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Airfoil Noise Reduction Using Flow Control in Blunt Trailing Edge

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Abstract. *Direct numerical simulations (DNS) of unsteady low-to-moderate Reynolds number flows past a NACA 0012 airfoil are carried out to assess the effects of suction and blowing on noise generation by blunt trailing edges. The tonal noise that arises from the interaction of wake and boundary layers with blunt trailing edges is computed for flow configurations at different freestream Mach numbers ($M_\infty = 0.1$ to 0.5), angles of incidence ($AoA = 0$ and 3 deg), and Reynolds numbers based on the airfoil chord ($Re_c = 5000$ and 100000). The influence of compressibility on sound generation is analyzed along with the effects of scattering by blunt trailing edges for different radii of curvature. Numerical results for the low Reynolds number flow studied show that the airfoil emits a single "narrow-band" tone; however, results for the moderate Reynolds number flow analyzed show that the airfoil emits multiple "narrow-band" tones superimposed on a broadband hump, depending on the flow configuration, indicating the existence of an acoustic feedback loop as discussed in literature. For those cases where a broadband hump with multiples tones are observed, trailing edge blowing and suction considerably modify the near-field region responsible for the noise generation. It is shown that the presence of secondary tones is very dependent on compressibility effects as well as angle of incidence and trailing edge bluntness. For some of the flow configurations analyzed blowing can completely eliminate the secondary tones.*

Keywords: *Airfoil Noise, Trailing Edge Bluntness, Direct Numerical Simulation*

1. INTRODUCTION

Airframe noise has become a significant source of noise in landing configurations since jet noise has been reduced due to the introduction of turbofan engines with larger bypass ratios. Currently, airframe noise can be identified as a potential lower noise barrier. Thus, any further reduction of aircraft noise on the approach can only be obtained if both engine and airframe noise are reduced equally Lockard and Lilley (2004); Dobrzynski (2010); Lele and Nichols (2014).

Trailing edge noise is a fundamental airframe noise mechanism Lockard and Lilley (2004). Recently, due to the more stringent noise regulations and, since air traffic and wind power generation has increased, the understanding of trailing edge noise generation and propagation is an overriding concern for the design of low-noise aerodynamic shapes including wings and high-lift components, as well as wind turbine blades, propellers and fans. Brooks *et al.* Brooks *et al.* (1989) identify vortex shedding due to laminar boundary layer instabilities and blunt trailing edges as sources of airfoil self-noise. Wolf *et al.* Wolf *et al.* (2012a, 2013, 2012b) performed numerical investigations of airfoil self-noise generation for turbulent flows past a NACA 0012 and a DU96 airfoil. They showed that tonal noise may appear in far-field acoustic predictions for blunt trailing edges even in the presence of fully turbulent boundary layers. The presence of tonal noise would then depend on both the trailing edge thickness and the boundary layer displacement thickness.

Several pioneering investigations were conducted in the 1970s in order to study airfoil tonal noise. These results showed that discrete tones are emitted from isolated airfoils or helicopter rotors at specific flow conditions Smith *et al.* (1970); Clark (1971); Hersh and Hayden (1971); Longhouse (1977). These findings triggered some of the first systematic, detailed and well known studies of airframe noise Paterson *et al.* (1973); Tam (1974); Fink (1975); Fink *et al.* (1976); Arbey and Bataille (1983). Paterson *et al.* Paterson *et al.* (1973) performed noise measurements from symmetric NACA airfoils with a Reynolds number range between 10^5 and 10^6 with various angles of attack; their results showed the existence of discrete tones and multiple tones in a ladder-like structure pattern in terms of frequency and free-stream velocity. They also proposed an average evolution of the dominant frequency in function of free-stream velocity, U_∞, as follows: f=$0.01U_\infty^{3/2}/(c\nu)^{1/2}$. In this equation, c is the airfoil chord and ν is the coefficient of kinematic viscosity. Furthermore, they measured span-wise surface pressure correlations on the airfoils and found strong correlation over a considerable extent along the airfoil surface. This indicated that the flow phenomenon associated with airfoil tonal noise generation is two dimensional.

Another theory for the power law of Paterson was proposed by Fink Fink (1975). Fink assumed that the discrete frequency was linked to the laminar boundary-layer of the pressure surface. Arbey and Bataille Arbey and Bataille (1983) repeated the experimental studies from Paterson in an open wind tunnel for three different NACA airfoils and showed that the tonal peak was a superposition of broadband contribution centered on a main frequency f_s and a set of regularly spaced discrete frequency tones f_n. The dominant frequency, f_s, was in agreement with Paterson's formula shown above. Arbey and Bataille also found the ladder-like structure from Paterson's original work but other results were in disagreement, for example, the higher magnitudes of the discrete tones and the different tone distribution.

Tam Tam (1974) suggested that the ladder-like structure of frequency as a function of flow velocity was due to a self-excited feedback loop between the trailing edge and the noise source in the near wake. Nash *et al.* Nash *et al.* (1999) performed experimental studies of airfoil tonal noise generation for a NACA 0012 profile with a Reynolds number of 1.45×10^6 and several angles of attack. A closed-working-section wind tunnel with and without an acoustic-absorbing lining on its walls was used in the experiments. The results from the hard-wall tunnel (without lining) revealed multiple frequency peaks, in agreement with the overall $U_\infty^{3/2}$ law for the dominant tones and with the local dependence $U_\infty^{0.8}$ for the secondary tones, as proposed by Paterson. However, the authors argued that these tonal frequency peaks were correlated to the resonant frequencies of the wind tunnel. Thus, they carried out measurements with lined walls simulating anechoic conditions. Under these conditions, it was found that a single dominant tonal frequency was observed with the power law $U_\infty^{0.8}$ instead of several peaks. Furthermore, no ladder-like structure of tonal frequency was observed, in disagreement with the previous studies of Paterson *et al.* Paterson *et al.* (1973), Fink, Fink (1975) and Arbey and Bataille Arbey and Bataille (1983). Nash *et al.* Nash *et al.* (1999) argued that the previous researchers may have been misled by spurious feedback loops which had arisen from the facilities, even in open jet studies.

Desquesnes *et al.* Desquesnes *et al.* (2007) performed 2D direct numerical simulations (DNS) for a flow past a NACA 0012 airfoil for Reynolds numbers 1×10^5 and 2×10^5, and angles of attack of 2 and 5 degs; their results showed the multiple tonal peaks consistent with the experimental observations from Arbey and Bataille Arbey and Bataille (1983). Kurotaki *et al.* Kurotaki *et al.* (2008) and Plogmann *et al.* Plogmann *et al.* (2013) also found multiple tones in their experimental results and the so-called ladder-like structure pattern. Meanwhile, Tam and Ju Tam and Ju (2011) conducted direct numerical simulations (DNS) on a NACA 0012 airfoil for three different trailing edge thicknesses in the Reynolds number range of 2×10^5 to 5×10^5 at zero angle of attack. Under these conditions, their numerical results showed only one airfoil tone for each simulation, in agreement wit the measurements of Nash *et al.* Nash *et al.* (1999) and, supporting the argument that the ladder-like structure pattern of isolated tones is not genuine. In addition, Tam and Ju Tam and Ju (2011) suggested that an airfoil with a thicker trailing edge would have a lower tonal frequency for the same flow velocity.

It is clear that since the 1970s, great efforts have been carried out to improve the understanding of the airfoil tonal noise phenomenon and the authors suggest the paper by Arcondoulis *et al.* Arcondoulis *et al.* (2010) for a review on the topic. As one can see in this reference, there are still several issues and disagreements in literature. Furthermore, there is a lack of detailed and systematic studies regarding the effects of trailing edge bluntness on airfoil tonal noise generation. In a recent study, the present authors Arias Ramirez and Wolf (2015) showed results on the topic for low Reynolds numbers, Numerical results for the low Reynolds number flow, showed that the airfoil emits a single "narrow-band" tone due to vortex shedding at the trailing edge. The present work extends the research on tonal noise generation for moderate Reynolds number flows past blunt trailing edges. Here, we also investigate the effects of trailing edge suction and blowing on the noise generation process. Two dimensional DNS are performed for different rounded trailing edges of a NACA 0012 airfoil as shown in Fig. 1 and Table 1. Flow configurations with different angles of incidence ($AoA = 0$ and 3 deg), Reynolds numbers based on the airfoil chord ($Re_c = 5000$ and 100000) and freestream Mach numbers ($M_\infty = 0.1$ to 0.5) are analyzed.

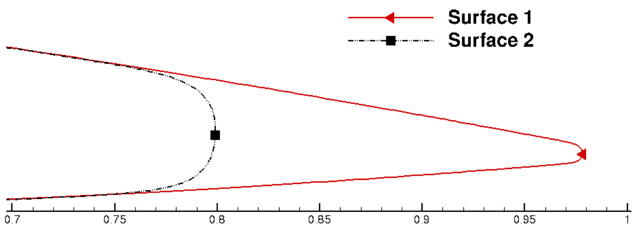
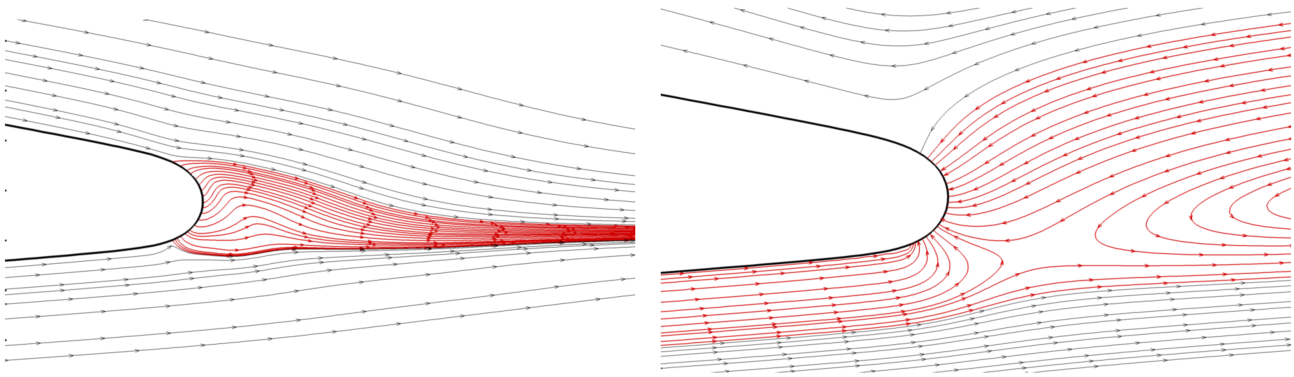


Figure 1. Trailing edges profiles of the configurations analyzed.

Table 1. Details of the configurations analyzed

Airfoil	Chord (c)	TE radius (r)	% r/c
Surface 1	0.98	0.0040	0.408
Surface 2	0.80	0.0250	3.125



(a) Blowing along TE surface 1 (thinner TE). (b) Suction along TE surface 1 (thinner TE).

Figure 2. Examples of suction and blowing along airfoil surface.

2. Flow Simulations and Acoustic Predictions

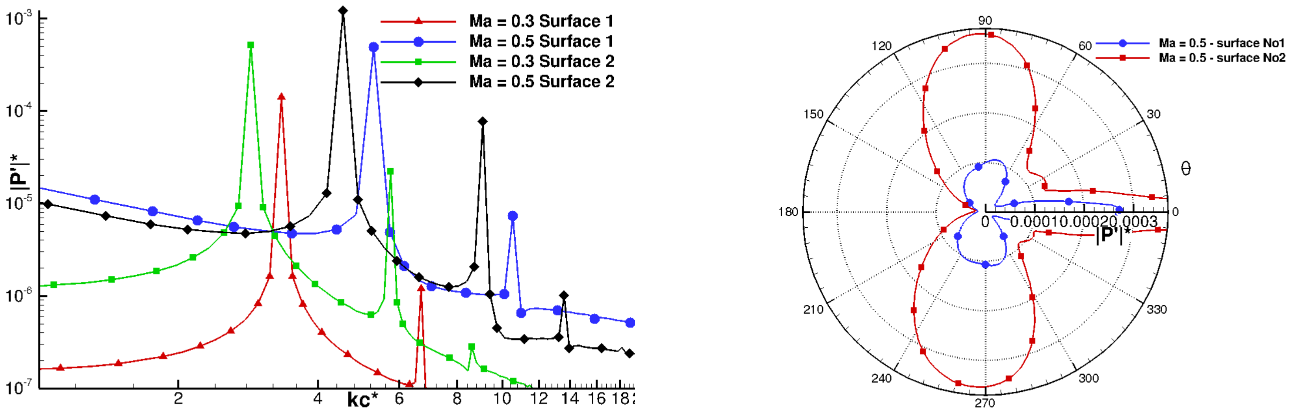
The general curvilinear form of the compressible Navier-Stokes equations is solved in conservation form. The numerical scheme for spatial discretization is a sixth-order accurate compact scheme Nagarajan *et al.* (2003) implemented on a staggered grid. Compact finite-difference schemes are non-dissipative and numerical instabilities arising from insufficient grid resolution, mesh non-uniformities and approximate boundary conditions have to be filtered to preserve stability of the numerical schemes. The high wavenumber compact filter presented by Lele Lele (1992) is applied to the computed solution at prescribed time intervals in order to control numerical instabilities. This filter is only applied in flow regions far away from solid boundaries. The time integration of the fluid equations is carried out by the fully implicit second-order scheme of Beam and Warming Beam (1978) in the near-wall region in order to overcome the time step restriction due to the usual near-wall fine-grid numerical stiffness. A third-order Runge-Kutta scheme is used for time advancement of the equations in flow regions far away from solid boundaries. No-slip adiabatic wall boundary conditions are applied along the solid surfaces and characteristic plus sponge boundary conditions are applied in the far field locations. The numerical tool has been previously validated for several simulations of compressible flows involving sound generation and propagation Wolf and Lele (2010); Wolf *et al.* (2012a); Arias Ramirez and Wolf (2015). A hybrid approach which uses direct calculation for near-field source computations and the Ffowcs Williams-Hawkings (FW-H) equation as the acoustic analogy formulation is applied in the present work. Here, the 2-D FW-H equation is solved in the frequency domain and includes convective effects. The dipole and quadrupole sources are included in the formulation in order to perform an assessment of the individual sources on the far-field noise. More details about the numerical formulation can be found in Ref. Arias Ramirez and Wolf (2015)

Two **O**–type meshes are used in the simulations. The first has 400×700 points in the azimuthal and normal directions, respectively, and it is employed for $Re_c = 5000$ calculations and the second mesh has 400×900 points and it is used for the $Re_c = 100000$ cases. The computational domain extends 45 chords from the airfoil in each direction and characteristic plus sponge boundary conditions are applied in the far–field locations to minimize wave reflections. A mesh refinement study was previously performed and only converged results are presented in the present work. The explicit and implicit meshes have 400×550 and 400×150 points, respectively, and 400×650 and 400×250 points. The former and latter meshes are employed for the $Re_c = 5000$ and 100000 calculations, respectively. The large number of grid points and relatively small domain size ensures that the acoustic waves generated by the airfoil remain well resolved as one moves away from the solid surface. Furthermore, the boundary layers are also accurately resolved along the airfoil surface. Information about the two **O**–type meshes employed in the calculations is summarized in Table 2. Figures 2 (a) and (b) show a schematic representation where steady blowing and suction are applied along the airfoil surface. Different configurations of suction and blowing are tested in the present work, including variations in blowing/suction velocity and location.

Table 2. Characteristics of meshes employed in the simulations.

Re_c	AoA	Implicit Mesh	Total Mesh	dz0	dz1
5000	3 deg.	400×150	400×700	5×10^{-4}	0.3
100000	0 and 3 deg.	400×250	400×900	2×10^{-4}	0.3

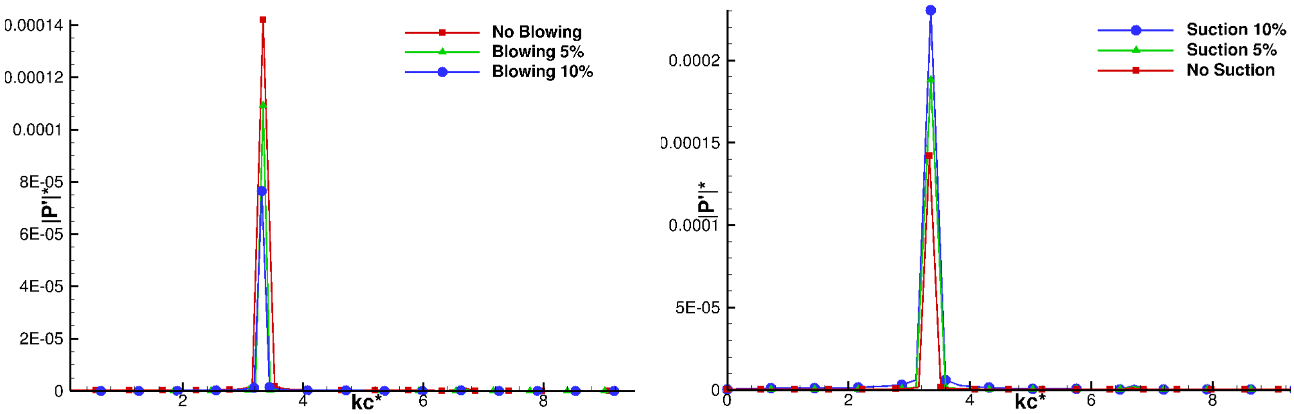
Here, $dz0$ and $dz1$ are the wall and far field mesh resolutions in the wall normal direction. The **O**–mesh domain extends 45 chords from the airfoil in each direction.



(a) Pressure spectra for $M_\infty = 0.3$ and $M_\infty = 0.5$

(b) $|p'|$ for an observer location at $5c$ distant from the airfoil, at vortex shedding frequency, and $M_\infty = 0.5$ for surfaces 1 and 2

Figure 3. Comparison between surfaces 1 (thinner TE) and 2 (thicker TE) for $AoA = 3$ deg and $Re_c = 5000$.



(a) Pressure spectra for blowing of $5\% M_\infty$ and $10\% M_\infty$

(b) Pressure spectra for suction of $5\% M_\infty$ and $10\% M_\infty$

Figure 4. Comparison of the effects of trailing edge suction and blowing for surface 1 (thinner TE), $AoA = 3$ deg, $Re_c = 5000$ and $M_\infty = 0.3$.

3. Results

This section presents results obtained by DNS for flow simulations at low and moderate Reynolds numbers. In order to examine the tonal noise generation, hydrodynamic and acoustic properties are compared for a NACA 0012 airfoil with different blunt trailing edges and at different angles of incidence and freestream Mach numbers. An assessment of the effects of suction and blowing is also presented for the different flow configurations investigated.

3.1 Flow configuration 1: $AoA = 3$ deg. and $Re_c = 5000$

Figure 3 presents a comparison in terms of pressure spectra obtained in the acoustic field for surfaces 1 (thinner TE) and 2 (thicker TE). As aforementioned, the acoustic signature in the far field is more pronounced for a blunter trailing edge. In Fig. 3 (a) one can see that the amplitude peaks are higher for the airfoil with surface 2, for both Mach numbers analyzed. At the same time, the vortex shedding frequency is also reduced for the blunter TE. Figure 3 (b) presents pressure directivity plots for the NACA 0012 with trailing edges of surfaces 1 and 2 for $M_\infty = 0.5$. It is clear that a blunter trailing edge generates higher noise levels for the current flow configuration.

In order to investigate possibilities for noise reduction, steady suction and blowing are performed along the blunt trailing edges analyzed (see Fig. 2 (c) and (d) for a visualization of the regions where suction and blowing are implemented). Figure 4 presents plots of pressure spectra (in linear scale) measured in the acoustic field for surface 1. Blowing and suction are applied in Figs. 4 (a) and (b), respectively. For both cases, the flow velocities are set as 5 and 10% of the freestream velocity, $M_\infty = 0.3$. As one can observe, blowing considerably reduces the far field noise at the specific probe location (above the trailing edge) and this noise reduction is dependent on the blowing velocity. On the other hand, suction increases the far field noise. Similar trends are observed for all freestream Mach numbers investigated.

In Fig. 5, one can see the effects of blowing for the blunter trailing edge analyzed (surface 2). Since suction has demonstrated to increase noise levels, only results for blowing will be shown in the figure. Figures 5 (a) and (b) present mean flow streamlines for the cases without and with $10\%M_\infty$ blowing. The flow is considerably modified in the latter case and flow separation behind the blunt trailing edge is delayed. This modification of the flow field is responsible for a change in the quadrupole field as shown in Figs. 5 (c) and (d). In these figures, one can observe that the overall field distribution of T_{11} is similar, however, peak values of the Lighthill stress terms are moved away from the airfoil surface. In Fig. 5 (e), one can observe the impact of the quadrupole field modification on the far field acoustic pressure. It is possible to see a large noise reduction in the amplitude of the vortex shedding tonal peak when blowing is applied.

3.2 Flow configuration 2: $AoA = 0$ deg. and $Re_c = 100000$

The next flow configuration analyzed consists of a NACA 0012 airfoil at $AoA = 0$ deg. and $Re_c = 100000$. For the present Reynolds number, one could expect near and far field pressure spectra with the presence of secondary tones superimposed on a broadband hump. We start showing pressure spectra obtained in the near field, where noise generation occurs, for $M_\infty = 0.1$ and 0.3 in Figs. 6 (a) and (b), respectively for surface 1. One can see that compressibility effects play an important role in the appearance of the secondary tones. For the lower Mach number flow analyzed, the main tonal peaks are easily observed but the presence of secondary tones is questionable. On the other hand, for the higher Mach number, the secondary tones are clearly present together with the broad hump. Figures 6 (c) shows mean flow streamlines for $M_\infty = 0.1$. the mean flow is symmetric and, for Fig 6 (d), stronger instabilities along the boundary layer are developed and the top surface of the airfoil presents a separation bubble, causing the flow to be non-symmetric.

Figure 7 shows results a comparison in terms of pressure spectra for the effects of suction and blowing along the trailing edge for the NACA 0012 airfoil with surface 1 at $AoA = 0$ deg, $Re_c = 100000$ and $M_\infty = 0.3$. One should remember that, for this case, the flow is non-symmetric and secondary tones as well as a broadband hump, are clearly seen. In Fig. 7 (a) spectra are shown in log scale and in Fig 7 (b) they are shown in linear scale for better visualization of the tonal peaks. From these figures, one can see that both suction and blowing tend to reduce the frequencies and amplitudes of the main peaks. However, these disturbances cause the formation of stronger secondary peaks and suction, in particular, introduces a strong broadband content to the spectrum.

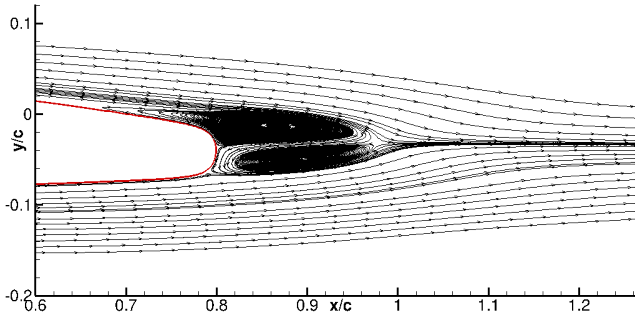
Figure 8 presents a comparison of results obtained by the NACA 0012 airfoil with surface 2. One can see a plot of pressure spectra for the flow configurations with and without blowing in Fig. 8 (a). For the present moderate Reynolds number, the spectrum from the blunt trailing edge shows the regular tonal peaks associated with the vortex shedding frequency and similarly to the low Reynolds number case analyzed, blowing reduces the amplitudes of the tonal peaks. In Fig. 8 (b), the magnitudes of pressure fluctuations along the airfoil surface are displayed at the vortex shedding frequency. When blowing is applied at the trailing edge, the scattered pressure is considerably reduced.

As observed for the low Reynolds number flow studied in case 1, the spatial distribution of quadrupole sources is severely affected by blowing for surface 2. These results can be seen in Figs. 8 (c) and (d) which show that quadrupole sources vanish close to the trailing edge surface when blowing is applied.

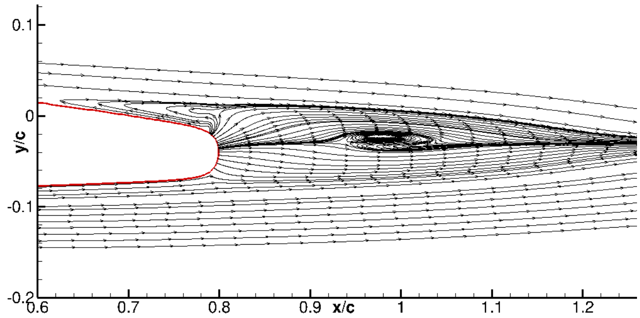
3.3 Flow configuration 3: $AoA = 3$ deg and $Re_c = 100000$

Flow configuration 3 consists of a NACA 0012 airfoil at $AoA = 3$ deg and $Re_c = 100000$. Again, the effects of compressibility, trailing edge bluntness and suction & blowing are assessed. Figure 9 shows results for the thinner trailing edge analyzed, surface 1. In Figs. 9 (a) and (b), pressure spectra can be compared for the undisturbed case and also when blowing and suction of $5\%M_\infty$ are applied. The freestream Mach number is set at $M_\infty = 0.3$. The plots of pressure spectra are presented in log-linear format in order to provide a clearer visualization of the tonal peaks. One can see that multiple tones are excited for the current configuration. When blowing is applied, the main tonal peaks are amplified; however, blowing considerably reduces the amplitudes of the secondary tones. On the other hand, suction reduces the amplitudes of the main tones and increases those of the secondary ones.

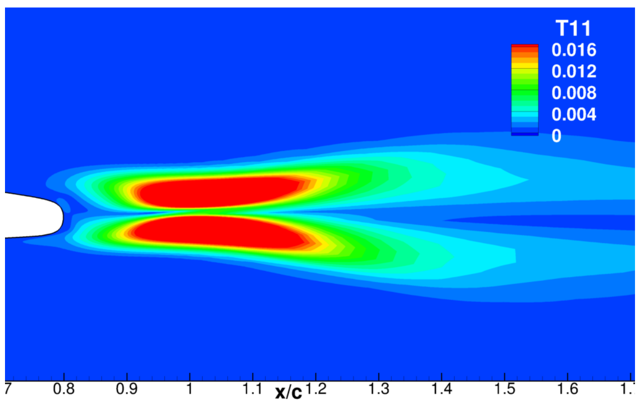
In Fig. 10, one can observe the results in terms of pressure spectra and flow fields for the numerical simulations of the NACA 0012 with surface 2 (thicker TE) for $Re_c = 100000$ and $AoA = 3$ deg. In the left and right columns, results are shown for freestream Mach numbers $M_\infty = 0.1$ and 0.3, respectively. Comparisons of spectra are presented when blowing is applied at the trailing edge region and for the case without blowing. Figures 10 (a) and (b) show pressure spectra in log-linear scales for $M_\infty = 0.1$ and 0.3, respectively. It is possible to see that multiple tones appear for the lower Mach number case while single tones are present for the higher Mach number flow. Steady blowing of $10\%M_\infty$ is applied along the trailing edge regions for both cases and the pressure spectra are modified in different ways. For $M_\infty = 0.1$, one can observe that the secondary tones disappear from the spectra and the peak value of the main tone is reduced (the amplitudes of the harmonics of the main peak are increased). The tonal peaks appearing in the spectrum of Fig. 10 (b), for $M_\infty = 0.3$, disappear when blowing is employed. However, small secondary peaks appear in the spectrum. For this latter case, the mechanism of noise reduction may be similar to that of the low Reynolds number case, when the amplitude of the quadrupole field is reduced and moved away from the airfoil surface due to blowing.



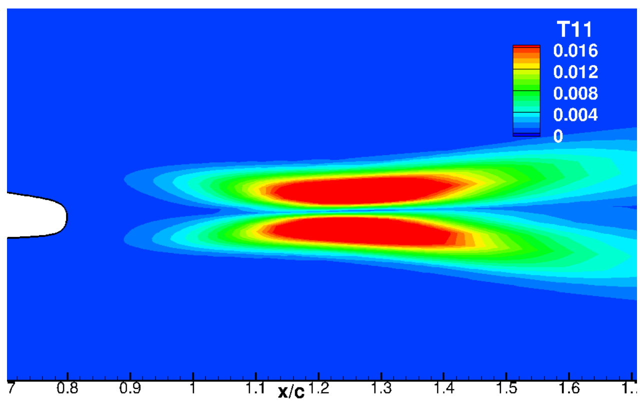
(a) Mean flow streamlines for no-blowing case



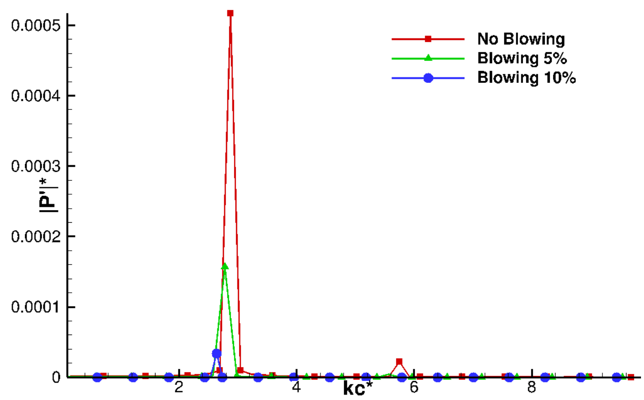
(b) Mean flow streamlines for blowing of $10\% M_\infty$ at TE



(c) Contours of magnitude of quadrupole sources, $\hat{T}_{11}$, at vortex shedding frequency for no-blowing case

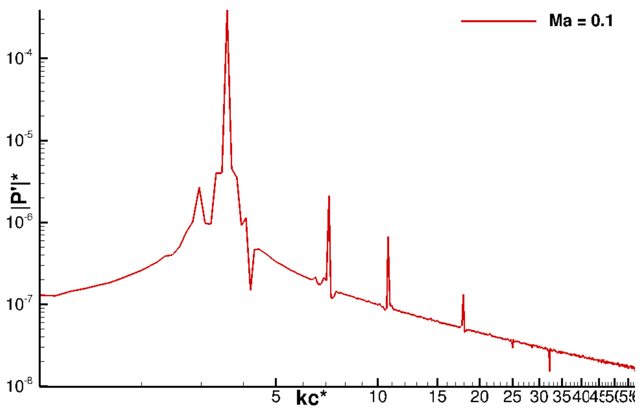


(d) Contours of magnitude of quadrupole sources, $\hat{T}_{11}$, at vortex shedding frequency for blowing of $10\% M_\infty$ case

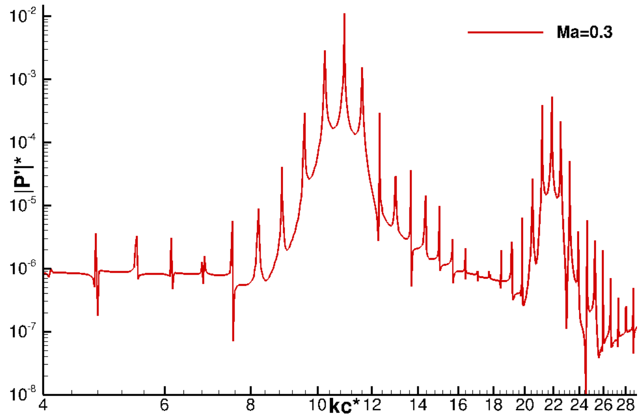


(e) Pressure spectra for blowing of $5\% M_\infty$ and $10\% M_\infty$ at TE

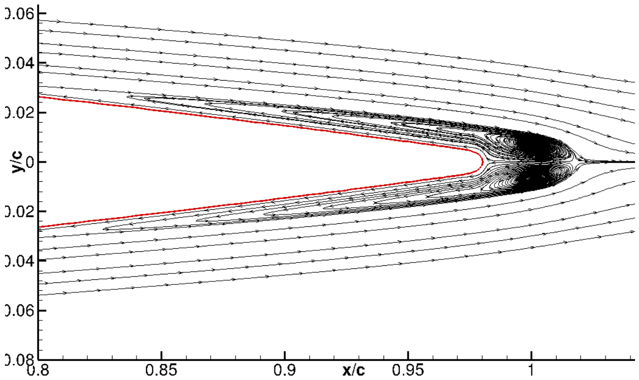
Figure 5. Effects of trailing edge blowing for surface 2 (thicker TE), $AoA = 3$ deg, $Re_c = 5000$ and $M_\infty = 0.3$.



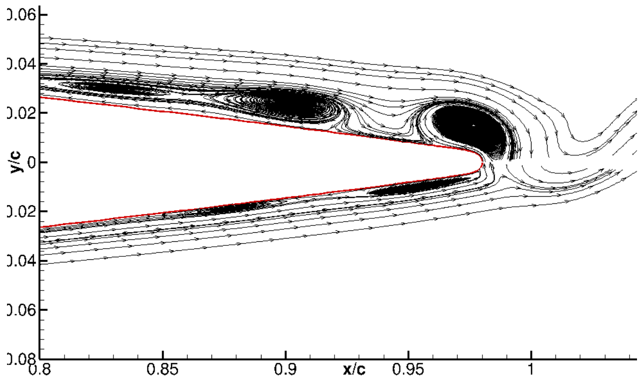
(a) Pressure spectrum for $M_\infty = 0.1$



(b) Pressure spectrum at $M_\infty = 0.3$

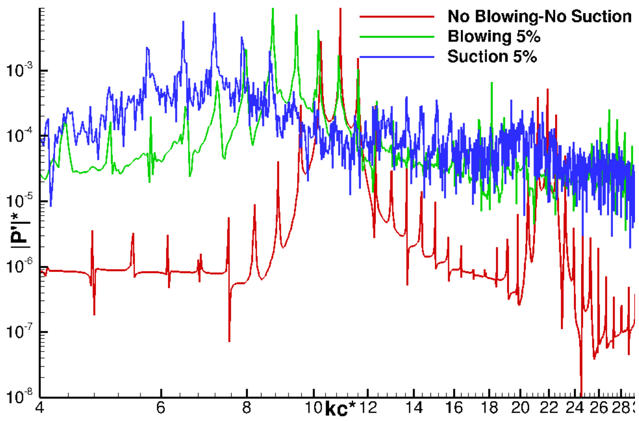


(c) Mean flow streamlines along TE for $M_\infty = 0.1$

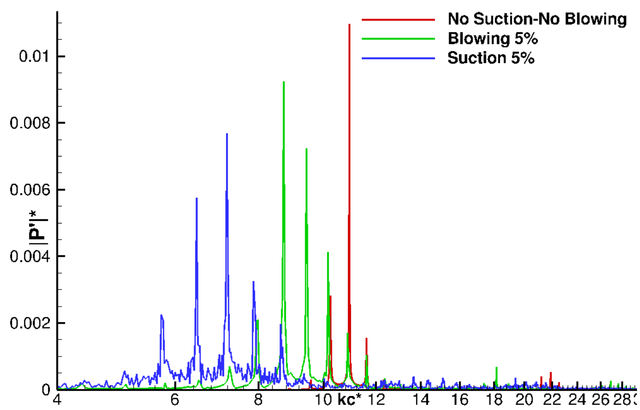


(d) Instantaneous streamlines along TE for $M_\infty = 0.3$

Figure 6. Results for surface 1 (thinner TE) at $AoA = 0$ deg, $Re_c = 100000$ for $M_\infty = 0.1$ $(left)$ and 0.3 $(right)$.

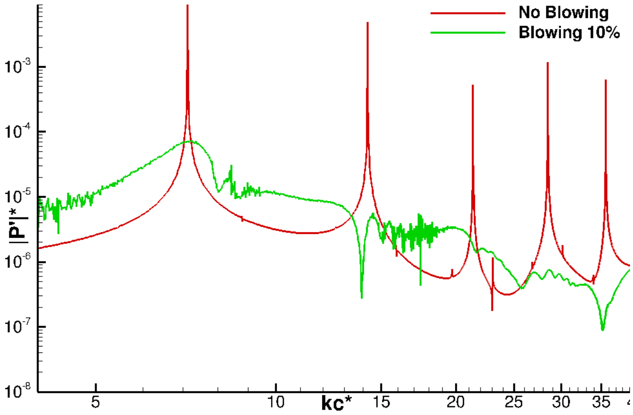


(a) Pressure spectra in log×log scale

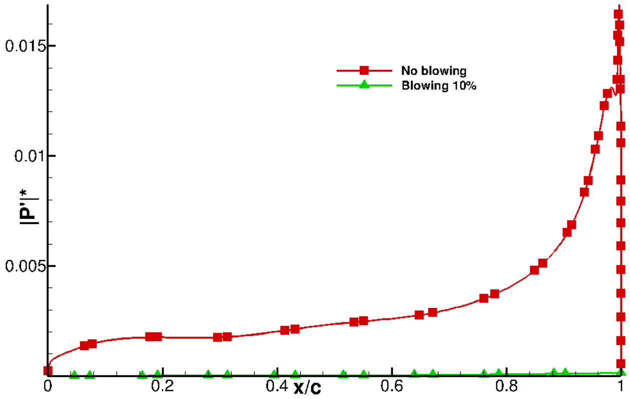


(b) Pressure spectra in log×linear scale

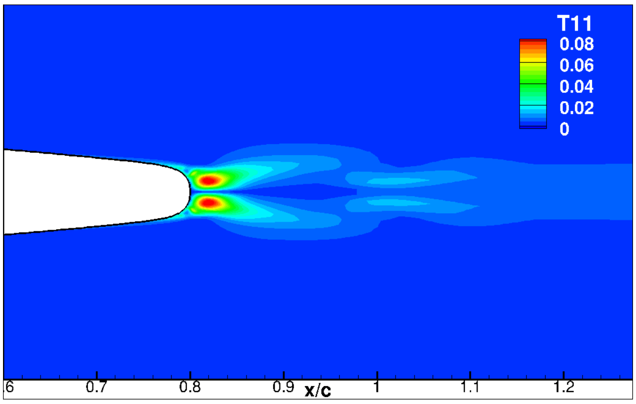
Figure 7. Results of blowing and suction of $5\% M_\infty$ $(left)$ for surface 1(thinner TE) at $AoA = 0$ deg, $Re_c = 100000$ and $M_\infty = 0.3$.



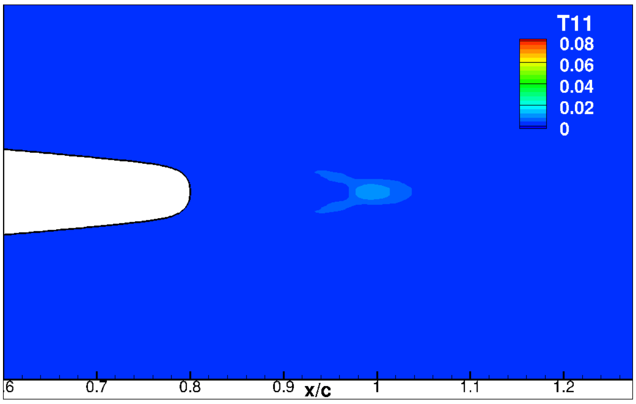
(a) Pressure spectra for blowing of $10\% M_\infty$ at TE



(b) Magnitude of pressure fluctuations, $|p'|$, along airfoil surface at vortex shedding frequency for $10\% M_\infty$ blowing at TE

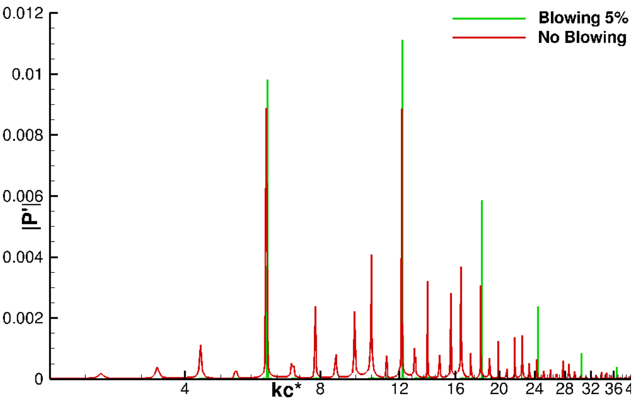


(c) Spatial distribution of quadrupole sources, $\hat{T}_{11}$, at vortex shedding frequency, without blowing

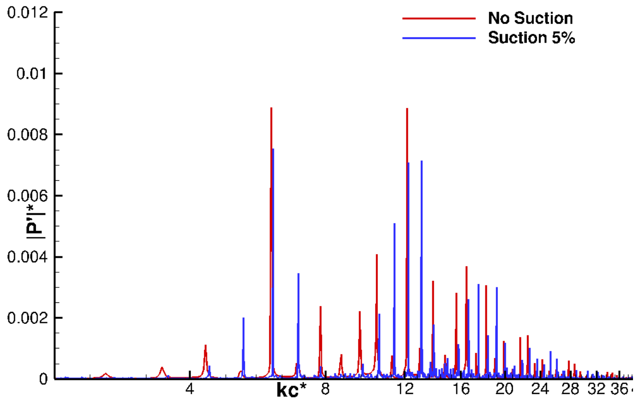


(d) Spatial distribution of quadrupole sources, $\hat{T}_{11}$, at vortex shedding frequency, for blowing of $10\% M_\infty$

Figure 8. Comparison of results for surface 2 (thicker TE) for $AoA = 0$ deg, $Re_c = 100000$ and $M_\infty = 0.3$.

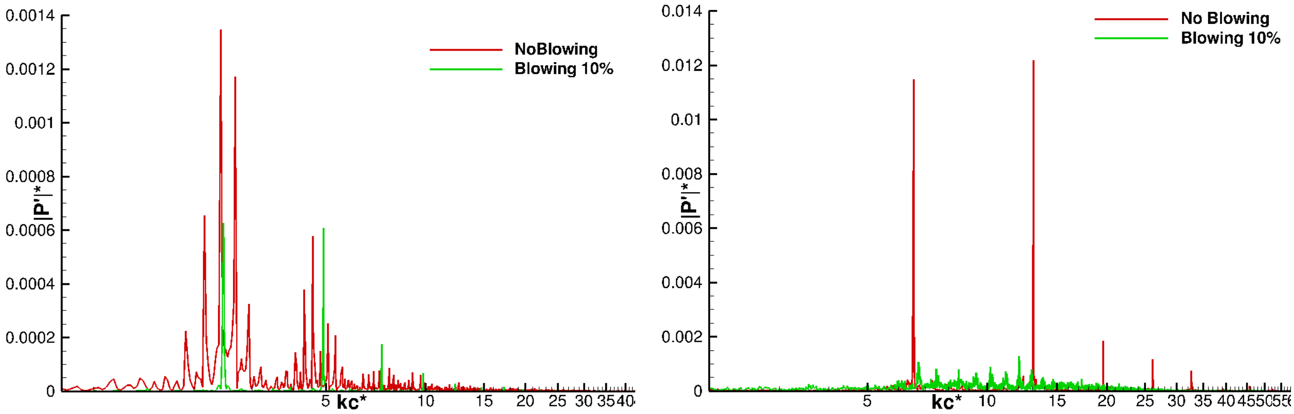


(a) Pressure spectra for $M_\infty = 0.3$ in log×linear scale for blowing of $5\% M_\infty$



(b) Pressure spectra for $M_\infty = 0.3$ in log×linear scale for suction of $5\% M_\infty$

Figure 9. Results for surface 1 (thinner TE) for $AoA = 3$ deg, $Re_c = 100000$ and $M_\infty = 0.3$.



(a) Pressure spectra for $M_\infty = 0.1$ in log×linear scale for blowing of $10\%M_\infty$

(b) Pressure spectra for $M_\infty = 0.3$ in log×linear scale for blowing of $10\%M_\infty$

Figure 10. Results of surface 2 (thicker TE) for $Re_c = 100000$, $AoA = 3$ deg.

4. Conclusions

Direct numerical simulations are conducted for compressible flows past a NACA 0012 airfoil with different blunt trailing edges. A study of tonal noise generation is performed for low and moderate Reynolds numbers including an assessment of compressibility effects as well as trailing edge suction and blowing.

Results for the moderate Reynolds number flow analyzed show that the airfoil may emit multiple "narrow-band" tones superimposed on a broadband hump, depending on the flow configuration. Compressibility effects play a major role in the tonal noise generation process when the airfoil with the thinner trailing edge is at zero angle of attack. For $M_\infty = 0.1$, the flow is symmetric and the presence of secondary tones due to an acoustic feedback loop is questionable. When the freestream Mach number is increased to $M_\infty = 0.3$, the flow becomes non-symmetric and secondary tones are clearly visible superimposed on a broadband hump. For $M_\infty = 0.1$, suction and blowing increase far field noise at the main frequency and secondary tones appear when blowing is applied. For $M_\infty = 0.3$, suction and blowing reduce the amplitudes of the tonal peaks which occur at lower frequencies. For the blunter trailing edge at $AoA = 0$ deg, the tonal noise mechanism is similar to that of the low Reynolds number flow and a single tone is excited along with its harmonics. Blowing reduces the acoustic scattering in a similar fashion as for the low Reynolds number case.

When the airfoil with a thinner trailing edge is at an angle of incidence, it exhibits a spectrum with secondary tones and the acoustic feedback loop is present. Compressibility effects do not play a major role here. For this case, the secondary tones disappear when blowing is applied but the main tones have their magnitudes amplified. Suction, on the other hand, reduced the magnitude of the main tones while increasing that of the secondary ones. Compressibility effects become important for the blunter trailing edge when the airfoil is at a non-zero angle of incidence. Differently from previous cases analyzed, here, secondary tones are observed for $M_\infty = 0.1$ and blowing eliminates these secondary tones and reduces the magnitude of the main tone. For $M_\infty = 0.3$, the spectrum resembles that of a low Reynolds number flow and blowing eliminates these tones.

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