RELATION BETWEEN TRAILING EDGE NOISE AND BOUNDARY LAYER INSTABILITY

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Abstract

The purpose of the present paper is to show the acoustic feedback loop plays an important role on the frequency-selection mechanism of the tonal noise emanated from the trailing-edge of 2-D wing. For raveling the frequency-selection mechanism, a thin splitter plate placed behind the trailing edge is used to suppress the tonal noise and then an artificial acoustic disturbance by means of loud speaker is introduced into the flow. The experimental result shows that acoustic feedback loop contributes to generation of the discrete T-S wave. Further experiment also shows that boundary layer only accepts acoustic disturbance with favorable frequency, which results in the step-like structures against the distance between speaker and wing model.

1. Introduction

It is well-known that discrete tones emanated from the trailing edge of the two-dimensional (2-D) airfoil in moderate Reynolds-number flows. Numerous researches regarding this issue have been performed, because these tones are emitted from wind-mill blades, aircraft slat, cooling fans of office appliances and so on. Why the acoustic noise is discrete in such flow system still remains unclear. Most prevailing explanation of the mechanism on the tonal noise generation is ascribed to acoustic feedback loop between trailing-edge (T-E) noise and Tollmien-Schlichting (T-S) wave growing near separated region in airfoil boundary layer on the pressure side of the wing[1].

However, there is no research, to authors' knowledge, which rationally showed that this feedback loop contributes to tonal-noise generation, because T-S instability is inherently unstable to broad-band frequencies. Our first purpose in the present research is experimentally to show that the acoustic feedback system chooses a certain frequency from broad-band components, which are naturally growing in boundary layer. For achieving this purpose, it is primarily necessary to establish how to suppress the tonal noise without altering the basic flow on the airfoil unlike roughness elements or additional excrescences extruded on the wing surface. It was eventually found that a splitter plate added to the trailing edge is effective on noise suppression in accordance with an idea from Roshoko [2]. Under such no tonal noise condition, synthetic feedback loop was newly formed using a loud speaker, whose input signal is obtained from the unsteady pressure sensor at 90% chord position on the pressure side of the wing. If the speaker power was off, the unsteady pressure sensor indicates broad-band T-S disturbances.

Another question on this issue is why the local relation of tonal-noise frequency against free-stream velocity shows step-like structure, although the general trend lies on the curve of 1.5 power of the free-stream velocity. On the other hand, Nash et al. [1] showed no step-like structure in the frequency-velocity relation in experiment conducted in anechoic environment. This result is in contrast with many other experimental results obtained in open channel flow free of sound reflection. Our second purpose is to have a clue of as to whether the step-like structure is generic or not in acoustic feedback system. For this purpose, an artificial sound pressure with given frequency and intensity is emitted from a loud speaker placed outside the flow to introduce marked fluctuation into boundary layer on the wing. Here, sound frequency and speaker position were varied.

2 Experimental Set-up

The experimental set-up is shown in Fig. 1. The experiment was conducted in the low turbulence wind tunnel with an octagonal test-section whose opposite side distance is 810mm in Institute of Fluid Science/Tohoku University. No side wall of the test section corresponding to the pressure side of the model was installed for hot-wire measurement and the other side wall was treated so as to avoid acoustic reflection, and also the anechoic ceiling and floor were newly constructed. The wind model was installed vertically in the middle of the test section. The angle of attack of the model to the free stream is set by the geared motor.

The wing model is made of stainless steel with an NACA0012 cross section whose chord length and spanwise length are 400mm and 750mm, respectively. In order to fully span the model in the test section, an additional dummy section with a length of 60mm made from the foam was put at the upper part of the model. Forty-six static pressure holes are drilled at both sides of the wing to measure surface pressure distributions. Also, twenty-two pressure sensors (Kulite, model XCS-062-5psiD) for unsteady pressure measurement were flush-mounted close to a row of the static pressure taps.

Several splitter plates with different streamwise length in 1mm thickness were prepared smoothly to attach the end of the trailing edge, resulting in preventing vortex shedding of rollup motions near the trailing edge of the model.

The streamwise mean and fluctuating velocities were measured by means of a single constant temperature hot-wire anemometer with a linearizer, which a priori was calibrated in the free stream. The hot-wire sensor is made of 5 micron tungsten wire, whose both sides were copper-plated. The plated parts were soldered on the sharpened brass prongs. So, the unplated sensing part is 1mm in length. The hot-wire probe was mounted on the three-dimensional traversing mechanism. The acoustic sound was captured by a microphone, B&K type 4138, which was located 2000mm behind the trailing edge at an angle of 75 degrees from the center line toward the pressure side of the airfoil. In case that the trailing-edge noise was suppressed by a splitter plate, a loud speaker was used to introduce artificial disturbances, which is placed l=2000mm behind the trailing edge at an angle of 60 degrees from the center line toward the pressure side of the airfoil.

The free-stream velocity U_∞ and the angle of attack of the model was chosen at 18m/s and 4 degrees, which corresponds approximately to a Reynolds number 0.5 million based on the model chord length and free-steam velocity.

3 Results and Discussion

3.1 Static pressure distribution

First of all, pressure distributions around the wing model were measured using the so-called scani-valve to find zero angle of attack of the model to the free-stream, where the distribution should be symmetric. Then, the model was set at an angle of 4 degrees. The pressure distribution at 4 degrees is shown in Fig 2. Also, the figure includes the other distribution with the splitter plate attached to the trailing edge, where the angle of attack was reduced to 3.4 degrees, because the real angle of attack was increased by insertion of the plate. As a result, no difference in pressure distributions on both sides between two cases is found.

3.2 Background noise and effect of splitter plate

Figure 3 shows two typical frequency spectra of the sound pressure level without and with the model in the test section. The sound spectrum without model indicates the background noise, while in the presence of the model at 4 degrees the tonal noise on the basis of the background noise is observed at 482Hz and its higher harmonic frequencies. On the other hand, once the splitter plate with the chordwise length of s=60mm was placed, a series of the discrete noise completely disappeared as shown in Fig. 4.

Subsequently, the other plates were individually investigated instead of the 60mm plate. The result is shown in Fig.5, indicating that the length of 60mm is enough to completely remove the tonal noise.

In order to search why the splitter plate is effective to suppress the trailing-edge noise, velocity distributions were measured in the direction normal to the wall near the trailing edge on the pressure side of the model. In passing, boundary layer is turbulent in the whole region on the suction side. Figures 6(a) and (b) show mean and fluctuating velocity distributions at different streamwise locations. It should be noted that no velocity fluctuation is visible at x/c=0.875, but it is suddenly amplified downstream of 90% chord location, where boundary layer is still kept laminar at the trailing edge, but turbulent boundary layer is established at the end of the plate. This suggests that no coherent fluid motions are shed from both sides of the model, resulting in no tonal noise generation.

From now on, subsequent experiments are made in the presence of the 60mm splitter plate, where no natural tonal noise is emanated.

3.3 Streamwise fluctuation in boundary layer

In Fig.7, power spectra of the streamwise *u*-fluctuation at the maximum location in the normal-to-wall direction at x/c = 0.925 are compared with and without the splitter plate. It is found without splitter plate that a discrete component at 482Hz and its higher harmonics are completely identical to those of the sound pressure, indicating that there is an acoustic linkage between u fluctuation in boundary layer and tonal noise. Another subsidiary indication concerning the existence of acoustic feedback is a bump in spectrum, which consists generically of broad-band components originating from T-S instability if acoustic sound was suppressed. Spectral comparison in Fig.7 shows that a certain frequency in a bump, which is not the most unstable, is selected by acoustic feedback loop.

3.4 Synthetic feedback loop

Further attempt was made to show that acoustic feedback system changes broad-band components drastically into discrete component. In case of no tonal noise, unsteady pressure signal at x/c=0.925 on the pressure side of the airfoil was extracted and filtered to remove lower and higher than T-S frequencies consisting of a bump as seen in Fig.7. This filtered pressure signal was power-amplified and acoustically emitted from a loud speaker. Short-averaged spectra of unsteady pressure at x/c=0.925 before and after the speaker is active, are compared in Figs. 8(a) and (b). Also, the time evolution of short-averaged spectrum is shown in Fig. 8(c), here the power amplifier was on at 7 seconds on the time chart. As soon as the speaker is active, only one frequency is selected from the bump due to T-S instability. This is a direct indication that discrete tone is generated by acoustic feedback. In the case of no tonal noise, additional experiment was conducted to extract significance of the signal phase relationship between streamwise velocity or static pressure fluctuation in the boundary layer and acoustic noise. In replace of unsteady pressure signal, random noise signal, which is made from a function generator, was emitted, whose amplitude is similar to natural T-S disturbances in spectrum as seen in Fig. 7, but phase is random. No effect in boundary layer was found at all. Instead of random noise signal, sinuous signal was continuously emitted. It was found that periodical signal within the frequency range unstable to T-S instability was accepted somewhere into the boundary layer. These experimental results show the phase relation is very important to construct acoustic feedback loop.

3.5 Sound receptivity to boundary layer

There is no doubt that external acoustic sound is internalized somewhere into wing boundary layer. Generally speaking, the wavelength of trailing-edge noise is much longer than that of vortical fluctuations in boundary layer. Therefore, internalization of acoustic sound into boundary layer, which is known as ‘receptivity’ problem, must be sensitive to the frequency and the distance between the wing model and loud speaker. Figure 9 shows the relation of the accepted frequency and the distance l between the trailing edge and the loud speaker. Two frequencies are accepted separately against each given distance. Also, the difference between two accepted frequencies is approximately 30Hz. The ladder-like structure appears, which is seen in similar trailing-edge noise experiments, indicating that the acoustic feedback loop puts a trailing-edge noise frequency under constraint as proposed by Tam [3] or others. This conclusion does not coincide with an observation by Nash et al. [1].

4 Concluding remarks

To ravel the generation mechanism of acoustic noise emanated from two-dimensional airfoil trailing edge, a new technique with the aid of a splitter plate was used to suppress acoustic sound. Once the trailing-edge noise is suppressed, naturally growing T-S waves with broad-band components were observed, indicating existence of the acoustic feedback loop between T-S waves developing in the boundary layer and trailing-edge noise. In order to further enhance this indication, acoustic disturbance, which is made from unsteady pressure fluctuations at x/c=0.925, was emitted into the flow. As soon as the speaker was active, one frequency was selected within the range frequency, which is unstable to T-S instability. This selected frequency is not the most unstable under the present condition.

In case of no tonal noise with a splitter plate, a similar experiment with acoustic forcing was made changing the distance between speaker and wing model. Two frequencies are accepted separately against each given distance. There exists the ladder-like structure in relation between acoustic frequency and distance, indicating that the acoustic feedback loop puts a trailing-edge noise frequency under constraint. This result implies that such constraint makes the ladder-like structure in relation of trailing-edge noise frequency versus free-stream velocity.

References

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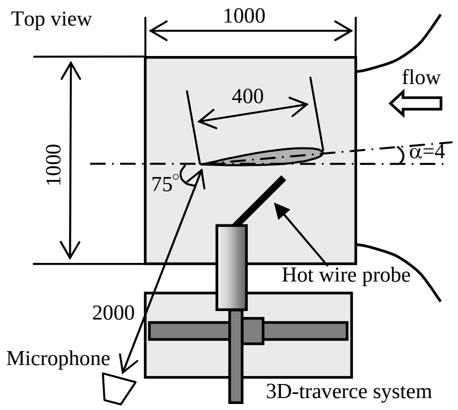

Fig. 1 Wind Tunnel and Model Set-up.

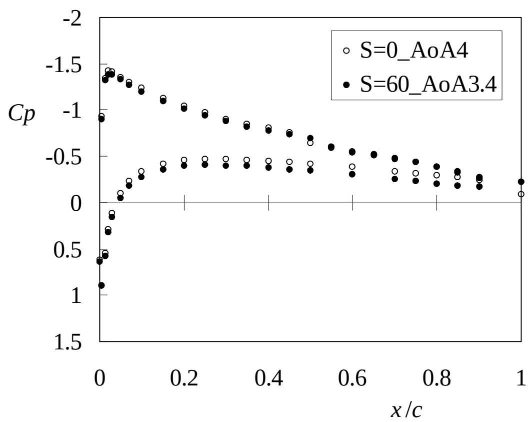

Fig. 2 Static Pressure Distributions with and without a Splitter Plate.

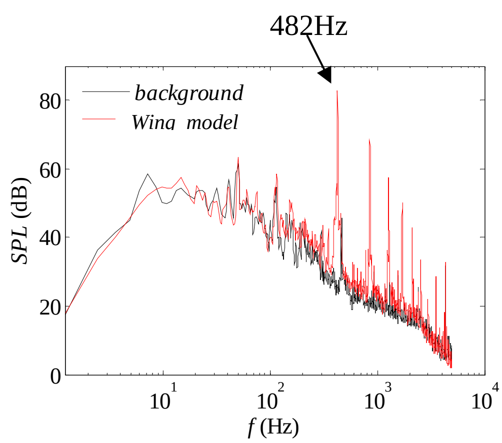

Fig. 3 Spectra of Sound Pressure with and without Model in the Test Section.

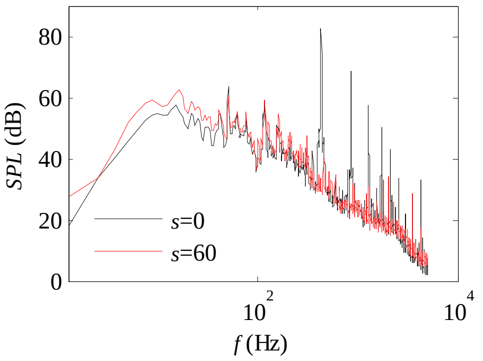

Fig.4. Comparison of Spectra of Sound Pressure Level with and without a Splitter Plate attached to the trailing edge.

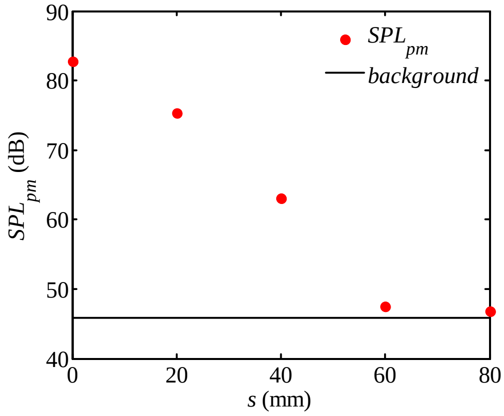

Fig.5. Effect of the Chordwise Length of a Splitter Plate on Tonal Noise.

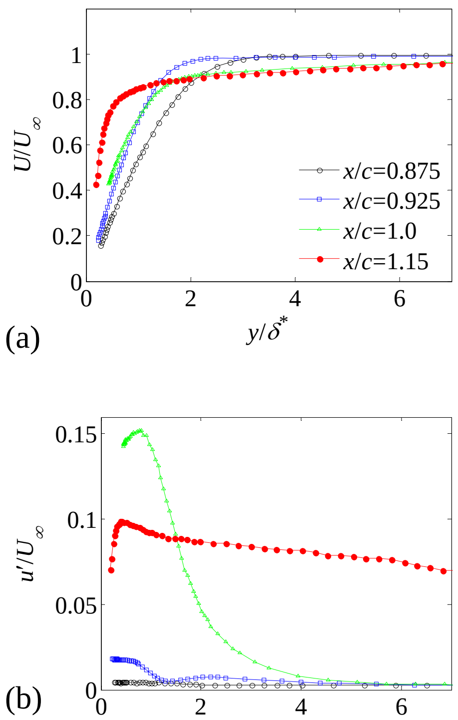

Fig.6. Distributions of (a) Mean and (b) Fluctuating Velocity at Different Streamwise Locations.

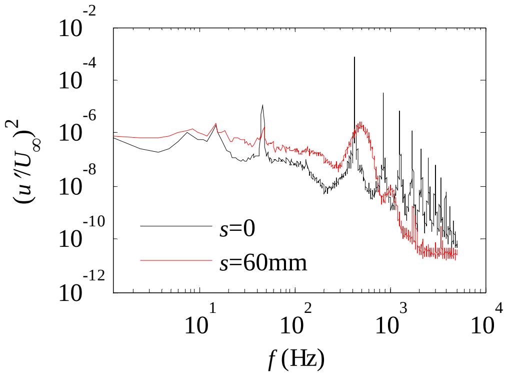

Fig. 7. Comparison of Power Spectrum of the *u*-fluctuation at x/c=0.925 with and without a Splitter Plate.

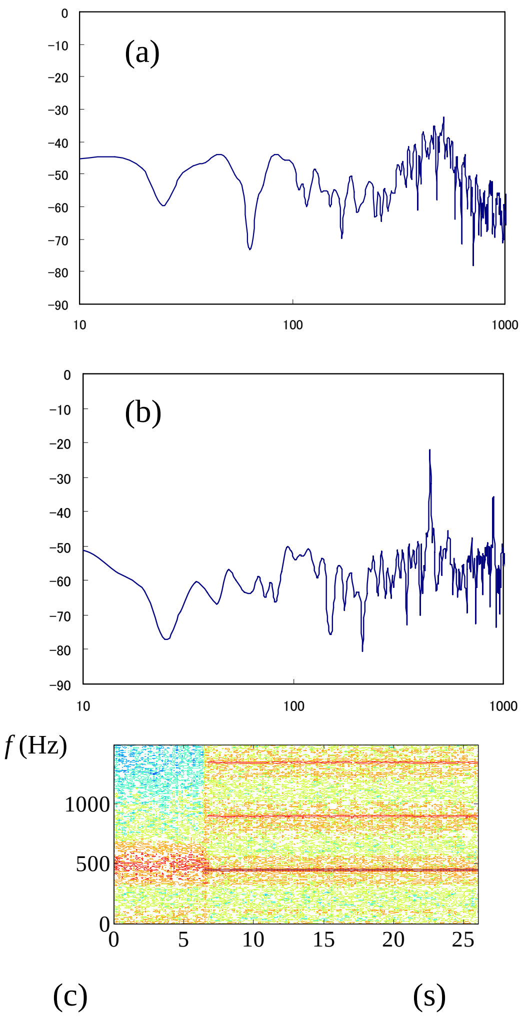

Fig. 8. Time Evolution of Short-Averaged Power Spectrum. (a) Speaker Off (b) After Speaker On at 7 Seconds (c) Spectral Evolution in Time Sequential Chart.

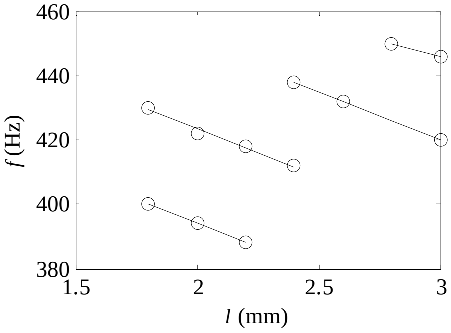

Fig. 9. The Relation between Selected Frequency and Speaker Location. *l* is the Distance between the Loud Speaker and the Trailing Edge.

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