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Effects of mean flow convection, quadrupole sources and vortex shedding on airfoil overall sound pressure level

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ABSTRACT

This paper presents a further analysis of results of airfoil self-noise prediction obtained in the previous work using large eddy simulation and acoustic analogy. The physical mechanisms responsible for airfoil noise generation in the aerodynamic flows analyzed are a combination of turbulent and laminar boundary layers, as well as vortex shedding (VS) originated due to trailing edge bluntness. The primary interest here consists of evaluating the effects of mean flow convection, quadrupole sources and vortex shedding tonal noise on the overall sound pressure level (OASPL) of a NACA0012 airfoil at low and moderate freestream Mach numbers. The overall sound pressure level is the measured quantity which eventually would be the main concern in terms of noise generation for aircraft and wind energy companies, and regulating agencies. The Reynolds number based on the airfoil chord is fixed at $Re_c = 408,000$ for all flow configurations studied. The results demonstrate that, for moderate Mach numbers, mean flow effects and quadrupole sources considerably increase OASPL and, therefore, should be taken into account in the acoustic prediction. For a low Mach number flow with vortex shedding, it is observed that OASPL is higher when laminar boundary layer separation is the VS driving mechanism compared to trailing edge bluntness.

1. Introduction

Understanding the physics associated with trailing edge noise generation and propagation is of paramount importance for the design of low-noise aerodynamic shapes such as wings and high-lift devices, as well as wind turbine blades, propellers and fans. The combined direct numerical simulation of both noise generation, and its subsequent propagation to the far field, is prohibitively expensive due to resolution requirements. Therefore, hybrid methods are typically employed, in which computational fluid dynamics (CFD) is used to calculate the near flowfield quantities responsible for the sound generation, which are in turn used as an input to a propagation formulation that calculates the far field sound signature. The flow physics associated with sound generation must be accurately captured in the CFD calculation in order to be used in this context.

In the previous work [1], the current authors used a high-fidelity numerical framework to investigate airfoil noise generation and propagation. Primarily, the compressible Navier–Stokes equations were solved using large eddy simulation

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Table 1
Summary of flow configurations analyzed. Re_c is the Reynolds number based on the airfoil chord, M_∞ is the freestream Mach number and AoA is the airfoil angle of attack. For all simulations, the trailing edge radius is given by $r/c \approx 0.0015$.

Configuration	Re_c	M_∞	AoA (deg)	Boundary layer tripping
1	408,000	0.115	0	Top and bottom sides
2	408,000	0.115	5	Suction side
3	408,000	0.115	5	Suction and pressure sides
4	408,000	0.4	5	Suction side

(LES) for accurate computation of noise sources from turbulent flows with a broad range of frequencies and spatial scales. Then, acoustic predictions were performed by the Ffowcs Williams and Hawkings (FWH) [2] acoustic analogy formulation. The numerical methodology was validated against experimental results from Sagrado and Hynes [3] and Brooks et al. [4] for different flow configurations. An assessment of mean flow convection and quadrupole source effects on acoustic pressure directivity was presented including an extensive investigation of vortex shedding (VS) effects on tonal noise.

In this paper, we will further analyze the effects of mean flow convection and quadrupole sources on the prediction of overall sound pressure level (OASPL) of a NACA0012 airfoil with rounded trailing edge. The overall sound pressure level is the measured quantity which eventually would be the main concern in terms of noise generation for aircraft and wind energy companies, and regulating agencies. Table 1 describes the flow configurations studied including the different freestream Mach numbers, angles of incidence and boundary layer tripping configurations analyzed. The Reynolds number based on the airfoil chord is fixed at $Re_c = 408,000$ for all flow configurations studied. The physical mechanisms responsible for airfoil noise generation in the aerodynamic flows analyzed are a combination of turbulent and laminar boundary layers, as well as vortex shedding (VS) originated due to trailing edge bluntness. Configuration 1 establishes a baseline case, at a sufficiently small freestream Mach number and with tripping on both sides of the airfoil, hence ensuring that the boundary layers from both sides are turbulent at the trailing edge. Effects of vortex shedding on OASPL are investigated for configurations 2 and 3 through tripping the boundary layers only along the suction side of the airfoil or along both sides simultaneously for otherwise identical flow configurations. When the tripping mechanism is not applied on the airfoil pressure side, a laminar boundary layer is developed due to the favorable pressure gradient. It was observed in Ref. [1] that laminar boundary layer separation is one of the physical mechanisms responsible for vortex shedding together with trailing edge bluntness. For all test cases, boundary layer tripping is always applied along the suction side of the airfoil and, therefore, a turbulent boundary layer is guaranteed to develop on this side. Finally, configuration 4 allows the investigation of the effects of moderate Mach numbers on OASPL.

2. Numerical methodology

The general curvilinear form of the compressible Navier–Stokes equations is solved using LES. The numerical scheme for spatial discretization is a sixth-order accurate compact scheme [5] implemented on a staggered grid. The current numerical capability allows the use of overset grids with a fourth-order accurate Hermitian interpolation between grid blocks [6]. The time integration of the fluid equations is carried out by a fully implicit second-order Beam–Warming scheme [7] in the near-wall region in order to overcome the time step restriction that would otherwise appear with the use of an explicit method. 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 except for the tripping region where suction and blowing are applied. In the present simulations, boundary layers are tripped by steady suction and blowing. The tripping methodology is applied in order to force transition to turbulent boundary layers and it is chosen to model the experimental tripping used by Brooks et al. [4] for similar flow configurations. The magnitude of suction and blowing is constant along the airfoil span with $|U_{\text{blowing}}| = |U_{\text{suction}}| = 0.03U_\infty$, chosen from numerical experimentation. Further details on the implementation of the suction and blowing conditions, including the specific positions of the suction and blowing regions for each test case, can be found in Ref. [1]. Characteristic plus sponge boundary conditions are applied in the far-field locations and periodic boundary conditions are applied in the spanwise direction. The dynamic subgrid model formulation of Lilly [8] is used to include the effects of unresolved turbulent scales. The numerical tool has been previously validated for several compressible flow simulations [1,6,9].

The FWH acoustic analogy formulation [2] is used for the aeroacoustic predictions. The frequency domain formulation presented by Lockard [10] is implemented and incorporates convective effects. Surface dipole integrations are computed along the airfoil surface and volume quadrupole integrations are computed along a subset region of the flowfield including the wake plus boundary layer regions. A Hanning filter is applied in an energy preserving manner [10] to dipole and quadrupole source terms before they are transformed to the frequency domain. Aeroacoustic integrals are computed using a 3-D wideband multi-level adaptive fast multipole method (FMM) [11,12] in order to accelerate the calculations of the FWH equation. The developed numerical capability allows the analysis of each noise source individually. Therefore, it is possible to investigate the separate effects of dipole and quadrupole sources as well as the effects of mean flow convection on the calculation of these sources. With the method applied in this work, the computational cost of the aeroacoustic integrals is considerably reduced.

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