Self-noise and directivity of simple airfoils during stall:  
An experimental comparison

Alex Laratro\textsuperscript{a}, Maziar Arjomandi\textsuperscript{a}, Benjamin Cazzolato\textsuperscript{a}, Richard Kelso\textsuperscript{a}

\textsuperscript{a}School of Mechanical Engineering, The University of Adelaide, South Australia 5005, Australia

Abstract

Noise measurements of NACA 0012, NACA 0021 and flat plate airfoils are obtained at a Reynolds number of 96,000, at angles of attack ranging from -30° to 30°. As the airfoils enter a separated flow regime the strength of the dipolar noise due to shed vorticity increases. This noise source grows in strength over a much smaller range of angles as the thickness of the airfoils is increased, causing the sound pressure level of the NACA 0021 airfoil in this frequency range to increase considerably more sharply than the other profiles. In addition the NACA 0021 test model displays noise behaviour that is consistent with an unstable transition between prestall and stalled flow states, compared to transitions involving intermediate flow states observed for the NACA 0012 and flat plate airfoils.

Keywords:  
Stall noise, Aerodynamic noise, Airfoil noise

1. Introduction

Self-noise of airfoils immersed in a moving fluid is one of the foremost topics of aeroacoustics and is of great interest to both the aeronautical and maritime industries. Much of the work in this area is still based on the pioneering work of Brooks et al. [1] who measured the noise generated by several airfoils at various Reynolds numbers and angles of attack and provided a semi-empirical framework for modelling the noise. Their work was primarily focused on the noise generated by the laminar and turbulent boundary layers at low angles of attack as these are the components of airfoil noise most often encountered by and therefore of most interest to industry. The work of Brooks et al. followed from that of Paterson et al. [2] on airfoil-tip vortex interaction, where a large increase in far-field noise and surface pressure fluctuations was observed between 250Hz and 1000Hz at stall. This phenomenon was subsequently noted by Fink & Bailey [3] when investigating airframe noise control methods, with increases

Email address: alex.laratro@adelaide.edu.au (Alex Laratro)
of up to 10dB in the noise level at stall. These publications agreed that the source of the noise near stall angles was eddies in the separated boundary layer radiating from the trailing-edge. However there has been little interest in the spectrum and directivity of noise generated by stalling airfoils since, as airfoils usually operate outside of the stall regime.

However, recently the wind industry has shown some interest in this noise as when a turbine airfoil stalls its noise spectrum shifts to lower frequencies and its directivity changes, meaning that the noise will travel further in the upstream and downstream directions [4, 5]. Stall can occur on the blades of rotating machinery such as wind turbines, fans, and compressors due to a variety of unpredictable factors such as unsteady inflow operation in the wake of upstream disturbances [6]. In the case of industrial fans, stall can also be caused by too large a pressure change across the fan and is generally avoided using passive stall control systems [7]. In the case of a pitch-regulated wind turbine, the pitch of the blades is constantly modified to maintain a target angle of attack and optimise power generation. A lack of information about the noise generated by airfoils near their stall angle makes optimisation of these factors challenging. As seen in recent noise predictions by Oerlemans & Schepers [8] and Gennaro et al. [9], noise prediction for turbomachinery uses the model of Brooks et al. [1] in order to predict the noise after stall. This approach models the onset of stall as a sudden change in the governing equations when the airfoil stalls, which can fail to capture nuances of the transition.

In recent years Moreau et al. [10] have collected data on the noise generated near stall that indicates that further investigation into the change in noise during the transition to a stalled state may be warranted. It was found that the noise generated by a NACA 0012 airfoil at the onset of stall exhibits broad low-frequency peaks before decaying and giving way to the better-known tones from large-scale separation. This regime, referred to as “light stall”, is of great interest because it is characterised by a non-trivial rise and decay. As unexpected variations in inflow can lead to small changes in angle of attack, properly modelling this transition is important for accurately modelling the sound generated by airfoils experiencing these variations. However Moreau et al. [10] also noted that the experimental conditions greatly affected the noise production. While the results described above are from an experiment with an airfoil with a 130mm span in a 300mm high jet, results from an airfoil of a 300mm span in a 130mm high jet were also present. When the self-noise of the airfoil in the narrower jet was measured, the light stall regime was not observed and the peaks from large scale vortex shedding were formed at a much lower angle of attack. While a computational analysis of the flow-field was performed, it was undertaken on the narrow jet airfoil setup. Analysis of the computational results indicated that there was some coupling between the flow around the airfoil and the shear layer of the jet which may have been responsible for the change in stall pattern. Investigation of the narrow jet stall noise at moderate angles of attack indicated that the main noise source was vorticies produced due to instabilities in the airfoil shear-layer near the leading edge. This is not representative of the usual “deep stall” regime which is characterised by large-scale vortex shedding
similar to that of a bluff body.

Due to the limited information about the noise generated near the stall angle, more experiments are required to characterise the noise produced in this regime. More data will both help to direct future noise control research, as well as providing additional validation cases for computational aeroacoustics codes. To that end the present research aims to gain an understanding of the spectrum and directivity of the self-noise of some common airfoils at low Reynolds numbers, which when combined with previous results, will provide a stronger framework for future research in this area. This is achieved by gathering data with a higher resolution in both frequency and angle of attack from multiple locations, and then analysing the directivity of the noise generated under stall conditions.

2. Method

Aerodynamic research reported in the literature often approximates the complex, proprietary airfoil designs found on turbomachines as simpler designs, with 4-digit NACA airfoils such as the NACA 0015 through NACA 0021 often used for aerodynamics research intended to be further developed [11, 12]. For this study the NACA 0021 was chosen as a thick airfoil profile, and data from two other profiles that have been more widely studied was also collected. One of these profiles was a NACA 0012, as a large amount of aerodynamic research has been performed on it, as well as much of the existing research into self-noise near stall. The final test model used was a flat plate, which like the NACA 0012 has been the subject of a large amount of aerodynamic research. In addition to this research, common aeroacoustic models use a thin plate as their basis [1, 10, 13].

Data were collected in the open-jet Anechoic Wind Tunnel at the University of Adelaide. The wind tunnel consists of a 75mm high by 275mm wide rectangular nozzle in a 1.4m by 1.4 by 1.6m chamber which is acoustically treated with foam wedges, making it anechoic down to 250Hz. Due to the low nozzle height, attempting high angle of attack airfoil experiments in this wind tunnel using a horizontally mounted airfoil would result in a large amount of shear-layer-airfoil interaction which will heavily affect the results [10]. Because of this the airfoil was mounted vertically, which significantly increases the distance between the airfoil model and the jet’s shear layer but decreases the airfoil surface area and therefore the noise produced. The data were collected at a free-stream velocity of 30ms\(^{-1}\) which corresponds to a Reynolds number of 96,000. This Reynolds number was chosen as aerodynamic data for the NACA 0021 at \(Re = 100,000\) had been previously recorded at the University of Adelaide, and more \(Re = 100,000\) data is present in the literature for the other test models. Using \(Re = 96,000\) also enables comparison with the noise data of Moreau et al.[10] which were recorded at a Reynolds number of 150,000.

2.1. Test models

Due to the restrictions of the nozzle, the span of the test model was limited to a maximum of 73mm, providing a 1mm clearance from the end-plates. A
50mm chord was chosen as a compromise between aspect ratio, planform area and blockage. A larger chord would result in a lower aspect ratio and more interference with the jet which was undesirable, and a smaller chord would further reduce the surface area and adversely impact the signal-to-noise ratio. In addition to this the true angle of attack, which is the angle of attack experienced by the airfoil after accounting for jet deflection, is given by $\alpha_t = \alpha_g / \zeta$, where $\alpha_g$ is the geometric angle of attack and $\zeta$ increases non-linearly with the ratio between chord and jet height. In the case of this experiment $\zeta \approx 1.3$ in a vertical configuration used compared to a value of 2.45 in a hypothetical horizontal configuration, which is overly restrictive for high angle of attack measurements.

The airfoil parameters and those from the work of Moreau et al. [10] are compared in Table 1. The relative flow width is the ratio of jet width to airfoil chord. Low values of this parameter result in interaction between the airfoil wake and the jet shear layer, leading to corruption of the noise measurements [10].

<table>
<thead>
<tr>
<th>Present work</th>
<th>Moreau et al. [10]</th>
</tr>
</thead>
<tbody>
<tr>
<td>Nozzle size</td>
<td>75 x 275mm</td>
</tr>
<tr>
<td>Airfoil profile</td>
<td>NACA 0012</td>
</tr>
<tr>
<td>Chord</td>
<td>50mm</td>
</tr>
<tr>
<td>Span</td>
<td>75mm</td>
</tr>
<tr>
<td>Thickness</td>
<td>6mm</td>
</tr>
<tr>
<td>Aspect ratio</td>
<td>1.46</td>
</tr>
<tr>
<td>Relative flow width</td>
<td>5.5</td>
</tr>
</tbody>
</table>

The airfoil test setup, shown in Figure 2, consists of the test airfoil coupled to a brushless DC motor and held between two acrylic end-plates. These end-plates constrain the flow and reduce the rate at which the shear layer forms as shown in Secton 2.2. The inward-facing surfaces of the end-plates were smooth except for the holes for the airfoil shaft, with all of the mounting components located on the opposite side.

Figure 1: Experimental setup in the Anechoic Wind Tunnel
One potential downside to using large end-plates is the introduction of hard surfaces that can scatter and reflect the sound produced by the airfoils. Any specular reflection from the plates may increase the amplitude of sound received from ±25% to ±50% span, which will increase the contribution to the measured sound from the boundary layer. This will introduce some uncertainty to the results with regards to the top of the boundary layer where the velocity is still close to the free stream, but as the noise sources being investigated scale with $U^5$ the noise generated by the segment of the airfoil inside the boundary layer will quickly become negligible regardless. Scattering of the sound will generate a diffuse secondary sound field which may obscure the directivity somewhat. But as the focus of the current work is on changes in the sound field the effect on the results due to the use of end-plates was not considered a high priority versus the advantages of their use. These advantages were to introduce space between the nozzle and the airfoil to increase the available arc for directivity data collection and space behind to allow development of the wake. It was observed during the experiment that there is an increase in turbulence as the flow exits from the end-plates, and when a shorter plate setup was trialled to improve the signal-to-noise ratio the large-scale vortex shedding noise expected at high angles of attack was not observed. This indicated improper development of the airfoil wake under these conditions, so testing was resumed with the extended plates. Another secondary trailing edge source is expected to form where the flow leaves the end-plates, as shown in Figure 2a but since the elevation angle between these locations and the microphones is small (Figure 2b) and the directivity scales with the squared sine of this angle the contribution from this source at the microphones will be small, with a directivity factor of -45dB for sound generated at the airfoil junction and -39dB for sound generated at the edge of the plate.

![Diagram showing primary and potential secondary noise sources and angles between end-plate surface and microphones](image)

Figure 2: Illustration of low directivity between end-plates and microphone array

### 2.2. Side-plate boundary layer

In order to investigate the effect that the end-plates had on the flow field the velocity and turbulent intensity were measured using a normal hot-wire...
probe. Measurements taken in the y-z plane (Figure 1b) at the rotation axis of the airfoil show that the shear layer of the free jet has expanded to almost 50% of the airfoil span by this location and the turbulence intensity is greater than 2% for over 90% of the airfoil as shown in Figure 3. However with the plates installed to constrain the flow the shear layer spans less than 30% of the channel and the turbulence intensity is also greatly reduced, maintaining a value less than 0.5% for much of the span.

![Flow profiles at x = 0, y = 0, $U_\infty = 30$ m/s](image)

Figure 3: Flow profiles at $x = 0$, $y = 0$, $U_\infty = 30$m/s

As the airfoil angle-of-attack is increased the boundary layer changes, which must also be taken into account. Hot-wire measurements of the flow in a plane behind the NACA 0021 test model at geometric angles of attack of 14° and 15° (Figures 4 & 5) show that when the airfoil experiences stall the proportion of the span affected by the boundary layer increases from approximately 30% to approximately 50%. However as will be shown in Section 3.4 it is possible to determine that the source of the observed sound is not within the boundary layer.
2.3. Data collection & processing

Acoustic data were gathered using a polar microphone array consisting of two arcs of 16 GRAS 40PH microphones from $-110^\circ$ to $-35^\circ$ and $35^\circ$ to $110^\circ$ with an angular spacing of $5^\circ$. GRAS 40PH microphones are free-field array microphones with a frequency response of $\pm 1$ dB from 250Hz between 50Hz and 5kHz. The array was positioned such that the microphones were located in the midplane of the airfoil and centred on the rotation axis, as shown in Figure 6.
Acoustic data were gathered in ten time series of fifteen second duration at a sampling frequency of 32kHz. When processing the data each of the ten time series were converted to a power spectral density using Welch’s method with a Hann window, a sample length of 8192 points, and an overlap of 50%. Initially the data were processed using a Hamming window, however it was found that the spectral leakage from the very low frequency pressure fluctuations in the wind tunnel was too great. The data were then reprocessed using a Hanning window which exhibits a much faster side-lobe roll-off, reducing spectral leakage as shown in Figure 7. While not shown in Figure 7 the global maximum of the spectrum is 98dB at 4Hz which is considerably higher than the rest of the data. Using these parameters gives a resolution bandwidth of 4Hz which sacrifices some of the definition of the narrow peaks but makes the spectra clearer. Taking the root-mean-square amplitude of the ten resulting spectra resulted in the final spectra shown in this article which represent a total of 1190 averages. These parameters are also collated in Table 2.
The data were collected by rotating the airfoil clockwise into positions from -5° to 40°, holding stationary every 1° for recording and then counter-clockwise from 5° to -40° in 1° increments, again holding stationary in 1° increments for recording. Once this was done for both of the airfoils, the process was performed a second time to show repeatability and check for any effect removal and replacement the airfoils may have had on the results.

Each time one of the airfoils was installed there was a tolerance of ±1° in location of the 0° datum. This is a source of error but not a cause for concern as determining the absolute angle tack at certain phenomena occur is not a focus of this research, and in cases where the error is greater than 1° it was identified and corrected for when examining the results, as they are expected to be symmetrical between ±5°.

In addition to the main dataset, a second dataset was obtained at the end of the experimental campaign in order to further investigate the rate of change of noise levels as the airfoils enter stall. This has been previously investigated at an angular resolution of $\delta \alpha_g = 0.5^\circ$ [14] but in the current work it is revisited with an angular resolution of $\delta \alpha_g = 0.1^\circ$. Nominaly the geometric angles of attack range from 11° to 16° for the NACA 0012 and 13° to 17° for the NACA 0021, but as the angular resolution is far higher than the error in the location of the 0° datum, only differences in angle of attack between two points in a set of contiguous data are meaningful. This could have been mitigated by gathering data about 0° before moving to the angles of attack of interest, but due to experimental constraints this would have required a reduction in the range of angles investigated.

Velocity data were acquired using a 2.5mm long, 5μm diameter, tungsten single-wire hot-wire anemometer at a sampling frequency of 32kHz. This frequency is higher than the cutoff frequency of the probe use, but because the data were acquired simultaneously with data from a reference microphone in order to cross-reference the sound and velocity measurements, and obtain magnitude-squared coherence, the frequencies could not differ. The hot-wire was positioned using a three-axis traverse with 6.25μm positional accuracy. While sample times and positional resolutions varied depending on the data (as detailed in Section 3.4), all measurements were taken in a single plane behind the airfoil at $x = 1.75c$ as shown in Figure 8b. The measurements presented below were primarily taken in the y-direction at $z = 0$, and in the z-direction at $y = -0.5c$. $y = -0.5c$ was chosen because it was the location where the signal from the Von Karman vortices shed from the trailing edge was expected to be the strongest based on the results of Rodriguez et al. [15], as shown in Figure 8a.
3. Results

In order to fully understand the differences between the noise generated by each of the three airfoils, four different visualisation methods are used. Firstly, a series of pseudocolour plots shows the relationship between frequency, angle of attack and the power spectral density of the sound pressure level. Then standard 2D spectra are used to focus on some of the changes in the spectrum with angle of attack as the airfoils enter the “light stall” regime and the changes in the frequency-integrated sound pressure level with angle of attack integrated of some of the spectral features of the spectra are displayed for the $\delta \alpha_g = 0.1^\circ$ case. Finally, some of the frequency-integrated sound pressure level of some of the aforementioned spectral features are displayed for $\delta \alpha_g = 1^\circ$ in directivity plots. Unless otherwise specified, the data presented in this section are from the first of the two experimental sessions with a clockwise rotation direction.
3.1. Pseudocolour plots

![Pseudocolour plot](image)

Figure 9: Example SPL PSD (dB/Hz re 20⁻¹⁰⁰Pa) of the NACA 0012 airfoil with noise regimes labelled.

Three noise regimes can be immediately identified from the plots of airfoil spectra versus angle of attack, shown labelled in Figure 9 for the NACA 0012 airfoil. These regimes are low angle of attack tonal trailing edge noise, high angle of attack "deep stall" tonal noise and a third regime between the two where the noise is concentrated into a low frequency band but exhibits less tonality. This third regime represents the onset of the "light stall" described by Moreau et al. [10], appearing quickly when the airfoils reach their respective stall angles and then slowly fading before the high angle of attack tonal noise peaks begin to form. As the high angle of attack peaks represent the large-scale "deep stall" vortex shedding from the fully-separated boundary layer, it is thought that this transitional regime represents an intermediate separated state with a different vortex shedding pattern.
The formation of peaks at the onset of stall occurs over a smaller range of angles of attack for the NACA 0021 than the NACA 0012, as previously discussed [14]. As there are recordings from either side of the airfoil in this data set the magnitude-squared coherence between the signals on either side of the chamber can be determined (Figures 11a, 11b & 11c). In this case a higher value of coherence at a given frequency indicates that the signals received on either side of the tunnel are generated by the same source. Coherence can be lowered by the presence of measurement noise, but will also be lower if there are multiple sources or if the sources are spatially distributed over a large enough area [16].
3.2. Spectrum plots

In this section the measured sound pressure level power spectral density from the experiment is presented. Figure 12, show the noise measured from the NACA 0012, NACA 0021 and flat plate models respectively at angles of attack from when they begin to exhibit the “light stall” noise described by Moreau et al. [10] until it reaches its peak strength. The primary difference between the noise generated by the NACA 0012 and NACA 0021 airfoils at light stall is a slight difference in the strength of the noise between 400Hz and 700Hz, making the stall noise portion of the NACA 0012’s spectra in this range appear flatter. Overall this means that the total noise generated at light stall by the NACA 0012 is a few dB higher, as further discussed in Section 3.3.

It is also immediately apparent that the rates of onset of the light stall noise signatures for these airfoils differ from one another, with the NACA 0012 entering an intermediate state where the noise generated between 400Hz and 700Hz has increased far more rapidly than that from 200Hz to 400Hz. This feature does not appear in the NACA 0021 data which appears to immediately transition to the light stall noise signature when moving from 10.7° to 11.4°.

The flat plate spectrum in Figure 12c show a much slower formation of the light stall peaks as the angle of attack increases. The light stall spectrum for the flat plate is more similar to that of the NACA 0012 than the NACA 0021, with a relatively flat distribution and the formation of a broad peak between 400Hz and 700Hz before the portion of the noise between 200Hz and 400Hz forms.
3.3. Noise levels

Integrating across the stall peaks and plotting the sound pressure level for the high-resolution data reveals more information about the rate at which the stall signatures form. As shown in Figure 13 the trends from the $\delta \alpha_g = 1^\circ$ data continue in the $\delta \alpha_g = 0.1^\circ$ data. There remains a smooth transition between the NACA 0012’s pre-stall and “light stall” states, whereas the transition appears even more abrupt and rapid for the NACA 0021 as it still occurs over very few data points. For the NACA 0021 there is a decrease in noise level during light stall between 676Hz 1044Hz, and the noise level in this frequency range also behaves differently to the noise at lower frequencies. Conversely the flat plate noise levels maintain a similar shape for the three frequency ranges inspected, and transitions over a much larger range of angles of attack.
It is interesting to note that while the noise data for the NACA 0021 shown in Figure 13 appears to show a transitional state between pre-stall and light stall at 10.9°, this is not actually the case. In Figure 14, which shows the data from each of the ten recordings that comprise the final spectrum at this angle, it can be seen that the pre-stall and light stall states are being alternately observed. The NACA 0021 flow field only exhibits either the stalled or pre-stall behaviour in each sample indicating that what appears to be a transitional state, like those observed for the NACA 0012 and flat plate, are actually the result of flow instability. Once the angle of attack is further increased the noise observed settles into a steady light stall behaviour.
3.4. Wake measurements

The results from Section 3.3 were used in order to target a study of velocity spectra downstream of the airfoil. Initially 10 second samples were collected without the airfoil in order to determine if any of the observed noise was being generated by the flow upstream. As shown in Figure 15 there is a peak in the velocity spectrum near 50Hz which corresponds to a peak in the sound spectra. This indicates that this noise is being generated by a flow oscillation stemming from an upstream source. There is no matching signal for the peak in the background noise around 100Hz indicating that this stems from a mechanical source, most likely the wind tunnel fan.

Subsequently investigation of the NACA 0021 airfoil, again using 10 second samples, at true angles of attack of $10.7^\circ$ and $11.4^\circ$ (shown in Figure 16) shows that when the light stall signature in the spectrum forms a corresponding broad peak appears in the velocity spectrum. The coherence between the velocity and sound measurements also greatly increases when this happens, indicating the sound is generated upstream of the observation plane. This peak in the
velocity spectrum is observed on both sides of the airfoil for approximately one chord length, but not in the immediate wake where the spectrum is dominated by a much larger broadband signal.

![Figure 16](image_url)

Figure 16: Velocity-sound pressure coherence of the NACA 0021 airfoil at z = 0mm, x = 1.75c, δy = 2.5mm, δz = 2mm

It can be seen that apart from the signals that are found in the flow without the airfoil installed there is little coherence in the velocity and sound spectra. However once the angle of the airfoil is increased and stall occurs peaks appear in the coherence that correspond to the light stall noise. The coherence disappears inside the wake, which is attributed to the large broadband velocity fluctuations in this region. The data shown in Figure 16 were then used to determine the location where the velocity and sound spectra are the most coherent for further study.

When investigating the spanwise distribution of velocity-sound pressure coherence, 30-second-long data samples were collected at locations from z = -40% to 40% span with δz = 2mm. An investigation of the spanwise coherence of the sound and velocity (Figure 17) indicates that the source of the light stall noise is concentrated in the mid-span, outside the plate-foil boundary layer. This indicates that the primary sound source is located near the z = 0 plane of the airfoil.
Reprocessing the data with frequency bands of $\delta f = 64\text{Hz}$ instead of $\delta f = 4\text{Hz}$ we can obtain a better idea of the coherence between the observed sound and wake velocity spectral peaks in their entirety. As seen in Figure 18, between 25% and 32% of the acoustic energy in the 288-352Hz frequency band is coherent in the central $\pm 20\%$ span, and this value falls away as the measurement location moves towards the edges of the test section. While 32% coherence appears low, it should be noted that this is expected due to the distance between the airfoil and the measurement location. Figure 19 shows an estimate of coherence loss between two points with chordwise separation, developed by Bertagnolio et al. [17]. This figure indicates that, for the current experimental setup, the expected coherence between surface pressures separated by a distance of $1.07c$ is between 0.3 and 0.4, depending on the choice of flow separation point. While the experimental results presented in Figure 18 are separated from the trailing edge and not measured along a hard surface, the model gives an indication that the expected coherence and the measured coherence are similar.
Figure 19: Velocity-sound pressure coherence of frequency bands of interest for the NACA 0021 airfoil at $y = 0.5c$, $x = 1.75c$, $\alpha_t = 11.4^\circ$, $\delta z = 2\text{mm}$ with $\delta f = 4\text{Hz}$

Figure 20 presents the velocity spectra from the $z = 0\text{mm}$ measurements as the airfoils stall. The data show that the velocity spectrum peaks at a frequency of approximately 300Hz, which corresponds to the light stall noise observed in the sound spectra (Figure 12). The formation of this peak and increase of the velocity power spectral density in this frequency range correspond to the change in noise level between given angles of attack. Because of these correlated observations and the coherence between the velocity and sound measurements it is likely that these velocity fluctuations are generated by the source of light stall noise.
<table>
<thead>
<tr>
<th>Frequency (Hz)</th>
<th>Velocity PSD (dB/Hz re 1m/s)</th>
</tr>
</thead>
<tbody>
<tr>
<td>10</td>
<td>-100</td>
</tr>
<tr>
<td>2</td>
<td>-90</td>
</tr>
<tr>
<td>10</td>
<td>-80</td>
</tr>
<tr>
<td>3</td>
<td>-70</td>
</tr>
<tr>
<td>10</td>
<td>-60</td>
</tr>
<tr>
<td>3</td>
<td>-50</td>
</tr>
<tr>
<td>10</td>
<td>-40</td>
</tr>
<tr>
<td>3</td>
<td>-30</td>
</tr>
<tr>
<td>10</td>
<td>-20</td>
</tr>
<tr>
<td>3</td>
<td>-10</td>
</tr>
</tbody>
</table>

(a) NACA 0012

<table>
<thead>
<tr>
<th>Frequency (Hz)</th>
<th>Velocity PSD (dB/Hz re 1m/s)</th>
</tr>
</thead>
<tbody>
<tr>
<td>10</td>
<td>-100</td>
</tr>
<tr>
<td>2</td>
<td>-90</td>
</tr>
<tr>
<td>10</td>
<td>-80</td>
</tr>
<tr>
<td>3</td>
<td>-70</td>
</tr>
<tr>
<td>10</td>
<td>-60</td>
</tr>
<tr>
<td>3</td>
<td>-50</td>
</tr>
<tr>
<td>10</td>
<td>-40</td>
</tr>
<tr>
<td>3</td>
<td>-30</td>
</tr>
<tr>
<td>10</td>
<td>-20</td>
</tr>
<tr>
<td>3</td>
<td>-10</td>
</tr>
</tbody>
</table>

(b) NACA 0021

<table>
<thead>
<tr>
<th>Frequency (Hz)</th>
<th>Velocity PSD (dB/Hz re 1m/s)</th>
</tr>
</thead>
<tbody>
<tr>
<td>10</td>
<td>-100</td>
</tr>
<tr>
<td>2</td>
<td>-90</td>
</tr>
<tr>
<td>10</td>
<td>-80</td>
</tr>
<tr>
<td>3</td>
<td>-70</td>
</tr>
<tr>
<td>10</td>
<td>-60</td>
</tr>
<tr>
<td>3</td>
<td>-50</td>
</tr>
<tr>
<td>10</td>
<td>-40</td>
</tr>
<tr>
<td>3</td>
<td>-30</td>
</tr>
<tr>
<td>10</td>
<td>-20</td>
</tr>
<tr>
<td>3</td>
<td>-10</td>
</tr>
</tbody>
</table>

(c) Flat plate

Figure 20: Velocity PSD at onset of light stall noise for the airfoil models, $y = 0.5c$, $x = 1.75c$, $z = 0mm$

3.5. Directivity plots

Figures 21 & 22 show how the directivity of the mean-squared pressure varies as the angle of attack increases. The directivity is presented as the integral of the spectra across the frequency range where light stall noise is observed (270-670Hz). This sound pressure level is then normalised so that the directivity value observed at $\theta = 90^\circ$ is equal to the expected convected dipole. When calculating the theoretical dipole, $\theta + \alpha$ is used as the observation angle as the theory assumes that this value is the angle relative to both the chord line and the convection velocity. In the case where the airfoil is at a non-zero angle of attack, the observer-convection velocity angle and observer-airfoil chord angle are often not parallel, and correcting for this effect can lead to the introduction of larger errors [1].

It can be seen that as the airfoils enter the light stall regime the directivity changes to a shape that is in good agreement with the theoretical dipole before shifting back to its original shape as the contribution to the noise level from the dipolar source falls. Outside of the range of angles of attack where the light stall noise is present the noise takes on a roughly rectangular directivity pattern. This is thought to be due to contributions to the measured noise by
the jet noise dominating the measurements. As the jet is trapezoidal in shape and the microphone array is circular and centred on the airfoil, the microphones further away from 90° are closer to the jet, thereby increasing the background noise level at those locations.

Figure 21: Measured directivity of a NACA 0012 airfoil, NACA 0021 airfoil, and flat plate from 280-670Hz, αt = 3-6.9°. Airfoil chord line represented by a straight line.
Figure 22: Measured directivity of a NACA 0012 airfoil, NACA 0021 airfoil, and flat plate from 280-670Hz, $\alpha_t = 7.6-11.4^\circ$. Airfoil chord line represented by a straight line.

3.6. Repeatability

Comparisons of spectra of the airfoils during light stall are shown in Figures 23 & 24. These plots compare the data obtained at the Microphones at $\pm 90^\circ$ in the standard tests with the data obtained at $\mp 90^\circ$ in the tests where the direction of rotation is reversed. The data show that even for each of the four tests the slight differences in the shapes of the peaks at light stall are reproduced. The angle at which the spectra are identical between experimental sessions differs slightly due to aforementioned misalignment of the $0^\circ$ datum, and it can be seen in some of the spectra (such as the anti-clockwise rotation direction spectrum of the first session in Figure 23) do not match as closely as the others. This is because the degree of misalignment is such that the data from that session was captured at angles of attack that differ enough to produce slightly different spectra but not enough to be reliably corrected.
Figure 23: SPL PSD of for at uncorrected “light stall” angles for the four sets of NACA 0012 data

(a) \( \theta = +90^\circ \) for the clockwise rotation and \( \theta = -90^\circ \) for the anti-clockwise rotation

(b) \( \theta = -90^\circ \) for the clockwise rotation and \( \theta = +90^\circ \) for the anti-clockwise rotation

Figure 24: SPL PSD of for at uncorrected “light stall” angles for the four sets of NACA 0021 data

(a) \( \theta = +90^\circ \) for the clockwise rotation and \( \theta = -90^\circ \) for the anti-clockwise rotation

(b) \( \theta = -90^\circ \) for the clockwise rotation and \( \theta = +90^\circ \) for the anti-clockwise rotation

Figure 25 shows that the directivity of the NACA 0012 at 15\(^\circ\) is repeatable for both rotation directions. In this figure the data from the anti-clockwise rotation direction are shown rotated 180\(^\circ\) so that the observation angle is consistent with the data for clockwise rotation direction. The figure also appears to show that while the data are mostly consistent on the suction side of the airfoil, the data from the pressure side is less uniform because the normalisation is referenced to the theoretical dipole at 90\(^\circ\). If the data are normalised with reference to -90\(^\circ\) the situation is reversed. In addition, agreement between the data for each rotation direction indicates that any asymmetry in the microphone array and wind tunnel is not significantly affecting the results.
4. Discussion

All three of the test models display significantly different growth rates of both noise and wake velocity fluctuations in the 280-670Hz frequency range. The NACA 0021 displays the fastest and sharpest growth rate, with the increase in noise level and wake velocity fluctuations at stall occurring over an angle of attack range of 0.2°, with both the flow displaying an unstable transition between prestall and stalled states in this range. Conversely the low frequency noise and wake vorticity produced by the NACA 0012 airfoil displays a much smoother transition over approximately 1°, as does the flat plate with a transition angle of attack range of approximately 6°. This means that for a given rate of change of angle of attack, the rate of change of low frequency noise level will vary significantly between the three airfoil profiles tested, and giving the noise of the NACA 0021 airfoil a more impulsive characteristic compared to the NACA 0012 or the flat plate.

When compared to the results of Moreau et al. [10], the noise level when no airfoil is present (shown in Figure 12) is significantly in the 280-670Hz frequency range. Similarly the noise level with the airfoil installed at a low angle of attack was lower than that seen once the airfoil enters a stalled state, and this may be due to the end-plates that comprise the test section in the present work. While the noise observed after stall is highly coherent with the wake velocity measurements the effects of the hard walls of the test section cannot be entirely discounted. As seen in Figure 15b there is a significant amount of vorticity near the walls of the test section. While this vorticity does not result in coherent noise production at stall (as shown in Figure 17), the observed grow rate of noise between 272Hz and 412Hz prestall may be affected.

Vorticity fields from a computational study of a flat plate by Tamai, indicate
the vorticity produced by the separated shear layer peaks near 7.5° [18]. This angle corresponds to the peak value of noise in observed from the flat plate in this study, indicating that the observed noise may be caused by this vorticity interacting with the airfoil. Furthermore, a direct numerical simulation of a NACA 0012 airfoil performed by Rodriguez et al., for a Reynolds number of 50,000 at angles of attack at 5°, 8°, 9.25° and 12°, indicate a similar pattern[15]. In the study by Rodriguez et al. vorticity grows from 5° to 12° angle of attack, with strong vortex shedding starting in the 9.25° case. Again this corresponds with the angle of onset for the noise in the light stall regime for the NACA 0012 in the present study, shown further in Section 3.3. Unfortunately to the author’s knowledge similar flow fields in this angle of attack range for the NACA 0021 airfoil are not available in the literature. Overall, the available flow-field data from the literature corroborates the results of this study.

5. Conclusion

Self-noise from NACA 0012 and NACA 0021 airfoils at a Reynolds number of 96,000 was measured in the anechoic wind tunnel at the University of Adelaide using a polar microphone array. It was found that at the onset of stall the noise produced in the range of 200Hz-700Hz increased suddenly by approximately 5dB and this was accompanied by a change in directivity to that of a convected dipole. Furthermore, the noise generated by the NACA 0021 airfoil increased by a smaller amount than the NACA 0012 and the flat plate in the range of 400-700Hz, but transitioned to this state much more rapidly than the NACA 0012. A subsequent wake study showed that the growth rate of the low frequency noise level is mirrored by that of wake vorticity in the same frequency range. This indicates that vorticity shed from the leading and trailing edge during light stall, previously reported in the literature, is the mechanism responsible for light stall noise. Even after investigating the transition with an angle of attack resolution of 0.1° no intermediate state between the pre-stall and stall noise regimes could be detected for the NACA 0021, compared with a smooth transition over a range of at least 1° for the NACA 0012 airfoil, and 6° for the flat plate. However, the NACA 0021 airfoil noise spectrum did briefly display an unstable transition between the two noise regimes before settling into producing steady light stall noise.

Acknowledgements

This work was supported by an Australian Government Research Training Program Scholarship


