Noise Simulation around NACA0012 Wingtip using Large Eddy Simulation

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Flow structures around a NACA0012 wingtip are investigated using a zonal LES/RANS hybrid method in order to understand the noise generation mechanism around the flap-edge. It is known from previous studies that the flow around a blunt wingtip is similar to that of the flap-edge. Grid dependency studies are performed for both time-averaged and unsteady components, and the results are assessed via comparison with the experimental data. It became apparent that the zonal LES/RANS hybrid results are more sensitive to the chordwise grid resolution, compared with these of RANS. Based on the validated data, the noise generation mechanism around the wingtip is discussed. The near-field flow structures relevant to noise generation are obtained successfully.

Key Words: Aerodynamics, Aeroacoustics, Computational Fluid Dynamics, High Lift Devices

Nomenclature

c : chord length
 C_p : pressure coefficient
 C_s : Smagorinsky constant
 u, v, w : velocity
 x, y, z : position
 α : filter constant
 Δt : time step
Subscripts
 ∞ : free stream value

1. Introduction

Over the past few decades, aircraft noise has become a major problem due to the increase in air travel. FAA regulations on noise in the vicinity of airports are becoming more stringent than ever, and it is clear that noise reduction technology is critical to future commercial aircraft development. Since engine noise has been declining during the last few decades, the airframe noise can no longer be ignored for the purpose of further noise reduction. This is especially true for the case of flow around high lift devices (HLD) during the landing phase, which generates noise at a level comparative to that produced by engines, because in this condition the engines are driven at low power.¹⁾ From previous studies, it is well known that the sound generated from the flap-edge is one of the causes of the airframe noise.²⁾ However, it seems to be difficult to simulate the aerodynamic fluctuations that lead to the noise generation around the flap. Most of the computations done in the past were based on Reynolds-Averaged Navier-Stokes (RANS) equations,^{2–7)} which basically assume a steady-state flow. Recently, Cummings et al.⁸⁾ calculated a flow around a three-dimensional three-element HLD configuration using detached

eddy simulation (DES), and qualitative comparisons were made for the power spectrum density of wing lift. However, it seems to be difficult to reproduce the aerodynamic fluctuations that lead to the noise generation around the flap.

Thus, the goal is to get further information related to the noise sources around the flap of HLD by performing accurate unsteady flow simulations, and estimating the noise sources around the flap-edge. However, unsteady computations around the flap of HLD are not feasible yet. Therefore, careful consideration should be made before approaching this objective. McNerny et al.⁹⁾ performed wind tunnel experiments with a blunt-tipped rectangular wing with NACA0012 airfoil section, and showed that the flow structure can be correlated closely with that of the flap-edge. In addition, in our previous study,¹⁰⁾ similarity of the steady-state flow field around a NACA0012 wingtip and that of the flap-edge was assessed using the RANS method. Based on these studies, this simple configuration is expected to have a similar noise generating mechanism as that of the flap-edge.

Based on previous discussions, a zonal LES/RANS hybrid method is applied to solve the unsteady vortical flow structures around the wingtip. Unsteady computations are performed varying the grid resolutions. The results are compared with the wind tunnel experiment performed at Japan Aerospace Exploration Agency (JAXA). Using the validated numerical data, detailed observation of the flow structure around the wingtip is performed. Finally, far-field sound pressure level (SPL) estimation based on Curle's equation is presented using the unsteady surface data as an input.

This paper is organized as follows. Section 2 briefly describes the computational model, numerical methods and setup of the simulations. Section 3 presents the computational results. The computed steady and unsteady components are presented. In particular, the unsteady components

Fig. 1. NACA0012 wingtip mounted inside the wind tunnel.

are compared with the experiment while varying the grid resolution. Based on the validated data, the flow field around the wingtip region is observed in detail. Finally, section 4 concludes this paper.

2. Numerical Setup

2.1. Computational model and flow conditions

The computational model is based on the experiment performed at the low-speed wind tunnel at JAXA (JAXA-LTW2),^{11,12)} which is a closed-circuit wind tunnel with a 2×2 m cross-section. The model is a rectangular wing with a NACA0012 profile as shown in Fig. 1. The shape of the wingtip is blunt with sharp edges. The model has a chord length of 0.4 m and span of 1.0 m. The test was carried out to provide the experimental data for CFD validation and to determine flow physics around the near-field of the wingtip.

The flow conditions for the computations were selected as follows. The mean flow velocity is 60 m/s, which corresponds to a Mach number of 0.175. This flow velocity is similar to that of the landing speed of the aircraft. In order to mimic the flow field of the flap-edge, a high angle-of-attack is desirable. In the experiment, however, flow separation was observed near the trailing edge over an angle-of-attack of 12 degrees. In order to maximize the noise generation from the wingtip, an angle-of-attack of 12 degrees is chosen. The Reynolds number, based on the chord length and mean flow velocity, is 1.8×10^6 .

2.2. Computational grids

A block-structured one-to-one point-matched grid was generated with the commercial software Gridgen and the grid topology is shown in Fig. 2. As shown in Fig. 2, x -, y - and z -directions correspond to the chordwise, spanwise and vertical directions, respectively.

Two grids are used in this study and the details are summarized in Table 1. The first grid consists of nearly six million grid points and is called the 6M grid hereafter. From previous studies, it is known that vortices of the x (chordwise) component around the wingtip region play important roles. Thus, a fine and uniform grid on the chordwise

Fig. 2. Grid topology and boundary conditions.

Table 1. Grid details.

Grid	6M	20M
Total points	6 million	20 million
No. of blocks	111	201
Grid points along camber	190	408
Grid points at maximum wing thickness	97	101
Grid points at TE	97	30

Front view Side view
(a) Spatial grid distribution around the wingtip
(every three points are drawn)

Trailing edge Max. thickness Leading edge
(b) Surface grid distribution on the wingtip
(all points are drawn)

Fig. 3. Coarse grid (6M).

cross-section is generated as shown on the left side of Fig. 3(a). The fine and uniform grid distribution is achieved inside the region where the distance from the wall is smaller than $0.1c$. On the other hand, the grid distribution along the chordwise direction is coarser than that of the cross-sectional

Fig. 4. Fine grid (20M).

plane as shown in Fig. 3(b). The second grid consists of approximately 20 million grid points and is called the 20M grid hereafter. This grid was designed to overcome the defect of the 6M grid. The grid resolution on the chordwise cross-sectional plane is almost the same as the 6M grid (See Fig. 4(a)). The fine grid region is extended to $0.15c$ for the 20M grid in the spanwise direction and also in the vertical direction at the trailing edge. The grid size in the chordwise direction is nearly four times smaller than the 6M grid around the mid-chord location. Consequently, the grid sizes in all directions around the wingtip are almost uniform. This allows the evolution of the complicated vortical flow structures. Around the trailing edge of the wingtip region, unnecessary grid concentration is avoided compared with the 6M grid (see Fig. 4(b)). Although, grid size in the chordwise direction has decreased to nearly $1/4$ and the fine grid region has been extended, the overall increase in grid points is suppressed to approximately three times that of the 6M grid.

2.3. CFD solver

Numerical methods used in the calculations are briefly described. The large eddy simulation (LES) code, which is called UPACS-LES, is used in this research. This code is developed based on the UPACS code,^{13–16)} which is a standard CFD code used at JAXA. The flow solver of the current version is based on a cell-centered finite-volume method on multi-block structured grids. The code is parallelized by a flexible domain decomposition concept and MPI. The governing equations are the compressible Navier-Stokes equations. There are several options for the choice of turbulence modeling; LES mode, RANS mode or LES/RANS hybrid mode.

Table 2. Computational cases.

	Turbulence model	Grid	Scheme
Case 1	RANS (S-A)	6M	Roe 3rd order
Case 2	RANS (S-A)	20M	Roe 3rd order
Case 3	Zonal LES/RANS	6M	Compact 6th order
Case 4	Zonal LES/RANS	20M	Compact 6th order

A standard Smagorinsky model with $C_s = 0.1$ is used for the LES computation¹⁷⁾ and the Spalart-Allmaras (S-A) one-equation model¹⁸⁾ is used for the RANS computation. In this study, only the unsteadiness around the wingtip is of interest, as this is where wingtip noise is generated. In order to reduce the computational cost, the wingtip region is computed via LES and the rest of the region is calculated via S-A one-equation model. The switching between LES and RANS regions is controlled by the switching function. In this study, the boundaries of LES and RANS are defined by planes at $y/c = -0.1$ and 0.15 for 6M and 20M grids, respectively.

The simulations are performed using characteristic boundary conditions along the far-field boundaries in the chordwise cross-sectional plane, except for extrapolation from the interior at the downstream boundary. The slip-wall boundary condition is applied to the spanwise cross sectional planes and no-slip boundary conditions are imposed at the wing surface (See Fig. 2).

Zonal LES/RANS simulations are performed in the following procedures. First, RANS (S-A) computation is performed in order to obtain a steady-state flow-field as an initial condition for the unsteady computations. Second, zonal LES/RANS computations are performed in order to obtain unsteady flow. The simulations are run over a long duration that consists of two wing flow-through time units in order to minimize the transient effects. No acceleration was added in order to enforce the unsteadiness of the flow. Finally, unsteady computations are performed in order to obtain the flow statistics. In order to reduce the convergence error in the flow statistics, the simulations are run over two wing flow-through time units.

The computed results are assessed by changing the turbulence modeling and grid resolution. The details of the computations are summarized in Table 2. Cases 1 and 2 are the steady-state RANS computations using the S-A model on 6M and 20M grids, respectively. The convection terms are discretized using the Roe scheme with 3rd-order MUSCL and no-limiter imposed. First-order time integration is performed via an implicit method using the Matrix-Free Gauss-Seidel (MFGS) scheme¹⁹⁾ with the local time stepping technique. The Courant-Friedrichs-Levy (CFL) number is set to 100 and computation is performed until convergence is achieved. Cases 3 (6M grid) and 4 (20M grid) are zonal LES/RANS hybrid computations. High-order schemes are used in the simulations in order to obtain the best resolution from the given grid. The convection terms are discretized using the sixth-order compact scheme developed by Kobayashi.²⁰⁾ The viscous terms are discretized

using the second-order central scheme. In order to avoid numerical instability, the sixth-order filter developed by Gai-tonde-Visbal²¹⁾ is used with a filter constant of $\alpha = 0.45$. The second-order time integration is performed using MFGS with three sub-iterations. The non-dimensionalized time steps using the chord length of the wing and sonic speed as reference values are set to $\Delta t = 2 \times 10^{-4}$ and 10^{-4} for 6M and 20M grids, respectively. These values are chosen in order to satisfy the CFL condition ($CFL < 1$) at most parts of the wingtip region, excluding the boundary layer. This constraint must be satisfied in order to calculate the acoustic waves and vortices accurately inside the LES region.

3. Computational Results

First of all, the numerical data are validated by detailed comparison with the experimental results. Both the time-averaged and unsteady components are assessed. In particular, the effects of the grid resolution on the power spectral density of pressure coefficient (C_p) are examined. Finally, using the validated data, detailed flow structures around the wingtip are examined.

3.1. Validation of the computed results (time-averaged flow field)

From previous studies, it is well known that the flow field around the wingtip is associated with the dual vortex structure. Figure 5 shows the time-averaged vorticity magnitude iso-surface with cross-sectional distribution of the stream-wise vorticity around the wingtip. The vortex over the upper surface and that on the wingtip are termed as primary and secondary vortices, respectively, in the established terminology.⁹⁾ However, these two vortices are rotating in the same direction. Thus, it is more appropriate to understand these two vortices as two independent primary vortices. In this study, these vortices are termed as upper and side vortices, respectively. These two primary vortices are generated from the lower and upper edges of the wingtip and merge into a single tip vortex as they move downstream.

The simulated flow field around the wingtip is compared with the experimental results. Figure 6 shows the chordwise

C_p distribution at two different locations. The C_p distribution along the camber line of the wingtip is shown in Fig. 6(a). The RANS results are in good agreement with the experiment. A local minimum is observed around $x/c = 0.5$, where the side vortex crosses the camber line. The C_p distribution along the wing cross-section at $y/c = -0.0225$ (9 mm from the wingtip) is shown in Fig. 6(b). In this figure, two local minima can be seen after the suction peak near the leading edge. The peak at $x/c = 0.3$ is caused by the upper vortex. Together with the movements toward the downstream, this vortex moves away from the upper surface (see Fig. 5), and a local maximum appears around $x/c = 0.6$. Going further downstream, a peak appears around $x/c = 0.8$, where the upper vortex comes closer to the upper surface due to the influence of the side vortex (see Fig. 5). These tendencies are the same for all of the results.

Figure 7 shows the streamwise velocity component distribution obtained via LES/RANS, as well as that of the particle image velocimetry (PIV) measurement data^{11,12)} at $x/c = 0.5$ chordwise cross-sectional plane. The velocity is normalized by the mean flow velocity and the same scale is used for all of the plotting. Since the LES/RANS results

Fig. 5. Time-averaged flow-field iso-surface of vorticity magnitude with cross-sectional distribution of streamwise vorticity (Case 1).

Fig. 6. C_p distribution in chordwise direction.

Fig. 7. Streamwise velocity component on $x/c = 0.5$ plane.

Fig. 8. Streamwise velocity profile.

Fig. 9. Comparison of power spectral density on the upper surface of the wingtip.

are unsteady, time-averaged data are used. Two primary vortices are observed near the wingtip, one over the upper surface and another adjacent to the flat wingtip. Axial velocity within the vortex core can be greater (jet type) or less (wake type) than the freestream velocity.⁹⁾ All of the results show that the upper and side vortices are of jet and wake types, respectively.

In order to perform qualitative comparison with the experimental data, streamwise velocity profiles along line A and B are shown in Fig. 8. The locations of the lines are given in Fig. 7(a), where line A passes through the upper vortex horizontally and line B passes through the side vortex vertically. For RANS results, both lines collapse into the same line, and are consistent with the experimental data. Grid

Fig. 10. Instantaneous iso-surface of vorticity magnitude colored by vorticity in the vertical direction.

Fig. 11. TKE on three cross-sectional planes (LES/RANS 20M result).

convergence is already obtained for the 6M grid. Along line A, local maxima are observed around $y/c = -0.03$. For LES/RANS 6M results, the difference between the experiment and RANS is observed around the local maxima. The development of two vortices is affected by the boundary layer close to these vortices. Current LES/RANS simulations try to solve the boundary layer with LES mode, but more grid resolution is required to fully resolve the profile. At least, we have checked the grid dependency, and by increasing the grid points in the streamwise direction, improvement in prediction accuracy can be clearly observed. The plot obtained using LES/RANS 20M is much closer to the experimental results compared with the 6M result. Along line B, two local minima are observed. The minima at $z/c = -0.05$ and -0.08 are caused by the upper and side vortices at the side of the wingtip. In this case, the secondary vortex is rotating in the opposite direction of the primary vortex. Once again, the LES/RANS 6M result under-pre-

dicts the level of the local minima, and improvement can be observed by increasing the grid resolution.

3.2. Validation of the computed results (unsteady flow-field)

Figure 9 compares the power spectral density (PSD) of C_p on the upper surface ($y/c = -0.0125$ which is 5 mm from the wingtip) at three locations. There is a substantial difference between the experimental results and the 20M cases. However, the results on two different grid resolutions provide information on the variation of the computational results. At $x/c = 0.05$, the PSD level of the experiment is lower than other locations ($x/c = 0.2$ and $x/c = 0.65$). This tendency is computed by both LES/RANS 6M and 20M. At $x/c = 0.2$, the PSD pattern varies between the grids. In the experiment, a broad peak is observed at 5 kHz. This peak is reproduced by the results from LES/RANS 20M while the estimated peak level is slightly higher than the experiment. Together with progress further downstream to $x/c = 0.65$,

the level of the low-frequency broadband component increases. The results of the 6M grid fail to predict the level of PSD around the frequency of 1 to 5 kHz, but it is captured by the 20M grid. By increasing the grid points, better agreement with the experimental results is obtained.

Figure 10 shows the instantaneous iso-surface of vorticity magnitude. The results of LES/RANS 6M and 20M are shown in Fig. 10(a) and (b), respectively. In Fig. 10(a), vortical structure in the x -direction is dominant, and the rest of the components are dissipated. This is because the grid size in the chordwise direction is coarser than that of the chordwise cross-sectional plane. On the other hand, complicated flow structure around the wingtip is visualized in Fig. 10(b). By increasing the grid points in the chordwise direction, a significant difference in the vortical flow field can be observed. Although the dual vortex system at the wingtip region is oriented to the chordwise direction, a high-resolution grid is required to allow the vortices to evolve in all directions.

3.3. Detailed observation of the flow-field

Detailed observation of the flow field is performed using the LES/RANS 20M data. Figure 11 shows the turbulent kinetic energy (TKE) distribution derived at the $x/c = 0.5$, 0.7 and 0.95 cross-sectional planes. High TKE regions are observed at the core of the upper and side vortices as shown in Fig. 11(a) and (b). Near the trailing edge, where the merging of the dual vortex is finished, a high TKE region is distributed over the upper surface of the wing (See Fig. 11(c)).

Figures 12 and 13 show the surface and spatial pressure fluctuation around the wingtip. The surface pressure fluctuation is significant, especially on the surface adjacent to the side vortex. In Fig. 13(a), the propagation of acoustic waves is clearly visualized. The frequency of this acoustic wave is

estimated from the wavelength and it is approximately 10 kHz. The noise generation mechanism of this frequency is still unclear and is left for future analysis. From Fig. 13(b) and (c), concentric rings of acoustic waves cannot be observed as in Fig. 13(a). This indicates that the noise source of this frequency is distributed along the chordwise direction of the wingtip.

4. Conclusions

Flow structures around a NACA0012 wingtip were investigated using a zonal LES/RANS hybrid method. The computational results were validated by comparing them with the experimental data, and grid dependency studies were performed. By using a relatively uniform grid around the wingtip region, the prediction accuracy of both the time-averaged and unsteady components was increased.

Fig. 12. Instantaneous surface pressure fluctuation distribution.

(a) $x/c = 0.5$
Front view

(b) $y/c = 0.0$
Top view

(c) $z/c = 0.0$
Side view

Fig. 13. Instantaneous pressure fluctuation distribution around the wingtip.

Using the validated data, flow physics around the wingtip were examined. The result of the fine grid showed that the flow is much more complicated than was reported in previous studies^{2-7,9)} The dual vortex structure in the streamwise direction was well known from steady flow calculations, but by performing unsteady flow simulation, small vortices pointing in the omni-direction along the shear layer were clearly visualized.

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