, for which the thickness is the same as that at the airfoil TE, in order to investigate the effects of the streamwise length on the suppression of sound. Each plate braced by metallic holders at both ends, for which the front parts were pulled by a looped wire, was spanned at the TE and smoothly attached to the TE of the wing model with the use of scotch tape, as shown in Fig. 1c. Since the attachments induced additional circulation, the distribution of static pressure around the model was altered. Therefore, the angle of attack for the model was slightly readjusted to ensure that it coincided with that of the baseline in the case without a plate. For instance, the angle of attack for the plate with a length of 60 mm was set at 3.4 deg, while the original angle was 4 deg.

D. Equipment

A constant-temperature hot-wire anemometer with an analogue linearizer was used for measuring the mean and fluctuating velocities (U and u) in the streamwise direction. The length of the sensitive part of the hot-wire sensor, which comprised a tungsten wire with a diameter of 5 μm, was about 1 mm, and the wire was copper-plated at both ends and soft-welded to the tips of the prong support. It was possible to move the hot-wire sensor continuously in the streamwise (x), the normal-to-wall (y), and the spanwise (z) directions by using a three-dimensional (3-D) traversing mechanism. The output of the anemometer versus the velocity was a priori calibrated in uniform flows. Acoustic sounds were measured by using a B&K 1/2 in. microphone, which was located 2 m behind the TE, at an angle of 75 deg from the centerline toward the pressure side of the airfoil, as illustrated in Fig. 1. A loudspeaker, which was also used to introduce artificial sound pressure, was placed at $l = 2$ m behind the TE, at an angle of 60 deg from the centerline toward the pressure side of the airfoil.

In most experiments, the freestream velocity U_∞ was set at 18 m/s, which corresponds to a Reynolds number of $Re = 0.45 \times 10^6$, based on the chord length and the freestream velocity. The angle of attack of the model in the case of no splitter plate was set at 4 deg.

III. Results and Discussion

A. Baseline Results

First, we present the baseline results regarding the characteristics of the TE noise and the behavior of the boundary layer developing on the plain airfoil without the splitter plate. Figure 2 shows a typical frequency spectrum of the sound pressure level (SPL) as measured with the microphone placed on the pressure side of the airfoil. The background noise spectrum without the wing model in the test section is also shown for comparison. Once the model is installed in the test section, a tonal noise with a fundamental frequency of 428 Hz and its harmonics become perceivable. The sound level of the fundamental component exceeds the background noise level by more than 50 dB SPL.

Next, similar measurements were performed by varying the freestream velocity. The frequencies of the fundamental, the first, and the second harmonics of the tonal noise versus the freestream velocity U are plotted in Fig. 3. It is found that the overall trend follows a function of $U^{1.5}$, although steplike structures with a local trend of $U^{0.8}$ appear, for which the characteristics are similar to those obtained by Paterson et al. [7], Akishita [8], and others. It is also well known that such a local trend implies that the flow around the model is free of reflection and resonance of the acoustic sound emitted from the model, while the steplike structures in acoustic resonant environments have no slope [9]. On the other hand, Nash et al. [10] reported that the ladderlike structure was not observed when they attached an acoustic liner to the ceiling and the floor in a closed test section with a horizontally spanning model. As a result, they concluded that the absence of frequency jumps implied that there was no acoustic feedback loop in their experiments and, eventually, one discrete frequency was selected. However, a careful inspection of their results has confirmed that there are notable discontinuities along the general trend curve. Also, such a ladderlike structure inherently exists in transitional boundary-layer flow contaminated by acoustic sound, as shown later.

Figure 4 shows a typical spectrum of the velocity fluctuations at $x/c = 0.925$ in the boundary layer over the airfoil. It is remarkable that unsteady velocity fluctuations with a series of discrete frequencies have the same frequencies as those of the tonal noise (see Fig. 2), indicating a strong relation between instability waves in the boundary layer and acoustic sound. Another important point is that each discrete component accompanies the sidelobes. In other words, it appears that a particular frequency component is selected from the sidelobes in the acoustic feedback loop.

Figure 5 shows the growth of the disturbance against the streamwise direction on the pressure side, where u' and u'_f denote the overall rms value of the velocity fluctuations and the rms value of the discrete frequency component, respectively. It is noticeable that the discrete frequency in the u component coincides with the frequency of the tonal noise in the sound pressure, showing definite linkage between the boundary layer and acoustic sound. As can be seen from the overall rms value, the disturbances start to grow exponentially downstream from $x/c = 0.875$. The discrete frequency component and its higher harmonics contribute to the exponential growth. During this growing process, the amplitude of u'_f is magnified up to 25 times within the interval between the locations corresponding to 0.86 and 0.96 of the length in a chordwise direction. Since the magnitude of the overall disturbance exceeds 17% of the freestream velocity U_∞ and is subsequently saturated slightly upstream of the TE of the airfoil, a considerably strong roll-up vortex is shed from the TE on the pressure side. This vortex shedding is accompanied by the emission of strong acoustic sound.

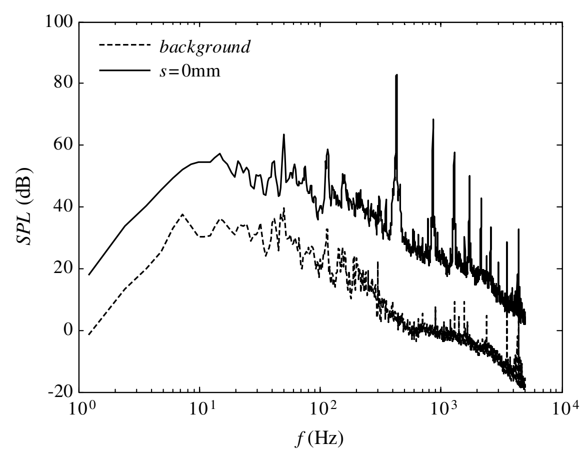


Fig. 2 Comparison of TE noise spectrum together with background noise spectrum, plotting −20 dB shift at 18 m/s; $s = 0$ denotes no splitter plate, and background represents no model in the test section.

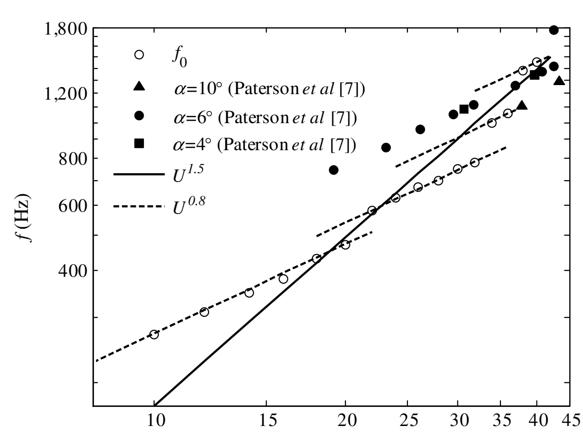


Fig. 3 The trend of the discrete tone frequencies versus the freestream velocity; $U \cdot f_0$ denotes the fundamental frequency of the discrete tone.

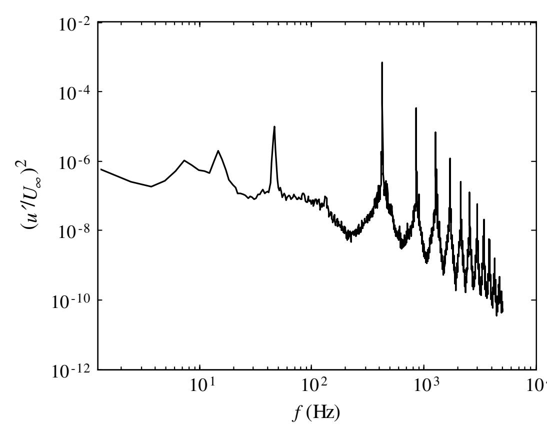


Fig. 4 Power spectrum of the u fluctuation in the boundary layer at $x/c = 0.925$.

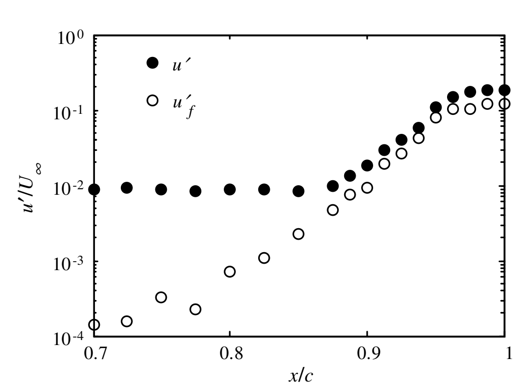


Fig. 5 Streamwise development of the maximum rms values of u and u_f.

The distribution of the mean velocity in the normal-to-wall direction on the pressure side is shown in Fig. 6. The inflection point on the velocity profile appears at around $x/c = 0.9$, where the velocity fluctuations start to grow rapidly. Such rapid growth originating from inflectional instability is similar to our previous observations in resonant environments [3,9] and is consistent with the observations of Nash et al. [10] under anechoic flow conditions.

B. Effects of the Splitter Plate

The attachment of a splitter plate is effective for eliminating tonal noise, as already reported by Longhouse [11], for axial flow fans. For the present experiment, a single plate with a length between 20 and 60 mm and a thickness of 1 mm was smoothly attached to the TE of the wing model. The spectra of the sound pressure, with and without the 60 mm splitter plate, are compared in Fig. 7. The discrete frequency components are completely suppressed to the level of the background noise in the presence of the plate.

Figure 8 shows the corresponding comparison of the power spectra of the maximum u fluctuations in the normal-to-wall direction at $x/c = 0.925$, with and without a splitter plate. In the case where a splitter plate is not used, it is observed that the dominant discrete frequency of the u fluctuation is identical to that of the acoustic noise. Once acoustic noise is suppressed by the plate, broadband disturbances associated with T–S instability waves are distinctly observed instead of discrete components. The center frequency of the broadband components, peaking at around 500 Hz, is remarkably close to the discrete frequency in the case where a splitter plate is not used. This comparative experiment strongly suggests that there exists an acoustic connection between TE noise and instability waves in the boundary layer, as reported by Tam [12] and others. Therefore, it is speculated that the frequency of the TE noise is selected from the broadband T–S disturbances in this positive feedback loop via an acoustic connection. It appears that the selected

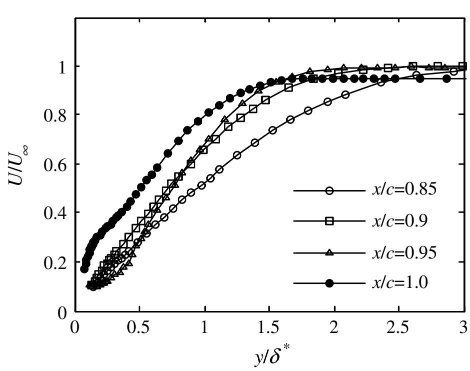


Fig. 6 Profiles of mean velocity in the boundary layer at various streamwise locations.

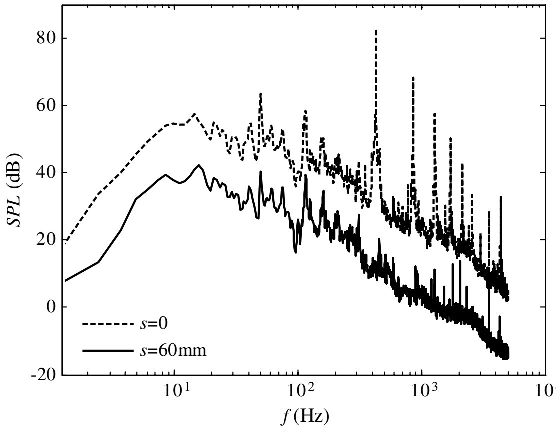


Fig. 7 Comparison of SPL, with and without the splitter plate. Spectrum with the splitter plate is shifted −20 dB.

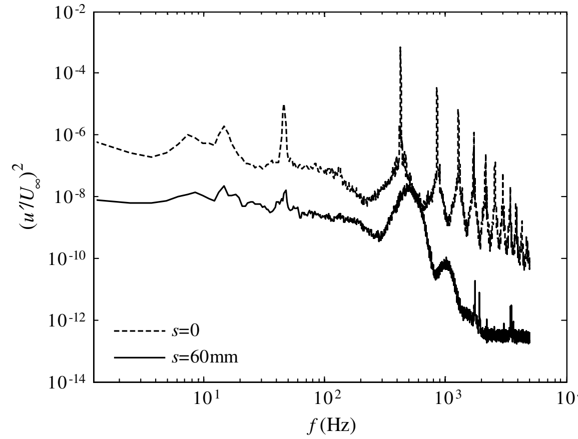


Fig. 8 Comparison of the power spectra of the u fluctuation at $x/c = 0.925$, with and without splitter plate. Spectrum with the splitter plate is shifted −20 dB.

component suppresses the growth of unselected continuous components comprising the sidelobe around the frequency of the TE noise in the initial boundary layer.

Similar measurements were performed by using splitter plates with different chordwise lengths. The SPL from the TE is plotted against the splitter-plate length in Fig. 9. The SPL (in units of decibels) decreased linearly with the increase of the plate length and, eventually, the plate with a length of 60 mm resulted in an almost complete suppression of the sound. In this case, we consider the reasons why a plate attached to the TE suppresses TE noise. In the absence of a splitter plate, the boundary layer on the pressure side of the model is consistently laminar downstream from the TE. Tonal noise is usually emanated in association with coherent vortex shedding at the TE. Therefore, it is more likely that the splitter plate prevents the occurrence of vortex shedding at the TE. Nevertheless, the fact that the boundary layer became turbulent on the extended plate indicates that coherent vortices are no longer shed from the TE.

C. Artificial Acoustic Feedback

It was shown in the previous section that broadband T–S instability waves are amplified in the boundary layer of the airfoil with splitter plates that suppress TE noise. However, it has not yet been determined how a particular discrete frequency is selected from among the broadband T–S components in the acoustic feedback loop. In the present paper, a clever approach was applied as follows. For the airfoil with the 60 mm splitter plate, an acoustic feedback loop was forced, using an unsteady static pressure signal at $x/c = 0.925$ on the pressure side of the airfoil. The low-frequency components, which are insensitive to T–S instabilities, were removed from this signal by means of a high-pass filter (300 Hz). Finally, the

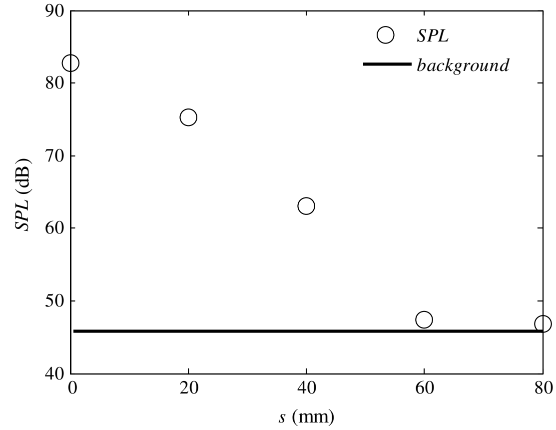
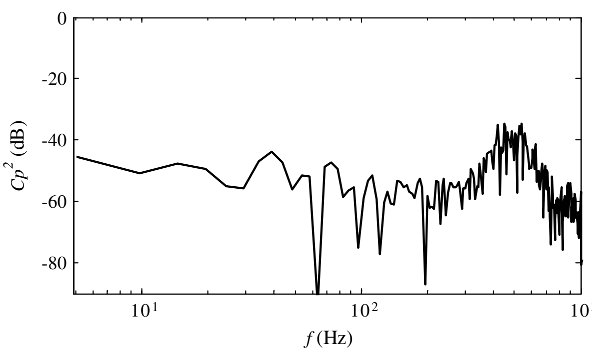
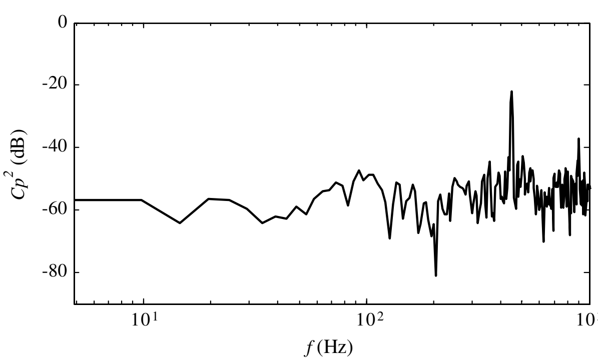


Fig. 9 SPL of the fundamental in discrete tones versus chordwise length of the splitter plate.

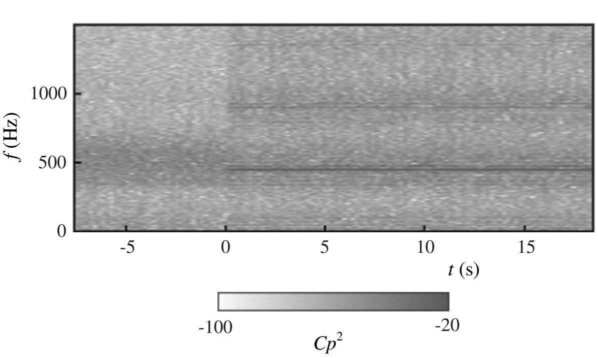
filtered signal was amplified and fed into a loudspeaker. For the new loop, acoustic disturbances based on the unsteady pressure signal at $x/c = 0.925$ were emitted from a loudspeaker, which was placed at a distance of $l = 2$ m behind the TE, at an angle of 60 deg from the centerline toward the pressure side of the airfoil. The time evolution of the frequency spectra of the unsteady pressure sensor at $x/c = 0.925$ is shown with contour mapping in Fig. 10. The horizontal axis and the vertical axis in Fig. 10c denote the time and the frequency, respectively, where data acquired for 2 s with a sampling rate of 4 kHz were transformed into spectrum with a frequency resolution of 0.5 Hz. The loudspeaker was activated at $t = 0$. Before the artificial disturbances were introduced, the spectrum peaking at round 500 Hz consists of broadband components originating from T–S instabilities as shown in Fig. 10a. Once the loudspeaker is activated, Fig. 10b shows that a single discrete frequency component with a frequency of 448 Hz, which is slightly higher than the component at 428 Hz observed in the case without a splitter plate, is selected from the broadband bump, and the second harmonic is also observed. This artificial feedback certainly demonstrates the presence of a direct link between unsteady disturbances in the boundary layer of the airfoil and TE noise.



a) An instantaneous spectrum before speaker activation



b) An instantaneous spectrum after speaker activation



c) Contour map of spectral evolution

Fig. 10 Spectral evolution of unsteady static pressure fluctuation at $x/c = 0.925$ against artificial acoustic forcing in the absence of natural TE noise, with a 60 mm splitter plate. The loudspeaker is activated at $t = 0$.

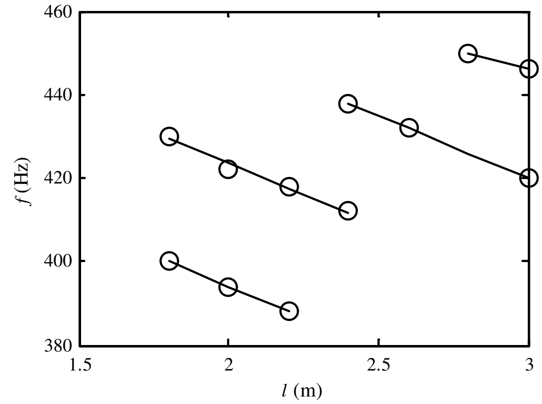


Fig. 11 The relation between selected frequency and speaker location. The distance between the loudspeaker and the TE is denoted by l.

It may be good to show data where different acoustic excitations had no effect on single frequency selection. Instead of using unsteady pressure signals as speaker input, a random signal from a white noise generator was emitted into the flow, in which no acoustic linkage was established. Similarly, a bandpassed signal peaking at about 500 Hz, which resembled the T–S disturbance spectrum without TE noise, was introduced, which again had no observable effect. These observations suggest that the acoustic feedback loop requires a constant phase relation with respect to external acoustic disturbances. In other words, the distance between the location of the speaker and the location of a certain point where the acoustic disturbance is internalized into the boundary layer of the airfoil is of paramount importance for the flow system associated with the sound. Therefore, similar experiments were repeated for different values of the distance l between the TE and the loudspeaker, and the results are shown in Fig. 11, where it is clear that two frequencies in a ladderlike structure are selected from the broadband disturbances. The difference between the two preferred frequencies is almost invariably 30 Hz. These characteristics indicate that the selection of the frequency underlies the feedback mechanism, as proposed by Tam [12] and others. It is concluded that the ladderlike structure is inherent to the acoustic feedback system, which does not agree with the previously mentioned experimental observations of Nash et al. [10] in the case without a ladderlike structure. The reason why the preferred frequencies are 30 Hz apart could not be enucleated in the present experiment. Further experimental investigation is required to unveil this riddle.

IV. Conclusions

The role of acoustic feedback in the frequency-selection mechanism of airfoil TE noise was examined by suppressing the acoustic noise, using a splitter plate attached to the airfoil TE and introducing an acoustic feedback loop using a loudspeaker.

The TE noise was completely suppressed by using a splitter plate with a chordwise length of 60 mm, which corresponds to 15% of the chord length of the airfoil. It was found that this is due to the emergence of turbulence in the boundary layer on the pressure side. Although the broadband spectra of the T–S waves on the pressure side were observed in the absence of acoustic emission, once acoustic disturbances extracted from unsteady pressure fluctuations near the TE on the pressure side were emitted, there was a drastic shift of the T–S waves from the broadband to the discrete spectrum. It can be concluded that an acoustic feedback loop plays an important role in the process of selection of the frequency of TE noise.

Instead of a natural feedback loop, the application of artificial acoustic feedback showed that the airfoil boundary layer favors a particular frequency, which resulted in steplike structures as a function of the distance between the loudspeaker and the TE that are similarly observed in conventional environments with natural discrete TE noise emission.

Acknowledgments

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