Computations on Wings with Full-Span Oscillating Control Surfaces using Navier-Stokes Equations

Guru P. Guruswamy
Ames Research Center, Moffett Field, California

May 2013
Since its founding, NASA has been dedicated to the advancement of aeronautics and space science. The NASA scientific and technical information (STI) program plays a key part in helping NASA maintain this important role.

The NASA STI program operates under the auspices of the Agency Chief Information Officer. It collects, organizes, provides for archiving, and disseminates NASA’s STI. The NASA STI program provides access to the NASA Aeronautics and Space Database and its public interface, the NASA Technical Reports Server, thus providing one of the largest collections of aeronautical and space science STI in the world. Results are published in both non-NASA channels and by NASA in the NASA STI Report Series, which includes the following report types:

- **TECHNICAL PUBLICATION.** Reports of completed research or a major significant phase of research that present the results of NASA Programs and include extensive data or theoretical analysis. Includes compilations of significant scientific and technical data and information deemed to be of continuing reference value. NASA counterpart of peer-reviewed formal professional papers but has less stringent limitations on manuscript length and extent of graphic presentations.

- **CONFERENCE PUBLICATION.** Collected papers from scientific and technical conferences, symposia, seminars, or other meetings sponsored or co-sponsored by NASA.

- **SPECIAL PUBLICATION.** Scientific, technical, or historical information from NASA programs, projects, and missions, often concerned with subjects having substantial public interest.

- **TECHNICAL TRANSLATION.** English-language translations of foreign scientific and technical material pertinent to NASA’s mission.

Specialized services also include organizing and publishing research results, distributing specialized research announcements and feeds, providing information desk and personal search support, and enabling data exchange services.

For more information about the NASA STI program, see the following:

- Access the NASA STI program home page at [http://www.sti.nasa.gov](http://www.sti.nasa.gov)
- E-mail your question to [help@sti.nasa.gov](mailto:help@sti.nasa.gov)
- Fax your question to the NASA STI Information Desk at 443-757-5803
- Phone the NASA STI Information Desk at 443-757-5802
- Write to:
  STI Information Desk
  NASA Center for AeroSpace Information
  7115 Standard Drive
  Hanover, MD 21076-1320
Computations on Wings with Full-Span Oscillating Control Surfaces using Navier-Stokes Equations

Guru P. Guruswamy
Ames Research Center, Moffett Field, California
Available from:

NASA Center for AeroSpace Information  
7115 Standard Drive  
Hanover, MD 21076-1320  
443-757-5802

This report is also available in electronic form at  
http://www.nas.nasa.gov/publications/reports/reports.html
Computations on Wings with Full-Span Oscillating Control Surfaces using Navier-Stokes Equations

Guru P. Guruswamy
NASA Advanced Supercomputing Division, Ames Research Center Moffett Field, CA

Abstract

This paper presents a time-accurate procedure for computing unsteady aerodynamic forces on wings with a full-span oscillating control surface by using sheared grids. In this procedure, the flow is computed by solving the unsteady Reynolds-Averaged Navier-Stokes (RANS) equations in conjunction with the one-equation Spalart-Allmaras turbulence model. A module to model control surface oscillations is embedded into a general purpose RANS solver. Unsteady pressures and forces are validated with experimental data for wall-to-wall mounted and finite-tip wings with full span control surfaces. The effect of Mach number on unsteady aerodynamic loads is investigated for the transonic flow regime.

I. Introduction

The use of active controls for fixed-wing aircraft [1] and rotorcraft [2] is becoming more important for modern configurations. Advanced transport designs, such as blended wing-bodies [3], need active controls to achieve safe and efficient flight. Efforts to reduce the vibrations of helicopter blades with use of active-controls are also in progress [2].

Modeling oscillating control surfaces using linear aerodynamics theory is well established. However, higher-fidelity methods are needed to account for nonlinear effects, such as those that occur in transonic flow [4]. The aeroelastic responses of a wing with an oscillating control surface, computed using the transonic small perturbation (TSP) theory, have been shown to cause important transonic flow effects [5] such as a reversal of control surface effectiveness that occurs as the shock wave crosses the hinge line. In order to account for flow complexities such as blade-vortex interactions of rotor blades [6] and leading edge vortex-induced aeroelastic oscillations of blended wing body transports [3, 7], higher-fidelity methods based on the Navier-Stokes equations are used.

Since the original development of moderate-fidelity TSP methods [5, 8], higher-fidelity flow equations have come into use to model oscillating control surfaces. The first successful computation using TSP for a fixed wing with an oscillating control surface in the transonic flow regime is reported in Ref. 8. Computations using transpiration boundary conditions (without physically moving the control surface) were made in Ref. 9, using the full-potential theory. Ref. 10 presents a procedure that uses the Navier-Stokes equations with moving-sheared grids and demonstrates up to 8 degrees of control-surface amplitude, using a single grid. Later, this procedure was extended to accommodate larger amplitudes, based on sliding grid zones [11]. Reference 12 reports an effort using a sheared grid to validate results for a transport wing. The sheared grid method implemented in the Euler/Navier-Stokes-based aeroelastic code ENSAERO [13] was successfully applied to active control design by industry [14].

To date, the sheared grid technique of modeling control surfaces has been successfully applied to complex geometries using patched structured grids [15] and has been implemented on parallel computers [16]. The aeroelastic code HIMAP [17] based on the Navier-Stokes equations using patched structured grids includes a control surface module that has been implemented in a parallel computing environment. In addition to such patched structured grid codes, aeroelastic capabilities have been implemented in OVERFLOW [18], which is based on the RANS equations and uses overset grids, which has been well established to model flows over non-oscillating control surfaces [19]. Recently, an oscillating control surface capability for rotorcraft was added to OVERFLOW using overset grids in gaps [20] and was validated for integrated air loads. Ref. 20 reports significant differences between computations and measurements, particularly for flap moments. While the approach presented in Ref. 20 promises to be accurate, a promise that has yet to be realized in such quantities as unsteady surface pressures, it also requires more grid points and a more complicated grid generation process. Time step restrictions associated with tightly spaced grid points in gaps can also be an issue with this approach.
As an alternate approach to the method of modeling gaps with overset grids, the present work presents a sheared grid capability [10] embedded as a module into the OVERFLOW code. In this approach the grid at the flap’s gap for the deflected control surface has the same topology as the grid for un-deflected control surface. Control surface deflections are modeled by shearing the grid at the gap. Sheared grids produce accurate results for moving control surfaces [10, 13, 15] and can be numerically more efficient than methods that use overset grids to model small gaps, which are common when active control surfaces are used. Figure 1 shows an example of a sheared grid in the vicinity of the flap interface. The continuity of the grid is maintained at the hinge line.

This work focuses on implementing and validating the sheared grid approach for use with overset grids similar to that developed for patched grids in Ref. 10. Grids are selected based on grid refinement studies. The effects of oscillating control surfaces on unsteady loads in the transonic flow regime are studied. In addition, this effort makes progress towards providing an efficient and robust high-fidelity analysis tool for designing active controls, particularly for the transonic regime. The approach presented is validated using results from an experiment performed at the National Aerospace Laboratory (NLR), Amsterdam [21], that the Advisory Group for Aerospace Research and Development (AGARD) selected as one of the detailed unsteady pressure experiments involving oscillating control surfaces. Another validation study is made for a finite-wing with a full-span oscillating control surface that was tested at the Royal Aeronautical Establishment (RAE) [22]. A search of the literature indicates that there has been no previous effort to utilize these data to validate flow solvers based on the Euler/Navier-Stokes equations. Recent papers [23, 24] that model moving control surfaces do not include any validation with experiment. Another recent paper [25] validates only forces with experimental data at a lower oscillating frequency than normal operational frequencies of flaps [20]. None of the recent papers [23, 24, 25] include validation of unsteady-surface pressures needed before validating integrated forces [20]. This effort involves detailed validation of unsteady surface pressures and also integrated forces. After validation, the effects of Mach number in the transonic range are presented.

II. Approach

In this paper, the RANS equations [26] are numerically solved using the Pulliam-Chaussee diagonal form of the Beam-Warming central difference algorithm [27], along with the one-equation Spalart-Allmaras turbulence model [28]. The solutions are computed using the OVERFLOW code [18], which is based on an overset grid system. The second-order spatial and temporal accuracy options available in OVERFLOW 2.2c are used throughout this analysis.

In the current version of OVERFLOW, control surface motions are prescribed externally [20] and the flow solver needs to pause while getting the deflection input. In this work, the shearing grid capability is embedded within the OVERFLOW code to avoid the computer overhead time associated with input/output and pause for data transfers. This will facilitate embedding an active control module [5] in OVERFLOW. Grid zones in OVERFLOW
are assigned to different processors by a load-balancing algorithm with no user control. In the present approach it is
assumed that the grid zone containing the control surface is assigned to a single processor without splitting the grid.
However, this limitation could be removed if an option to assign specific processors to selected grid zones, similar
to that implemented in HiMAP [17], were available in OVERFLOW. Confining a shearing grid module to one zone
makes all computations within that module implicit and allows larger time steps for integration. Any impact on
parallel efficiency can be minimized using node-filling algorithm successfully implemented in HiMAP [17].

The control surface deflection at a given time \( t \) is defined as

\[
\delta = \delta_0 + \delta_\beta \sin(\omega t)
\]

where \( \delta_0 \) and \( \delta_\beta \) are the mean and maximum amplitude of the oscillatory control surface deflections, respectively.
The \( \omega \) and \( t \) parameters are circular frequency of the control surface oscillation in radians per sec and time in sec,
respectively. All unsteady components are scaled with \( \delta_\beta \) in radians.

### III. Results for Full Span Control Surface of Wall-to-Wall Mounted Wing

#### A. Wind Tunnel Model

In this work computations are made about an AGARD configuration [21] that involves a wall-to-wall mounted
blade with a full-span oscillating control surface. A schematic diagram of the model is shown in Fig. 2. The model
measures 420 mm in span and 180 mm in chord, and is made of NACA64A006 airfoil sections. The leading edge of
the full-span control surface is located at 75% chord, and from the figure in the report it is estimated that the gap
is about 0.5% of chord. A total of 38 upper and lower surface pressure orifices, including 12 on the control surface,
were used to measure data; the number of orifices used is relatively high compared to similar tests conducted to
date. The measured data are given in the form of magnitude/amplitude and phase angles of unsteady pressures and
sectional lift forces. The phase angle is the difference in period between the peaks of control surface deflection and
the pressure/force response. The amplitude is the absolute value of the maximum response above the time-averaged
value. The magnitude and phase angles are computed by performing a Fourier analysis [29] of time the responses.

#### B. Grid for Flow Solver

A module was written to generate a dynamic moving grid for the oscillating control surface with a single O-H
(wrap around airfoil in x-direction and stacked spanwise in z-direction) grid topology. This module uses the
hyperbolic grid generator available in the Chimera Grid Tools (CGT) software package [30]. Since the gap in the
model is very small it is not modeled in the grid. The airfoil surface grid is sheared by applying the displacements
due to control surface deflections in the normal direction (z) and a new grid is generated.

The final grid selected is based on grid refinement studies for both spatial and temporal accuracies. The majority
of results from the experiment are between free-stream Mach numbers (\( M_\infty \)) 0.8 and 0.9. \( M_\infty = 0.854 \) for which both
mean steady and unsteady pressure data are available is selected as a representative Mach number for grid
refinement studies. At this \( M_\infty \), the peaks in unsteady pressures due shock wave and hinge line are predominant
compared to other lower and higher Mach numbers. The associated Reynolds number \( Re_c \) based on chord \( c \) is
2.44 million.

First, a baseline grid with attributes based on previous experience with algebraic grids [10, 13] is selected. Near-
body grid spacing of 0.0000125 chord, near-body stretching factor of 1.125, average chordwise spacing of 0.005 and
outer boundary (OB) grid surface location at 15c are selected. The grid spacing at the leading edge is about 0.0025c.
With these parameters a baseline (BL) 2D O-grid with 403 points in the wrap around direction and 80 points in the
normal direction is generated using the hyperbolic grid generator [30]. Since the wing is mounted between walls and
measurements are made at the mid-span, 3 planes are used in the spanwise direction to run OVERFLOW in 2-D
mode. Effect of selected parameters on sectional lift coefficient (\( c_l \)) and drag coefficient (\( c_d \)) are studied to determine
the final grid.

Corresponding to one of the test cases steady-flow computations at \( M_\infty = 0.854, \delta_\beta = 1.08 \) deg and \( Re_c \) = 2.4
million, each run for 10000 steps, are performed first, using a variable time-step option in OVERFLOW without
sub-iteration. As seen in Fig. 3, \( c_l \) and \( c_d \) converge in about 5000 iterations. The rest of the steady-state computations
are made using 6000 iterations.

To study adequacy of the grid in the chordwise direction, starting from the BL grid (403 points), computations are
made using a series of grids with increased spacing: double the spacing (203 points), quadruple the spacing (103 points) and octuple the spacing (53 points), while keeping the remaining of the grid parameters the same. This is accomplished by modifying the baseline grid (403 points) using the cubic spline interpolations in CGT [30]. Figure 4 shows the effect of grid refinement in the x-direction on $c_l$ and $c_d$. As seen, both $c_l$ and $c_d$ converge for the baseline grid.

Next normal grid spacing near the surface is varied from 0.00001 to 0.00003. As observed in Fig. 5, the initially selected spacing of 0.0000125 is considered adequate. Effects of decreasing stretching factor at surface in Fig. 6 show that the selected surface stretching factor of 1.125 is adequate. These selected parameters yield an average $y^+$ value of 0.9712 which is considered adequate to resolve flows at the surface.

Using the above selected grid parameters the effect of location of the outer boundary (OB) is studied next. From Fig. 7 it is seen that the outer grid needs to be at least at 25 chords which requires a grid size of 403 x 85.
Among all parameters considered for grid refinement studies it is observed that the outer boundary location has the strongest effect on $c_l$ and $c_d$. As a result, effect of outer boundary location is studied for unsteady cases.

Unsteady computations are performed at $M_\infty = 0.854$, $\alpha = 0.0$ deg, $Re_\text{c} = 2.44$ million, $k = 0.24$ and $\delta_0 = 1.05$ deg corresponding to a test case in the experiment. Computations are performed with a variable number of steps per cycle (NSPC). It was found that NSPC = 1200 produced a stable solution without using Newton sub-iterations (NWIT). The rest of the time step convergence studies are made using NSPC = 1200.

Since Newton iterations are required to maintain 2nd order time accuracy, the next study involved the variation of number of Newton iterations (NWIT). Computations are made for 4 cycles with increasing NWIT. Figure 8 shows plots of $c_l$ for NWIT = 5, 10 and 15. The solutions during the first cycle of the computation have not achieved periodic behavior and should not be quantitatively evaluated. The solutions for cycles 2-6 are periodic. The percentage change in 6th cycle’s peak $c_l$ from NWIT = 5 to 10 is 5% whereas from 10 to 15 is 0.5%. The responses are considered converged for NWIT = 15. Remaining of the unsteady computations are made using 4 cycles of oscillations with NSPC = 1200 and NWIT = 15.

Figure 9 shows the effect of OB location on magnitude and phase angle of $c_l$ obtained from Fourier analysis of the time response. Results in Fig. 9 show that the OB location at 35 chords is adequate, which requires 90 grid points in the normal direction.

C: Test Cases

Reference 21 reports unsteady measured results for various Mach numbers from 0.80 to 0.988 for $k$ ranging from 0.06 to 0.40. In this paper three cases near $k = 0.25$, a typical frequency for real wings, are selected for comparison. They are $M_\infty = 0.804$ ($k = 0.253$, $\delta_0 = 1.11$ deg, $Re_\text{c} = 2.35$ Million), $M_\infty = 0.854$ ($k = 0.240$, $\delta_0 = 1.05$, $Re_\text{c} = 2.41$ Million) and $M_\infty = 0.879$ ($k = 0.234$, $\delta_0 = 1.08$, $Re_\text{c} = 2.44$). For all three cases unsteady computations are made by oscillating the control surface for 4 cycles with NSPC = 1200 and NWIT = 15 using a grid of size (403 x 90). Fourier analyses are performed on each solution response. Comparison of time averaged pressures with experiment are shown in Fig. 10. Comparisons for all three cases are good. Results at $M_\infty = 0.804$ compare better than the other two Mach numbers.

Figure 11 shows detailed comparison of in-phase and out-of-phase components of upper surface pressures with experiment for $M_\infty = 0.854$. Computed trends agree well with experiment. Computed peaks near the shock-wave are sharper than those from measurements. The in-phase peak from the experiment is more smeared than for the out-of-phase component. Out-of-phase components compare better than in-phase components near the shock-wave location. Both computed and measured results show peaks near x/c = 0.75 (hinge location) for in-phase component.
but not for out-of-phase components. Comparisons at the leading edge for the out-of-phase components are less favorable than the in-phase component.

Fig. 9 Effect of outer boundary location on magnitude and phase angles of unsteady $c_l$ for $M_\infty = 0.854$, $Re_c = 2.41$ million, $k = 0.24$, and $\delta_\beta = 1.05$ deg.

Fig. 10 Comparison of time averaged upper surface computed pressures with experiment at $M_\infty = 0.804$ ($k = 0.253$, $\delta_\beta = 1.11$ deg, $Re_c = 2.35$ Million), $M_\infty = 0.854$ ($k = 0.240$, $\delta_\beta = 1.05$ deg, $Re_c = 2.41$ Million) and $M_\infty = 0.879$ ($k = 0.234$, $\delta_\beta = 1.08$ deg, $Re_c = 2.44$ million).

To study the effect of Mach number unsteady loads computations are made for $M_\infty$ varying from 0.80 to 0.90 in increments of 0.1. Average values of $k = 0.242$, $\delta_\beta = 1.08$ deg and $Re_c = 2.40$ million are used for these computations. Figure 12 shows plots of in-phase and out-of-phase components for the oscillatory lift coefficient with increasing $M_\infty$. The in-phase component decreases with increasing $M_\infty$ and the out-of-phase component decreases up to about $M_\infty = 0.86$ then increases. Results from the experiment at $M_\infty = 0.804$, $0.822$, $0.854$ and $0.879$ show similar trends. Generally the in-phase components compare better than out-of-phase components.

Fig. 11 Comparison between computed and measured values of in-phase and out-of-phase components of upper surface oscillatory pressures for $M_\infty = 0.854$, $k = 0.253$, $\delta_\beta = 1.05$ deg and $Re_c = 2.41$ million.

Fig. 12 Comparison between computed and measured values of in-phase and out-of-phase components of sectional lift coefficient for various $M_\infty$ at $k = 0.242$, $\delta_\beta = 1.08$ deg and $Re_c = 2.40$ million.
VI. RAE Wing with Finite Tip

Reference 22 reports wind tunnel results for a low-aspect ratio (1.25) wing (NACA 64006A airfoil section) with a moving full-span control surface (see Fig. 13). The hinge line is located at 67% of chord. A case from experiment at $M_x = 0.243$, $k = 0.88$ and $\delta_0 = 3.5$ deg for which unsteady pressure results are available at 68% span station is selected.

The same chordwise O-grid topology from the NLR case is used for this wing since both have the same NACA64A006 airfoil section. The final grid 403 x 90 obtained after grid refinement studies for NLR case is selected. A H topology is used for the grid in the spanwise direction. Since details are not available, the tip is modeled by tapering the thickness to zero. This requires a total of 58 points in the spanwise direction with 38 points on the wing. Grids are clustered at the root and the tip to accommodate a smooth shearing of the grid.

Since this is a 3-D case, the adequacy of spanwise outer boundary location of the grid is verified. Figure 14 shows the effect of location of far span on sectional lift coefficient at 68% span for $M_x = 0.243$, $k = 0.88$, $\delta_0 = 3.5$ deg and $Re_c = 2.0$ million. The selected outer boundary location of 10 span lengths is adequate. The remaining of the computations are made using a grid with 403 chordwise, 58 spanwise and 90 radial grid points. Figure 15 shows the 3-D grid with details at the tip.
To verify the adequacy of time step size, computations are made for increasing NSPC with NWIT = 15 at $M_\infty = 0.243, k = 0.88$ and $\delta_\beta = 3.5$ deg. It is found that responses become periodic in 4 cycles. Figure 16 shows the effects of NSPC on in- and out-of-phase components of oscillatory $c_l$ at 68% span. Results converge at NSPC = 4800 where NWIT = 10 and 15 give almost identical results.

Figure 17 shows the comparison of computed results with experiment for in-phase and out-of-phase components of upper surface oscillatory pressure. Both components compare well in trend with the experiment. In-phase components compare better than out-of-phase components. Differences near the leading edge are higher than along the rest of the chord. Similar differences in out-of-phase values are reported in Ref. 22 in which comparisons with the linear theory are made. It is noted that because of the low Mach number the out-of-phase values are significantly smaller than in-phase values.

V. Conclusions

Computations of the flows over full-span oscillating control surfaces are performed using the Reynolds-Averaged Navier-Stokes equations. An embedded sheared-grid approach in the context of a general purpose overset based CFD code is used to model the oscillating control surfaces. The procedure is validated by comparing unsteady pressures and integrated forces with experiments. Two test cases with the same airfoil section, one a wall-to-wall mounted wing and another a finite-span cantilever wing, are considered. Grids are selected based on detailed grid refinement studies that involved effects of chordwise spacing, near surface spacing, near surface stretching factor and outer boundary locations. Steady computations show that results are more sensitive to outer boundary location than other parameters. Unsteady computations on finite tip wing need more steps per cycle than computations on 2D airfoil whereas the number of Newton sub-iterations is almost same for both. In general, in-phase components compared better with experiment than out-of-phase components. Moving control surface module developed for a general purpose CFD code will facilitate its application for analysis and design of active controls. The present method will be extended to model part-span control surfaces on parallel computers.

Acknowledgments

The author thanks Henry Lee of NASA Ames Research center for providing a script for the CGT grid tool to generate 2-D grids. Terry Holst, chief of Fundamental Modeling and Simulation Branch at NASA Ames Research Center and CFD lead of Rotary wing project, made valuable suggestions on grid refinement studies.
References


