An Experimental Study of Tonal Noise Mechanism of Laminar Airfoils

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This experimental works summarize the (1), aeroacoustic results of airfoil self-noise generated at Reynolds numbers between $1.0 \times 10^5$ and $6.0 \times 10^5$. The test facility is a newly built low noise and low flow turbulence open jet wind tunnel, where a NACA0012 airfoil was mounted to the exit nozzle horizontally. The noise measurements were performed at three effective angles of attack at 0.0, 1.4 and 4.2 degrees. In most cases, multiple discrete tones, embedded in a broad spectrum “hump”, were recorded. The boundary layers on the pressure side of the airfoil were mainly laminar in nature, although instability waves were predicted to grow at the adverse pressure gradient region of the airfoil. (2), aerodynamic data on the airfoil wake development in an attempt to study the noise source. The airfoil incidences were chosen as 0, 2 and 5 degrees. It was found that the radiated tones from the airfoil were always accompanied by localized large scale wake oscillation. The acoustic radiation of this wake tone is believed to be triggered by resonance/feedback between the wake oscillation and the point of first instability on the pressure side of the airfoil. Linear stability analysis on the boundary layer confirmed that the frequency at which maximum amplification of the Tollmien-Schlichting wave occurred always coincide with the main tone noise frequency. It is hoped that the results presented in this paper can shed some lights on the tone generation mechanism that characterise the airfoil noise with laminar boundary layers.

I. Nomenclature

<table>
<thead>
<tr>
<th>Symbol</th>
<th>Definition</th>
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<tbody>
<tr>
<td>$A$</td>
<td>Total amplification rate of the T-S wave</td>
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<tr>
<td>$C$</td>
<td>Chord of airfoil, m</td>
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<tr>
<td>$f_d$</td>
<td>Discrete tone frequency, Hz</td>
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<tr>
<td>$f_{d\text{max}}$</td>
<td>Frequency corresponds to the dominant discrete frequency, Hz</td>
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<tr>
<td>$f_s$</td>
<td>Main tonal frequency, Hz</td>
</tr>
<tr>
<td>$H$</td>
<td>Nozzle height, m</td>
</tr>
<tr>
<td>$n$</td>
<td>Integer modeling the ladder-jumping of $f_{d\text{max}}$, Eq. 2</td>
</tr>
<tr>
<td>$Re$</td>
<td>Reynolds number</td>
</tr>
<tr>
<td>T-S</td>
<td>Tollmien-Schlichting waves</td>
</tr>
<tr>
<td>$v_j$</td>
<td>Jet velocity of the open jet wind tunnel, m/s</td>
</tr>
<tr>
<td>$\theta_{\text{free air}}$</td>
<td>Effective angle of the airfoil determined by Eq. 3, deg</td>
</tr>
<tr>
<td>$\theta_{\text{tunnel}}$</td>
<td>Geometrical angle of the airfoil, deg</td>
</tr>
<tr>
<td>$\nu$</td>
<td>Kinematic viscosity of air, m$^2$/s</td>
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II. Introduction

Research on reducing aircraft noise has been an ongoing process. To achieve this aim it is imperative to understand the noise generation mechanisms, whose are mostly hydrodynamic in origin. High-lift devices in airframe and rotors in aircraft engine are known to produce the so-called trailing edge self-noise. This is a general term used to describe noises that are scattered from airfoil trailing edges, and in this category the acoustic disturbances can possess very different spectrum behaviors and amplitudes. In this paper, we concentrate on the mechanism of “tone” noise generated by an airfoil. Starting from the early 70’s, there have been considerable studies to understand the characteristics and mechanisms of airfoil tone noise\(^1\). When low or moderate Reynolds number flow passes through an airfoil where laminar boundary layer was predominant at the pressure surface, narrowband and high intensity vortex shedding noise, which is widely known as tone, will normally be radiated. Among the publications, Paterson et al.\(^2\) performed a systematic study on isolated airfoil noise in an anechoic environment. They identified the Strouhal number dependency of the tonal noise and then proposed, based on calculation of laminar boundary layer on a flat plate, an empirical formulation as:

\[
f_v = 0.011 v_{j}^{1.5} / (C \nu)^{0.5}
\]

(1)

where \(f_v\) is the main frequency of the tone, \(v_{j}\) is the jet velocity, \(C\) is the chord and \(\nu\) is the kinematic viscosity of air. Another key finding from Paterson et al. is the so called “ladder” structure for the tonal frequency in relation to the flow velocity. Locally the tone frequency follows a velocity dependency of 0.8. Over some selective velocities the dominant tonal frequency is observed to jump to other curves with the same \(v_{j}^{0.5}\). The \(v_{j}^{0.5}\) dependency as shown in Eq. 1 is indeed the average frequency evolution of the discrete tones, also referred to as the main tone frequency.

Using the experimental results of [3], a modified frequency evolution law for the discrete tones was derived by Tam\(^3\) as follow:

\[
f_d = 6.85 n v_{j}^{0.8}
\]

(2)

The term \(n\) is an integer which models the ladder-jumping of the discrete frequency by varying stepwise from one line to other sequentially. In the paper Tam proposed that the discrete tone is caused by a self-excited feedback-loop that comprises of the propagations of hydrodynamics and acoustics disturbances. In this model vortices were shed at the trailing edge into the wake. At some distances downstream of the wake these periodic vortices would be amplified into large scale, localized oscillations\(^4\) where tone noise would be generated. The acoustic wave then radiated upstream to oscillate the boundary layer near the trailing edge, hence forming a loop. A total phase change around the loop should be \(2\pi\), where \(n\) is an integer similar to that of Eq. 2. This aeroacoustic feedback hypothesis was later examined and modified by others\(^5\)-\(^8\). In their models, noise corresponding to the main centre frequency, \(f_s\) (generated by boundary layer instabilities at the trailing edge – a scattering process) is the only acoustic source within the feedback loop. This means that the acoustic wave originated from the vicinity of the trailing edge will propagate upstream to reinforce the point where instabilities were first initiated. The instabilities will grow downstream until reaching the trailing edge, and the loop is said to be completed. Recently, Kingan and Pearse\(^9\) also adopted this model to theoretically predict the tone frequency of laminar airfoil.

The instabilities in the boundary layer mentioned earlier are those of Tollmien-Schlichting (T-S) waves. The boundary layer at the pressure side should be predominantly laminar to allow T-S wave to grow until reaching the trailing edge. Lowson et al.\(^10\) commented that an effective tonal-noise generation is only possible when the T-S waves are amplified by the separating shear layer, which is later confirmed experimentally.\(^11\) Based on linear instability analysis, the radiated tone frequency is similar to the most amplified frequency of the instability in the boundary layer on the pressure side.\(^12\) The arguments of the role of boundary layer at the suction side in relation to the radiated tone remain divisive. Recent DNS study\(^3\) seems to suggest that suction side hydrodynamic scattering is also important. They found that the amplitude of the tone was contributed by the phase difference between the boundary layer instabilities on the pressure and suction sides of the airfoil. However, based on cross-correlation analysis of the fluctuating velocity and acoustic pressure fields\(^13\), low correlation was observed at the suction side and suggestion was made that the boundary layer at the suction side is insignificant to the tonal noise radiation.

Ever since it became an important concern in civil aircraft noise, the understanding of the physics and mechanisms of the airfoil tone noise has improved greatly. However, there are still several unresolved issues in this field. For example, the “structure” of the aeroacoustic feedback loop is not yet conclusive. Also, whether the boundary layer at the suction side is important to the generation of tonal noise remains debatable. In this paper, we concentrate on the first issue – aeroacoustic feedback loop. At first, results from an aeroacoustic measurement of the airfoil tonal noise at different Reynolds numbers (from low to

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moderate) and different angles of attack were presented. The hydrodynamic features related to the airfoil tones were also studied. It is hoped that this paper can produce some information on the fine acoustical structure of the tone and also shed some lights on the tone origin of a laminar airfoil.

### III. Experimental Setup

![Fig. 1](image.png)

**Fig. 1** Experimental set up for the aeroacoustic test of the open jet wind tunnel. M1 – M6 denote the farfield microphones used during the measurement at polar angle, α from 110° to 60° respectively (10° interval). Only results measured by M3 (α = 90°) are presented in this paper.

**ISVR anechoic chamber**

The free field measurement of the airfoil noise was conducted in the newly built open jet wind tunnel in ISVR, which is situated in an 8m x 8m x 8m anechoic chamber. The nozzle exit is a rectangular section with dimension of 0.15m (height) x 0.45m (width). This wind tunnel can achieve turbulence intensity as low as 0.1% and Mach number as high as 0.3, while maintaining a low background noise.

The airfoil under investigation here is a NACA0012 with chord 0.15m and span 0.45m. The airfoil was held by side plates and attached to the nozzle lips, as shown in Fig. 1. The airfoil has an aspect ratio of 3 which is sufficient to maintain a two-dimensional flow around its surfaces. A hinge system on the side plates allows the airfoil to be rotated, thus facilitate different angle of attacks. In this study, airfoil noise pertains to three geometrical angles of attack, 0°, 5° and 15° were examined. Note that the geometrical angles in the context of open jet wind tunnel with a finite nozzle height will be smaller than the effective angles that occur in a free air. This is because testing an airfoil in a finite-size open jet wind tunnel can cause flow curvature and downwash deflection of the incident flow, which affects the pressure gradient distribution on the airfoil surfaces. A 2D open wind tunnel correction scheme to calculate the effective angle, \( \theta_{\text{free air}} \) from the geometrical angle, \( \theta_{\text{tunnel}} \) as follow:

\[
\theta_{\text{free air}} = \theta_{\text{tunnel}} / \zeta
\]

where

\[
\zeta = (1 + 2\sigma) + \sqrt{12\sigma}, \quad \sigma = \frac{\pi^2}{48} \left( \frac{C}{H} \right)^2
\]

The terms \( C \) and \( H \) are the airfoil chord and nozzle height respectively. Using Eq. 3, the effective angles for 5° and 15° are 1.4° and 4.2° respectively in the current study. In the subsequent analysis, the non-zero angle of attack will take reference to the effective angle, \( \theta_{\text{effective}} \). The range of jet velocity under investigation here is 10m/s – 60m/s, which corresponds to Reynolds numbers of 1x10^5 – 6x10^5 respectively based on the chord length. Acoustic data was measured by a microphone located at 1.25m perpendicular to the trailing edge (see Fig. 1). This corresponds to a polar angle of 90°. Noise data was sampled at 30 kHz for 13.33 seconds by a 24-bit National Instrument A/D card. The digitized data was filtered at 15 kHz to avoid signal aliasing. The data was then windowed-FFT and averaged to obtain the power spectra density, \( \text{psd} \) of the noise with a 1Hz bandwidth.
SES wind tunnel

Aerodynamic measurements of the global velocity field pertinent to the same airfoil model were conducted by PIV technique. It is difficult to utilize the open jet wind tunnel to perform PIV measurement due to the limited blow down time. In this study the airfoil model was mounted vertically inside the working section (0.6m x 0.9m) of an open-circuit, suction type wind tunnel in SES, Southampton. Detailed experimental arrangement is shown in Fig. 2. The hydrodynamic development and noise radiation of the airfoil were measured at three angles of attack at 0°, 2° and 5° and freestream velocity at 15m/s. This corresponds to Reynolds number based on the chord length of 1.5x10^5.

Figure 2 also illustrates the measuring system for this study. The PIV system consists of Nd:YAG lasers (50 mJ), a monochrome CCD camera (spatial resolution 1,280 x 1,024 pixels with 12 bits in grey level) and

![PIV system setup](image)

**Fig. 2**  PIV set up for the experiment performed in the closed working section wind tunnel.

![Acoustic spectra examples](image)

**Fig. 3**  Examples of acoustic spectra measured at polar angle of 90° at 1.25m from the trailing edge for, (a). 0° at v_j = 15m/s, (b). 1.4° at v_j = 40m/s, and (c). 4.2° at v_j = 54m/s
a pulse generator. It was positioned to capture area around the airfoil trailing edge and a large area in the downstream wake region. Due to the angle of illumination from the laser, flow at the suction side cannot be measured. During the measurement, care was taken to ensure a uniform ingestion of smoke to the working section.

Despite the apparently high background noise of the wind tunnel, early examination concluded that the discrete tone frequencies possess significantly higher amplitude than the background level. To measure the radiated noise, a microphone was mounted flush to a square panel at the side of the working section. This microphone has a horizontal axis that is perpendicular to the trailing edge of the airfoil at the pressure side (see Fig. 2). The acoustic data was also sampled at 30 kHz for 13.33 seconds. The background noise of the facility, i.e. without airfoil inside the working section, at 15m/s was also measured.

**IV. Results**

**Aeroacoustic assessment of the airfoil tones**

Aeroacoustic assessment of the airfoil noise was conducted in the anechoic chamber. The analysis begins with some of the acoustic spectra in Fig. 3a–c for the 0.0°, 1.4° (θ_{freeair}) and 4.2° (θ_{freeair}) cases respectively. There are several points to note. The noise spectra contain two distinct characteristics – a broadband-“hump” and some equally-spaced discrete tones embedded within. These are consistent with the experimental observation of others.⁶,⁹. The implication of this result is that two types of noise mechanisms may be present in the acoustic spectrum. Another important feature is that the amplitudes of the discrete tones are amplified as the angle of attack increases.

Figures 4a–c show the contours of the acoustic spectra for the above angles. The impact of leading edge noise at low frequency becomes more prominent from 30m/s onwards. Although it is possible to extract the leading edge noise from the spectrum using microphones-pairing technique, it is not attempted here because the noise contributions associated with the leading edge and trailing edge are well separated in the

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![Fig. 4](image-url) **Fig. 4** Comparisons of measured (a-c) and predicted (d-f) acoustic spectra for the laminar airfoil noise at (a,d) 0.0°, (b,e) 1.4° and (c,f) 4.2° angles of attack.
frequency domain, with the latter occurs at a higher frequency.

Using an empirical model\textsuperscript{15}, the predicted tonal noise footprint is shown in Fig. 4d–f respectively. Comparisons between the measured and predicted far field spectra are good for both the acoustic intensity and the main central frequency of the tone, $f_\text{s}$, is shown to increase with flow velocity $v_j$. This exhibits a Strouhal number dependence of the trailing edge noise, thus emphasizing the hydrodynamic origin of the noise source. To confirm the importance of T-S wave on the generation of tonal noise, a tripping element was placed near the leading edge on the pressure side to artificially create a fully turbulent boundary layer at 1.4° and 4.2°. No tone was recorded for the two angles. On the contrary, the tripping element has no diminishing effect on the discrete tone when it was replaced to the suction side.

The mean flow and boundary layer characteristics over the surface of the airfoil were modelled using the XFOIL software. XFOIL is a powerful analysis tool developed by Drela\textsuperscript{16}. Figure 5 shows the estimated onset and end of separation points at the airfoil pressure side for 0.0°, 1.4° and 4.2° at various Reynolds numbers. Note that the separation region was determined by identifying the negative skin friction distributions. Perhaps not surprisingly, the onset of separation is pushed towards the trailing edge as the angle of attack increases. The lengths of separation bubbles are also found to be of inverse relationship with the Reynolds numbers for the 0.0° and 1.4° cases. It was found that the tonal noise begins to diminish from about 40m/s (Re = 4$\times$10$^5$) and this corresponds to the separation bubble encompassing region between 70% – 80% from the leading edge for the 0.0° case. At the remaining 20% the boundary layer is expected to be either laminar or transitional. This is consistent with the observation of Lowson et al.\textsuperscript{10} in that separation bubble disappears from approximately this Reynolds numbers on a NACA0012 due to the initiation of boundary layer transition. Without the separating shear layer acting as the amplifier for the incoming T-S wave noise cannot be scattered effectively at the trailing edge.

At 1.4°, slightly higher intensity noise is still apparent at 50m/s, or Re = 5$\times$10$^5$ (see Fig. 4b) and the corresponding separation bubble at the pressure side encompassed region at 81% – 91% from the leading edge. At 4.2°, the acoustic footprint contour exhibits finer structure of very high amplitude tones as shown in Fig. 4c. Flow separations were found to occur within 81% – 86% of the airfoil chord from the leading edge but do not reattach again at Reynolds numbers between 1.5$\times$10$^5$ – 5$\times$10$^5$. From the figures, higher jet velocity is needed to produce the first tone when the angle of attack is increased. This is attributed to the decrease of T-S instability wave amplification on the pressure side due to the decaying level of the adverse pressure gradient. Larger Reynolds number, compared to the zero angle of attack case, is thus required to sustain the growth of the T-S wave and generate the first airfoil broadband hump/discrete tone.

The frequencies corresponding to the spikes of the discrete tones at some jet velocities were superimposed into the acoustic footprint contour in Fig. 4a–c (indicated by $\bullet$). Following a similar approach\textsuperscript{6}, the frequencies correspond to the highest spike, $f_{d_{\text{max}}}$, at each jet velocity were also shown (indicated by $\Box$). The empirical tonal noise formulations of Eq. 1 and 2 are included in the figure for comparison. The main central frequency (the yellow to reddish colour) follows closely Paterson et al.’s average evolution law of $v_j^{1.5}$ (Eq. 1). On the other hand, although discrete tone is not easily discernable from the acoustic footprint contour due to its qualitative nature (except the 4.2° case), ladder-like structure is still apparent from the variation of $f_{d_{\text{max}}}$ with $v_j$ as shown by the rungs. All the rungs exhibit a 0.8-power dependence on $v_j$ which agrees better with Tam’s equation (Eq. 2).

\begin{figure}[h]
\centering
\includegraphics[width=\textwidth]{fig5.png}
\caption{Predictions of flow separation start and end points (as the fraction of the chord, x/c) at the pressure sides of the airfoil at different angles of attack and Reynolds numbers.}
\end{figure}
The NACA0012 airfoil was removed from the open jet wind tunnel in the anechoic chamber and then mounted inside a closed working section of the SES wind tunnel, from which a detailed PIV study was performed. The flow cases in this study comprise of three angles of attack at $0^\circ$, $2^\circ$ and $5^\circ$, with no correction of angles is necessary. The freestream velocity was set at 15m/s. This value is chosen because the acoustic data will vary considerably among the test cases, and this allows a more wide-ranging study on the tone noise mechanism. Moreover, the background noise of the wind tunnel was not too high when operated in this flow speed. Figures 6a–c show the root-mean-square values of the vertical velocity fluctuation, $v_{rms}$ for $0^\circ$, $2^\circ$ and $5^\circ$ cases respectively. Acoustic spectra pertinent to the above cases are shown in Fig. 6d–f respectively. If the boundary layer disturbances are considered to be spatially growing T-S waves, with a real frequency and a slowly changing complex wavenumber, the growth rates and the total amplification of the perturbations can be calculated by solving the Orr-Sommerfeld equation. In this paper the total amplifications of T-S waves at the pressure side pertinent to the above test cases are shown in Fig. 6g–i.

The acoustic spectrum for the $0^\circ$ case (Fig. 6d) recorded three discrete tones that were above the background noise, with the most dominant ones occurred at 570Hz. Discrete tones were also observed for the $2^\circ$ case (Fig. 6e). The most dominant tone occurred at 575Hz, which was quite similar to the $0^\circ$ case. However, no obvious discrete tone is found for the $5^\circ$ case (Fig. 6f). Cross-examination between the acoustic spectra (Fig. 6d–f) and the predicted total amplifications of T-S waves (Fig. 6g–i) shows remarkable agreement between the experimental results and the linear stability analysis. The discrepancies between the most amplified frequency and the dominant tone are approximately 12% and 2% for the $0^\circ$ and $2^\circ$ cases respectively. Considering the level of approximation in the linear stability analysis, such discrepancies are acceptable for the prediction of the most amplified frequency. For the $5^\circ$ case, T-S wave is

![Fig. 6](image_url)

**Fig. 6** Contour maps of $v_{rms}$ distributions for (a), $0^\circ$, (b) $2^\circ$ and (c) $5^\circ$. Acoustic spectra for (d), $0^\circ$, (e) $2^\circ$ and (f) $5^\circ$. T-S wave total amplification for (g), $0^\circ$, (h) $2^\circ$ and (i) $5^\circ$. Note that all of the above cases are under freestream velocity of 15m/s.
most amplified at about 532Hz but only with a weak amplification factor of the perturbations at 76. This value is significantly lower than the 1.8x10^3 and 0.14x10^3 for the 0° and 2° cases respectively, and it is not sufficient to initiate tonal noise.

Now examining the distributions of v_m in Fig. 6a–c, it was found that tones pertinent to the 0° and 2° cases always coincide with some localized large scale oscillations in the wake. For the 5° case, neither the tone nor large scale wake oscillation was observed. This important result strongly suggests that the dominant tone could also be initiated by instabilities at the wake. Similar hypothesis has also been proposed by Tam (1974). Although it is not yet conclusive as to why there is no broadband hump in the noise spectra for the 0° and 2° cases, the aeroacoustic study in the current work lends support to the notion that the broadband hump and the discrete tones should co-exist at low Reynolds number flows and at low angle of attack. Furthermore, large amplifications of T-S wave are predicted (see Fig. 6g and 6h) which should result in effective acoustical scattering at the trailing edge. The lack of broadband hump in the noise spectra is thus probably caused by the masking effect of the background noise. Near wake interaction of the boundary layer coherent structures at both sides of the airfoil at 0° can develop into a periodic vortex pattern in the wake. At some distance downstream the rolled up vortex will be amplified into large scale velocity fluctuations. The hydrodynamic fluctuation in the wake will radiate acoustic wave with narrowband frequencies. Sound waves will radiate back to and reinforce the first point of instability on the airfoil surfaces to complete the feedback loop.

**At the same Reynolds number,** reduced spatial difference between the localized fluctuation in the wake and the trailing edge can happen even when one increases the angle of attack to a small extent. To provide a possible explanation, it is considered that for the same Reynolds number, the level of adverse pressure gradient at the pressure side of an airfoil will be reduced as the angle of attack increases. In this case the amplification of the boundary layer perturbation is also reduced. At the same time the suction side boundary layer undergoes earlier transition which reduces the inflectional profile near the trailing edge for the amplification of the instability waves. When the instabilities from both sides of the surfaces eventually coalesce in the near wake, the reduced coherence of the resulting vortex shedding would lose its periodicity, and hasten the localized velocity fluctuations as the consequences.

There is also another point to note. It was found that the dominant tone intensity for the 2° case is smaller than the 0° case. The localized velocity fluctuation in the wake is originated from the shedding of the boundary layer instabilities. The smaller amplification of the T-S wave at the pressure side not only reduces the broadband hump generation from the trailing edge (confirmed by the acoustic spectra obtained in the aeroacoustic study), but also reduces the discrete tones that are radiated from the localized region inside the wake.

For the 5° case, the low amplified T-S wave (as predicted from the linear stability analysis) is dissipated effectively into the wake. As the result no localized large scale velocity fluctuation is expected to be produced. When this important component of the feedback-loop mechanism is missing, no tone is generated.

**V. Concluding Remarks**

In this work detailed aeroacoustic and aerodynamic assessments of the laminar airfoil were performed. The measured acoustic spectra generally agree well with the empirical prediction model\[^3\]. Ladder-structure was observed and discussed along with other statistical behaviours of the radiated tones. Several important points can be gleaned from this work. These are:

1. The noise spectra usually comprise of a broadband hump – embedded with multiple discrete tones.
2. When T-S instability wave in a laminar boundary layer is scattered into noise, the dominant frequency is expected to centre on the characteristic frequency of the instability. This is supported by the linear stability analysis in that the mode corresponds to the maximum amplification occurs at the tone frequency.
3. Radiation of noise by scattering the hydrodynamic disturbance at the vicinity of the trailing edge requires the local boundary layer to contain inflectional velocity profile, i.e. in separated state.
4. The intensity of the radiated tone is proportional to the length of the separation region.
5. f_j is found to follow \( v_j^{-1.0} \); while \( f_d \) follows \( v_j^{0.8} \); \( f_{d_{max}} \) also follows \( v_j^{0.8} \), but exhibits ladder-structure. These are true for all the angles of attack examined in this study.
6. The average amplitude of the tone increases as the angle of attack increases (see Fig. 3 and Fig. 4). This is attributed to the downstream-shifting of the separation region at the pressure side, coupled with a strong growth of boundary layer instabilities. In general, the instability noise generation is complex, which depends on the coupling effects of T-S wave amplification, angle of attack (hence...
the adverse pressure gradient level on the pressure side), Reynolds numbers and the extent of the separation region.

7. The boundary layer instabilities, when shed from the trailing edge (and in the process noise was scattered – broadband hump), will propagate into the wake. If the instabilities had been amplified and perturbed strongly in the boundary layer before being shed from the trailing edge, they may developed into a localized large scale velocity fluctuation in the free shear layer which then emit discrete frequency tone. The sound wave propagates upstream to reinforce the location on the airfoil surface where boundary layer instabilities were first originated. A feedback loop that comprises both the hydrodynamic and acoustic fields is then completed.

Acknowledgements

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References