DIRECT NOISE COMPUTATION OF LOW AND MODERATE MACH NUMBER FLOWS WITH WAKE INTERACTION

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Abstract. *Direct noise calculations of a model airframe noise problem are conducted in order to assess the effects of wake and boundary layer interaction on aeroacoustics. Simulations of unsteady laminar flows including both noise generation, and its subsequent propagation to the far field, are performed for a two-dimensional configuration composed of a cylinder placed above a NACA 0012 airfoil at 5 deg. angle of incidence. The Reynolds number based on the airfoil chord is set at* $Re_c = 5000$ *and the Reynolds number based on the diameter of the cylinder is* $Re_d = 200$*. An investigation of freestream Mach number effects on sound radiation is presented for* $M_\infty = 0.1$*,* 0.3 *and* 0.5*.*

Keywords: *Direct noise computation, High-order schemes, Wake interaction*

1. INTRODUCTION

In practical problems of interest, especially the prediction of low and moderate Mach number airframe noise, the large disparity in energy scales between the hydrodynamic and acoustic fields makes it impractical to directly compute far field noise. Hybrid methods have been widely utilized as a solution to this problem where the source field is computed separately from the acoustic field using an acoustic analogy. Typically, the analogy is derived from Lighthill's work (Lighthill, 1952), with the Ffowcs-Williams and Hawkings (1969) formulation commonly used in airframe noise problems. The analogy includes contributions from sources integrated along the geometry surface (monopoles and dipoles) as well as volumetric sources along boundary layer and wake regions (quadrupoles). Due to the cost of computing the volume integrals, quadrupole terms are often neglected. In low Mach number flows, this approximation is reasonable since the effects of quadrupoles are small relative to the effects of dipoles and monopoles. However, recent studies have shown that, even at moderate Mach numbers, quadrupole sources can have a non-negligible effect on far field acoustic predictions (Wolf and Lele, 2010, 2012). Additionally, it has been demonstrated that quadrupole sources have a significant impact on predictions involving airframe configurations, particularly those with wake interactions.

With growing interest in noise predictions of complex airframe configurations, the issue of quadrupole noise is of paramount importance. At flight Reynolds numbers, turbulent wakes form and interact downstream of the geometries studied. In these cases, it is unclear when quadrupoles may play an important role and when they may be neglected. Casper *et al.* (2004) found that the inclusion of quadrupole terms was necessary for accurate predictions of a multi-element airfoil at $M_\infty = 0.2$. Additionally, Spalart *et al.* (2010) found evidence to suggest that quadrupole noise for a simple landing gear array may still be significant at $M_\infty = 0.115$.

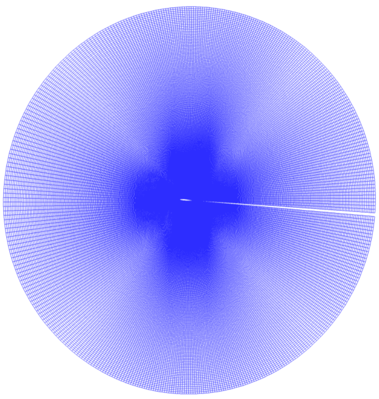
In the present work, direct noise calculations of a model airframe noise problem are conducted in order to assess the effects of wake and boundary layer interaction on aeroacoustics. Simulations of unsteady laminar flows including both noise generation, and its subsequent propagation to the far field, are performed for a two-dimensional configuration composed of a cylinder placed above a NACA 0012 airfoil at 5 deg. angle of incidence. The Reynolds number based on the airfoil chord is set at $Re_c = 5000$ and the Reynolds number based on the diameter of the cylinder is $Re_d = 200$. An investigation of freestream Mach number effects on sound radiation is presented for $M_\infty = 0.1$, 0.3 and 0.5. The results presented in this paper will be used to validate those obtained by a hybrid methodology, in future work, where an assessment of quadrupole source effects will be presented.

2. NUMERICAL FORMULATION

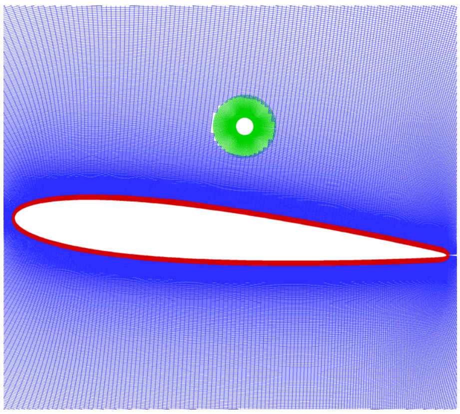
The general curvilinear form of the compressible Navier Stokes equations is solved in conservation form. The numerical scheme for spatial discretization is a sixth-order accurate compact scheme (Nagarajan *et al.*, 2003) implemented on a staggered grid. Compact finite-difference schemes are non-dissipative and numerical instabilities arising from insufficient grid resolution, mesh non-uniformities, approximate boundary conditions and interpolation at grid interfaces have to be filtered to preserve stability of the numerical schemes. The high wavenumber compact filter presented by 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 current numerical capability allows the use of overset grids with a fourth-order accurate Hermite interpolation between grid blocks (Bhaskaran and Lele, 2010). The time integration of the fluid equations is carried out by the fully implicit second-order scheme of Beam and Warming (1978) in the near-wall region in order to overcome the time step restriction. 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 farfield locations. The numerical tool has been previously validated for several simulations of compressible flows involving sound generation and propagation (Wolf and Lele, 2010, 2012; Yu *et al.*, 2010).

3. RESULTS

This section discusses results obtained by the direct calculation of noise generated by the unsteady flow past a cylinder in the proximity of a NACA0012 airfoil. The flow configurations analyzed allow a study of sound generation due to interaction of boundary layers and wakes, including vortex shedding. The flow Reynolds number based on the airfoil chord and cylinder diameter are set $Re_c = 5000$ and $Re_d = 200$, respectively. The freestream Mach numbers analyzed are $M_\infty = 0.1$, 0.3 and 0.5 and the flow angle of incidence is $AoA = 5$ deg. The present grid configuration consists of body-fitted O-grid blocks around airfoil and cylinder surfaces and a background O-grid block that resolves the acoustic far field. The airfoil grid block is composed of 400×60 grid points, in the periodic and wall normal directions, respectively, and the cylinder grid block is composed of 240×50 grid points, in the periodic and wall normal directions. These grid blocks are designed to accurately resolve the laminar boundary layers that develop along the surfaces. The far field background grid block has 400×625 points, in the periodic and wall normal directions, respectively, with small stretching to accurately capture the propagated sound waves. In Figs. 1 (a) and (b) one can see the full view of the computational grid with approximately 300000 grid points and an enlarged view of the airfoil and cylinder mesh regions, respectively.



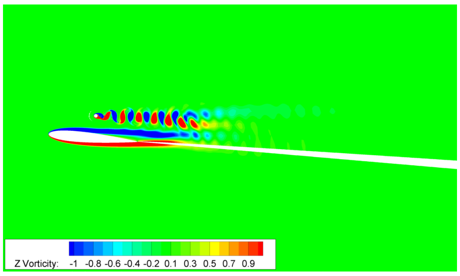
(a) Far field background grid block.



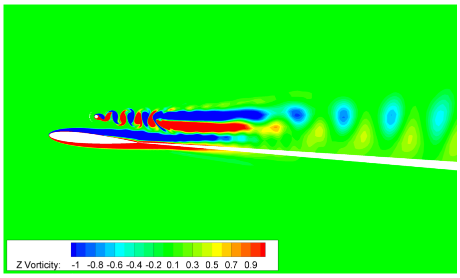
(b) Enlarged view of the grids over airfoil and cylinder.

Figure 1. Details of the computational grid used for the flow configurations analyzed.

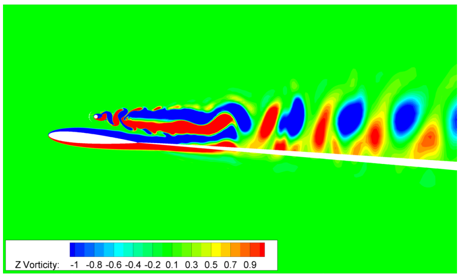
Figures 2 (a), (b) and (c) present contours of z-component of vorticity along airfoil and cylinder wake regions for $M_\infty = 0.1$, 0.3 and 0.5, respectively. One can observe that vortex shedding forms behind the cylinder for all cases analyzed. For the lower Mach number case studied, the cylinder vortex shedding is convected downstream and the airfoil wake does not interact with shed structures. However, for the higher Mach number flows investigated, strong wake interaction is observed. These effects are more pronounced for the $M_\infty = 0.5$ flow configuration.



(a) $M_\infty = 0.1$.



(b) $M_\infty = 0.3$.



(c) $M_\infty = 0.5$.

Figure 2. Contours of z-component of vorticity along airfoil and cylinder wake regions.

In Figs. 3 (a), (b) and (c), one can observe contours of dilatation for $M_\infty = 0.1$, 0.3 and 0.5, respectively. It is expected that dipole sources dominate sound radiation for the $M_\infty = 0.1$ case. Therefore, pressure fluctuations along cylinder and airfoil surfaces should drive far field sound radiation. For this low Mach number case, acoustic waves propagate at low frequencies and diffraction effects along the airfoil trailing edge can occur. For the higher Mach number cases, the effects of quadrupole sources should considerably contribute to noise radiation. Furthermore, for these cases, acoustic waves propagate at high frequencies and scattering effects of quadrupole sources will occur along the cylinder and airfoil surfaces. It is possible to see in Fig. 3 that levels of dilatation are stronger for the higher Mach number flow configurations since more energy is propagated as sound in these cases. Moreover, one can clearly notice the Doppler effects on upstream and downstream sound propagation for these cases.

4. CONCLUSIONS

This work presents an investigation of wake and boundary layer interaction effects on the aeroacoustics of a model airframe noise problem. Direct noise calculations are performed for a two-dimensional configuration composed of a cylinder placed above a NACA 0012 airfoil at 5 deg. angle of incidence. For the current flow configurations, laminar boundary layers develop along cylinder and airfoil. The Navier Stokes equations are solved using a high-order compact finite difference scheme with high-resolution properties. Time advancement of the equations is performed using a hybrid implicit-explicit methodology and the current numerical capability allows the use of overset grids. An assessment of freestream Mach number effects on sound radiation is presented for unsteady aerodynamic problems with wake interaction.

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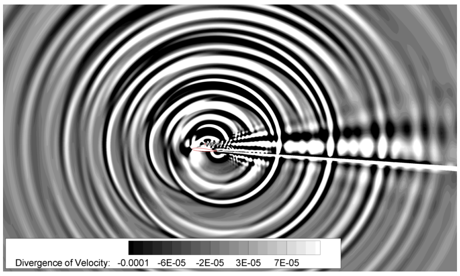
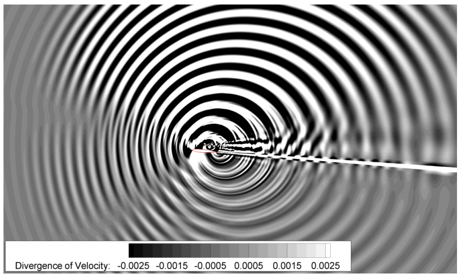
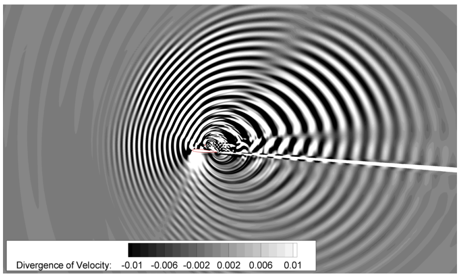


(a) $M_\infty = 0.1$.



(b) $M_\infty = 0.3$.



(c) $M_\infty = 0.5$.

Figure 3. Contours of dilatation.

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6. RESPONSIBILITY NOTICE

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