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James R. Forsythe Dept. of Aeronautics, USAF Academy

Shawn H. Woodson Naval Air Systems Command



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James R. Forsythe^{*}

Dept. of Aeronautics, USAF Academy

Shawn H. Woodson[†]

Naval Air Systems Command

This paper will present unsteady computational fluid dynamics calculations of the abrupt wing stall phenomenon (AWS) on the pre-production F/A-18E using Detached-Eddy Simulation (DES). DES combines the efficiency of a Reynolds-averaged turbulence model near the wall with the fidelity of Large-Eddy Simulation (LES) in separated regions. Since it uses LES in the separated regions, it is capable of predicting the unsteady motions associated with separated flows. DES has been applied to predict the unsteady shock motion present on the F/A-18E at transonic speeds over several angles of attack. Solution based grid adaption is used on unstructured grids to improve the resolution in the LES region. DES mean flow results are compared to leading Reynolds-averaged models showing improved predictive capability. Unsteady surface pressures are shown to be in good agreement with experimental measurements. The presence of low frequency pressure oscillations due to shock motion in the current simulations and the experiments motivated a full aircraft calculation, which showed low frequency high-magnitude rolling moments that could be a significant contributor to the AWS phenomenon. Preliminary single degree of freedom about the roll axis calculations in conjunction with DES are also performed to mirror wind tunnel free-to-roll tests.

Introduction

URING envelope expansion flights of the F/A- $\mathbf{D}_{18\mathrm{E/F}}$ in the Engineering and Manufacturing Development phase, the aircraft encountered uncommanded lateral activity, which was labeled "wing drop". An extensive resolution process was undertaken by the Navy and its contractors to resolve this issue. A production solution was developed, which included revising the flight control laws and the incorporation of a porous wing fold fairing to eliminate the wing drop tendencies of the pre-production F/A-18E/F. The wing drop events were traced to an abrupt wing stall (AWS) on either the left or right wing panel, causing a sudden and severe roll-off in the direction of the stalled wing. An important distinction between wing drop and AWS is that wing drop is the dynamic response of an aircraft to an aerodynamic event, while AWS is an aerodynamic event that can trigger a wing drop.[1]

Unsteady measurements on a model of a preproduction F/A-18E were made by Schuster and Byrd [2], motivated by the following statement: "Since AWS and the resulting lateral instabilities are dynamic or, at best highly sensitive quasi-static phenomena, measurement of unsteady wing surface pressures, loads, and accelerations were incorporated into the test procedures to investigate the potential unsteady causes and/or indicators of AWS." The initial findings from these tests showed highly unsteady surface pressures indicative of shock oscillation.

Although Reynolds-averaged Navier Stokes (RANS) CFD calculations were fairly successful in predicting mean flow characteristics indicative of AWS[3], they failed to predict unsteady shock oscillations even when run time accurate. The turbulence models employed in RANS methods necessarily model the entire spectrum of turbulent motions. While often adequate in steady flows with no regions of reversed flow, or possibly exhibiting shallow separations, it appears inevitable that RANS turbulence models are unable to accurately predict phenomena dominating flows characterized by massive separations. Unsteady massively separated flows are characterized by geometry-dependent and three-dimensional turbulent eddies. These eddies, arguably, are what defeats RANS turbulence models, of any complexity.

To overcome the deficiencies of RANS models for predicting massively separated flows, Spalart *et al.* [4] proposed Detached-Eddy Simulation (DES) with the objective of developing a numerically feasible and accurate approach combining the most favorable elements of RANS models and Large Eddy Simulation (LES). The primary advantage of DES is that it can be applied at high Reynolds numbers as can Reynolds-averaged techniques, but also resolves geometry-dependent, unsteady three-dimensional turbulent motions as in LES. The initial applications of DES were favorable[5, 6, 7] and formed the main mo-

^{*}Assistant Professor, Senior Member of AIAA

[†]Aerospace Engineer, Associate Fellow of AIAA

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tivation for application to abrupt wing stall.

Unsteady shock oscillations have been highlighted by Dolling [8] as a problem for steady state methods. The supersonic separated compression ramp pulses at low frequency. The resulting time-averaged surface pressures are smeared due the time averaging of a moving shock. Accurately predicting this flow has eluded CFD researchers for decades. Dolling [8] suggests that better agreement with time-averaged experimental data could be obtained if the CFD simulation included the global unsteadiness of the shock motion, then took a time average. This is the approach that is taken in the current research.

Besides obtaining an improved time-averaged prediction, however, it is also desired to complement unsteady wind tunnel methods[2] with CFD to gain further insight into the potential of the unsteady flow to contribute to the AWS phenomena. The CFD complements the experiments by providing results unaffected by aeroelastic effects, and more detailed flow visualizations.

The baseline case considered is an 8% model of a preproduction F/A-18E with $10^{\circ}/10^{\circ}/5^{\circ}$ flaps (leadingedge flaps/trailing-edge flaps/aileron flaps) at Mach 0.9 and no tails. DES calculations are performed on a baseline and adapted grid and compared to unsteady wind tunnel measurements and RANS models. Although not a comprehensive validation, confidence is built in the DES method for this class of flow. Preliminary DES single degree of freedom calculations are also presented that are intended to mirror wind tunnel free-to-roll tests[9].

In order to obtain approval for releasing this paper to the public, quantitative information has been removed from most vertical scales as per guidelines from the Department of Defense.

Governing Equations and Flow Solver

The unstructured flow solver *Cobalt* was chosen because of its speed and accuracy (*Cobalt* is a commercial version of $Cobalt_{60}$). Strang *et al.* [10] validated the code on a number of problems, including the Spalart-Allmaras model (which forms the core of the DES model). Tomaro *et al.* [11] converted $Cobalt_{60}$ from explicit to implicit, enabling CFL numbers as high as one million. Grismer et al. [12] then parallelized the code, yielding a linear speedup on as many as 1024 processors. Forsythe et al. [13] provided a comprehensive testing and validation of the RANS models: Spalart-Allmaras, Wilcox's $k - \omega$, and Menter's models. The Parallel METIS (ParMetis) domain decomposition library of Karypis and Kumar [14] and Karypis *et al.* [15] is also incorporated into *Cobalt*. ParMetis divides the grid into nearly equally sized zones that are then distributed among the processors.

The numerical method is a cell-centered finitevolume approach applicable to arbitrary cell topologies (e.g, hexahedrals, prisms, tetrahdra). The spatial operator uses the exact Riemann Solver of Gottlieb and Groth [16], least squares gradient calculations using QR factorization to provide second-order accuracy in space, and TVD flux limiters to limit extremes at cell faces. A point implicit method using analytic first-order inviscid and viscous Jacobians is used for advancement of the discretized system. For time-accurate computations, a Newton sub-iteration scheme is employed, and the method is second-order accurate in time.

The compressible Navier-Stokes equations were solved in an inertial reference frame. To model the effects of turbulence, a turbulent viscosity (μ_t) is provided by the turbulence model. To obtain k_t (the turbulent thermal conductivity), a turbulent Prandtl number is assumed with the following relation: $Pr_t = \frac{c_p \mu_t}{k_t} = 0.9$. In the governing equations, μ is replaced by $(\mu + \mu_t)$ and k (the thermal conductivity) is replaced by $(k + k_t)$. The laminar viscosity, μ , is defined using Sutherland's law.

Reynolds-Averaged Models

In order to provide a baseline for comparison, computations were performed with two of the leading Reynolds-averaged models. The first model used was the Spalart-Allmaras (SA) one-equation model[17]. This model solves a single partial differential equation for a variable $\tilde{\nu}$ which is related to the turbulent viscosity. The differential equation is derived by, "using empiricism and arguments of dimensional analysis, Galilean invariance and selected dependence on the molecular viscosity." The model includes a wall destruction term that reduces the turbulent viscosity in the log layer and laminar sublayer, and trip terms that provide a smooth transition from laminar to turbulent. For the current research, the trip term was turned off, and the flow assumed fully turbulent.

The second model used was Menter's Shear Stress Transport (SST) model[18, 19]. The method is a blend of a $k-\epsilon$ and $k-\omega$ model which uses the best features of each model. The model uses a parameter F_1 to switch from $k-\omega$ to $k-\epsilon$ in the wake region to prevent the model from being sensitive to freestream conditions. The implementation used includes a compressibility correction as detaild in Forsythe *et al.* [13].

Detached-Eddy Simulation

The original DES formulation is based on a modification to the Spalart-Allmaras RANS model[17] such that the model reduces to its RANS formulation near solid surfaces and to a subgrid model away from the wall[4]. The basis is to attempt to take advantage of the usually adequate performance of RANS models in the thin shear layers where these models are calibrated and the power of LES for resolution of geometrydependent and three-dimensional eddies. The DES formulation is obtained by replacing in the S-A model the distance to the nearest wall, d, by \widetilde{d} , where \widetilde{d} is defined as,

$$\widetilde{d} \equiv \min(d, C_{DES}\Delta) \,. \tag{1}$$

In Eqn. (1), for the computations performed in this project, Δ is the largest distance between the cell center under consideration and the cell center of the neighbors (i.e., those cells sharing a face with the cell in question). In "natural" applications of DES, the wall-parallel grid spacings (e.g., streamwise and spanwise) are at least on the order of the boundary layer thickness and the S-A RANS model is retained throughout the boundary layer, i.e., d = d. Consequently, prediction of boundary layer separation is determined in the 'RANS mode' of DES. Away from solid boundaries, the closure is a one-equation model for the sub-grid-scale (SGS) eddy viscosity. When the production and destruction terms of the model are balanced, the length scale $d = C_{DES}\Delta$ in the LES region yields a Smagorinsky eddy viscosity $\tilde{\nu} \propto S \Delta^2$. Analogous to classical LES, the role of Δ is to allow the energy cascade down to the grid size; roughly, it makes the pseudo-Kolmogorov length scale, based on the eddy viscosity, proportional to the grid spacing. The additional model constant $C_{DES} = 0.65$ was set in homogeneous turbulence^[5], and was used in the following calculations.

Although Strelets [6] formulated a DES model based on Menter's Shear Stress Transport model, the current calculations used only the Spalart Allmaras based version. This approach was used since the separated region is handled in LES mode and should therefore not be too sensitive to the underlying RANS model. Spalart Allmaras based DES has the advantage that it is one versus two equations, and is in general more robust since the gradients of the turbulence model variable is linear aproaching the wall.

Results

Calculation Details

As previously mentioned, the configuration examined was an 8% scale pre-production F/A-18E with $10^{\circ}/10^{\circ}/5^{\circ}$ flaps set. All of the calculations other than the single degree of freedom calculations were carried out on a model with no vertical or horizontal stabilizer (no tails). The force coefficients presented here are compared to a no tails wind tunnel model. Wing surface pressures are compared to a wind tunnel model with tails, however there was seen to be good agreement in surface wing pressures between a model with tails, and that without. The Mach number for all cases was 0.9, and the Reynolds number was 3.8×10^6 per foot, leading to a chord based Reynolds number of 3.98×10^6 . This Reynolds number was set by adjusting the freestream temperature and setting standard day sea level pressure. In order to compare frequencies and times to unsteady wind tunnel data, the resulting

times in the CFD calculations were scaled by the ratio of the CFD freestream velocity to the wind tunnel freestream velocity (a factor of 1.28). The wind tunnel comparisons are from the model tested in NASA Langley's 16 ft Transonic Tunnel (16TT). The wing was instrumented with both steady and unsteady pressure taps as shown in Figure 1. This paper will focus on the G row (highlighted), since it is directly behind the snag (in the streamwise direction), where the shock induced separated flow occurred furthest forward.



Fig. 1 Experimental pressure ports.

The grids used were unstructured grids created using the tetrahedral grid generator VGRIDns[20]. The Cobalt utility blacksmith was used to recombine the high aspect ratio tetrahedra in the boundary layer into prisms. The "Baseline" grid was 7.3×10^6 cells for half the aircraft. The average first y^+ for the grid was 0.2 with a geometric growth rate of 1.25. An adapted grid was created in an attempt to improve on poor DES results on the baseline grid at 9° angle of attack. The utility (fv2usm) was used to convert the *Cobalt* solution file to a format readable by RefineMesh (a companion to VGRIDns - see Morton et al. [21]). The solution used for adaption was the time averaged solution from a DES 9° angle of attack run. A level of vorticity was selected that contained the separation bubble, and the grid spacing reduced by a factor of 0.6 in each coordinate direction. This should in general lead to $(1/0.6)^3 = 4.63$ times the number of points. However since this reduction in spacing was only applied in a narrowly focused region, the grid only increased from 7.3×10^6 to 9.1×10^6 cells. Cross sections of the "Baseline" and "Adapted" grids are shown in Figure 2. A sample instantaneous DES solution at 9° angle of attack is shown in Figure 3 on the G row. The LES character of DES is clearly shown - as the grid spacing is reduced, smaller and more turbulence length scales are resolved. This reduces the modeling errors by increasing the resolved turbulence. By comparing Figure 2 to 3, it is also seen that the increased density of points is efficiently placed where needed in the separation bubble. Although the adaption was carried out at a single angle of attack, the grid was used for the other angles. For lower angles, the separation bubble is further aft, so the adapted region included the separation bubble. For angles higher than 9° , the separation bubble was larger than the adapted region. Both the baseline and the adapted grid were constructed to provide a "natural" application of DES. That is the grid spacing in the wall parallel direction was larger than the boundary layer thickness, allowing the boundary layer to be treated by RANS upstream of separation.



Fig. 2 Baseline vs. adapted grid for F/A-18E with no tails.



Fig. 3 Baseline vs. adapted grid for F/A-18E with no tails.

For the RANS calculations, the code was run at a specified maximum global CFL of 1.0×10^6 to accelerate the convergence to steady state. Previous unsteady solutions using RANS models had all failed to obtain any significant levels of unsteadiness. Convergence was assessed by monitoring forces and moments during the run. When the change in forces and moments was less than 1% over 500 iterations, the solution was considered converged. This occurred between 2,000 and 4,000 iterations depending on the angle of attack.

DES calculations were of course performed timeaccurate. Three Newton subiterations were used, based on previous experience. To ensure a proper choice in timestep, a timestep study was performed on the adapted grid. The timesteps examined were $0.64x10^{-5}$, $1.28x10^{-5}$, and $2.56x10^{-5}$ seconds. These timesteps corresponded to non-dimensional (by chord and freestream velocity) timesteps of 0.006, 0.012, and 0.024 respectively. The flow was first initialized by running the middle timestep for 4000 iterations. Then the calculations were run for 8000, 4000, and 2000 iterations respectively over the same length of physical time (0.0512 seconds). Power spectra of the half-aircraft rolling moment for the three timesteps is plotted in Figure 4. There is fairly poor agreement on the power at the low end of the spectra (below 100 Hz) for the smallest timestep. It should be noted, however, that the length of time integrated over is quite small (only able to define 20 Hz), and the low end of the spectra may need longer sampling to define it well. The middle frequency range agrees fairly well for all timesteps (between 100 and 2000 Hz). The largest timestep starts to fall below the others at 2000 Hz. This represents about 20 iterations per cycle, a reasonable value for a second order accurate code. The middle timestep falls off at about 4000 Hz. This middle timestep is used for all the subsequent calculations. It should also be noted that this spectra provides strong evidence that DES is acting in LES mode since there is a broad range of frequencies resolved, and a healthy inertial subrange. For the subsequent DES calculations, the flow was initialized over a time of 0.0512seconds, then time averages were taken over at least an additional 0.0512 seconds.



Fig. 4 Power spectral density plot of half aircraft rolling moment at various timesteps, 9° angle-of-attack.

Steady/Time-Averaged Results

One of the motivating factors behind using a turbulence resolving method such as DES is to provide a more accurate time-averaged solution, mean lift and drag for example. This has proven true for a broad range of massively separated flows, such as cylinders, spheres, airfoils/forebodies/aircraft at high alpha, but has not been examined on a shock separated flow.

Time averaged-DES lift, drag, and moment coefficients are plotted vs. RANS calculations, and experimental values in Figures 5, 6, and 7 respectively. The experimental results were for the same configuration, i.e. without tails. The DES on the baseline grid follows the lift curve nicely up until 9° , where it drops in lift relative to the experiment. This discrepancy is what prompted the creation of the adapted grid, which matched the experiments better. The adapted grid matches the experiments quite well at all angles, with the largest discrepancies at 12° and 16° . This slight error could perhaps be removed/reduced by adapting a grid to the flow solution at these angles, since the adapted grid was tailored to 9° , which has a smaller separation bubble than the higher angles. The Spalart-Allmaras RANS results over predict the lift at all angles, even at the low angles. Parikh and Chung [22] performed SA calculations on an F/A-18E with the same flap settings and picks up the lift break between 9° and 12° , where we don't have calculations. The Menter's SST model captures the low angles better but the lift curve breaks slightly early. The drag curve (Figure 6) shows essentially the same trends over prediction by SA at all angles, an underprediction by SST near the lift break, and good agreement for the adapted DES.



Fig. 5 Lift Coefficient vs. alpha for the no tails F/A-18E.

The pitching moment coefficient (Figure 7) shows the most sensitivity to the model. Since the current



Fig. 6 Drag Coefficient vs. alpha for the no tails F/A-18E.

grid has no tails, the moment coefficients are quite different than those presented by Parikh and Chung [22]. The adapted DES grid shows quite good agreement throughout the entire angle of attack range. SA underpredicts the moment, while SST overpredicts it at all but the two lowest angles.



Fig. 7 Pitching Moment Coefficient vs. alpha for the no tails F/A-18E.

To understand the differences between the models, pressure coefficients along the G row are plotted vs. experiments in Figures 8, 9, and 10 for 2° , 9° , and 12° respectively. Figure 8 suggests that experimentally there is separation over the trailing edge flap/aileron at 2° . Adapted DES does a good job of picking up the pressure level on the aileron correctly, although the agreement at the trailing edge is not perfect (neither is the pitching moment at this angle). SST only slightly overpredicts the pressure, hence the close but slight overprediction of lift. SA overpredicts the pressure on the flap by a significant amount, which is likely the cause for the overprediction in lift throughout the low angles.



Fig. 8 Time-averaged pressure coefficient vs. chord location for the no tails F/A-18E on the G row, 2° angle-of-attack.

At 9° (Figure 9) the experiments show a smoothly varying pressure distribution from the snag back to about the half chord. Schuster and Byrd [2] showed with unsteady pressure measurements that this pressure distribution occurs due to the time-averaging of an unsteady shock that moves back and forth over the wing. This is certainly a difficult effect for the RANS models to pick up. In this case both SA and SST predict relatively sharp shocks - with SST separating early, and SA late. The DES adapted solution, as will be discussed in the following section, contains a moving shock, that when time-averaged gives a smeared out pressure profile. The time averaged pressures suggest that the unsteady shock stays too far forward compared to the experiments.

At 12° (Figure 10) the flow is separated over the entire chord from the leading edge of the wing. SA overpredicts the pressure (and therefore the lift), while DES and SST match quite well. The fact that SST matches so well here suggests that the errors in pitching moment are arising from a location other than behind the region along the G row.

Unsteady Results

To assess the accuracy of DES in computing unsteady effects associated with AWS, comparisons are made to the unsteady experimental data of Schuster and Byrd [2]. The effect of the unsteady shock on the mean pressure profile is shown in Figure 11. This plot shows instantaneous pressures at four different times as well as the average pressure for the DES calculation at 9° angle of attack. Although the instantaneous



Fig. 9 Time-averaged pressure coefficient vs. chord location for the no tails F/A-18E on the G row, 9° angle-of-attack.



Fig. 10 Time-averaged pressure coefficient vs. chord location for the no tails F/A-18E on the G row, 12° angle-of-attack.

shocks are all sharp, when time averaged a smooth pressure profile results.

Comparisons between the DES calculations and the experiments are shown in Figures 12, 13, and 14. Surface pressures along the G row are plotted, where the experiments had six unsteady pressure taps and ten steady taps. Additionally, there were five steady pressure taps on the bottom of the wing. It was impracticable to store the entire set of CFD results for all timesteps, so the CFD calculations were "tapped" on the G row, and pressures saved every five iterations for subsequent post processing. For the baseline calculations, only the 16 experimental taps on the top of the wing were used. For the refined grid calcula-

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Fig. 11 Pressure contours from the DES adapted calculation at four instants in time, and time-averaged at 9° angle of attack

tions, 100 equally spaced points on the G row were tapped on both upper and lower surfaces to allow for more detailed analysis of the shock motion. Pressure statistics were calculated from the experiments and CFD, including the mean, standard deviation, and the minimum and maximum values of pressure. For both the CFD and experiments, any individual pressure that fell outside a three-standard-deviation (3σ) band about the computed mean was excluded for the maximum or minimum pressure value. For the CFD calculations this mainly smoothed out the min and max coefficients of pressure behind the shock location.

Statistics at 7° are plotted for the baseline grid in Figure 12. The five experimental mean pressures near the bottom of the plot are from the lower wing surface where the CFD pressures were not examined. The agreement in the mean, maximum, and minimum pressures on the top surface is quite good. The shock in the CFD is slightly too far forward and the range of pressure oscillations is slightly underpredicted.

Statistics at 9° are plotted for the baseline and adapted grids in Figure 13. The oscillations in the baseline grid were underpredicted and the shock too far forward. The adapted grid helped improve the results - increasing the amount of shock oscillation, and moving the mean shock location further aft. These improvements showed up as an improved mean lift prediction as previously discussed.

Statistics at 12° are plotted for the adapted grid in Figure 14. The agreement of the maximum, minimum, and average pressure to the experiments is quite good. The pressures had only weak oscillation since the flow was fully separated, and there was no shock oscillation as in the 7° and 9° cases.



Fig. 12 Min, Max, and average pressure coefficient on the G row, 7° angle-of-attack.



Fig. 13 Min, Max, and average pressure coefficient on the G row, 9° angle-of-attack.



Fig. 14 Min, Max, and average pressure coefficient on the G row, 12° angle-of-attack.





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To get a feel for how the frequencies of shock motions compared to wind tunnel tests, pressure coefficients were plotted vs. time at the six unsteady taps on the G row in Figure 15. The flat low pressure spots on these plots correspond to times when the shock is aft of that particular spot. This plot first of all shows that the integration time for the CFD was quite small. The second tap shows that the shock starts off in front of the tap, moves behind it for a short time, then back in front. Therefore only one shock oscillation is captured in this calculation. This shock oscillation appears to be at a frequency similar to the wind tunnel results. However the CFD needs much longer sampling to provide a useful comparison of frequency content. It appears the shock never reaches the third through sixth taps in the CFD. For the experiments the shock reaches as far aft as the fifth tap, but only very rarely. This shows that even if the CFD were perfectly matching the physics of the experiment, it may take very long integration times to match time averaged surface pressure distributions. Thus there is potential room for improvement in the current calculations by taking longer time averages. The first tap in the CDF calculation has a very strong increase in pressure that was seen visually to be separation off the leading edge of the snag. Although this does not seem to occur in the experiments at this angle, similar events were seen at $9.5^{\circ}[2]$. The higher frequency content in the CFD may simply be due to the higher sampling rate — 15.6kHz vs. 1kHz.

To determine if unsteady shock oscillation could be a contributor to the AWS phenomenon, half-aircraft rolling moment is next examined in Figure 16. The half aircraft rolling moment was calculated by taking the rolling moment of the half-aircraft and nondimensionalizing by the span and half the wing area. This of course leads to a non-zero mean coefficient, but a feel for the level of unsteadiness in rolling moment can be obtained by comparing the peak to peak differences. The differences in peaks in Figure 16 although not shown on the axis was considered "significantly large" and a potential contributor to triggering an AWS event. A small slice of this rolling moment plot is shown in Figure 18 with flow visualizations at seven instants in time. Figure 18a corresponds to a large rolling moment, since it has low lift, which would produce a right roll. In Figure 18b, a tiny separation bubble forms on the snag, further reducing lift and increasing the rolling moment. The shock then moves back in Figure 18c-e until the lift is at a maximum, and the rolling moment is at a minimum. From that point it moves forward in Figure 18f-g. The cycle can then repeat.

What is significant is that this shock motion causes a rolling moment change at a low frequency - approximately 25 Hz. This would scale to 2Hz for the full scale aircraft. This was however only a half aircraft calculation, so care must be taken in drawing conclusions from this plot. The net rolling moment will depend on the flow on the other wing. It would in general be possible for the shock locations on the other wing to be perfectly correlated, and therefore have zero rolling moment. Given the chaotic nature of the flow, however, this seems extremely unlikely. Another possibility is that the shocks on both wings oscillate in a very narrow frequency range, which could give rise to a very low beat frequency. Schuster and Byrd [2], however, shows that the shock motion occurs in a broadband frequency range so this would not be expected.



Fig. 16 Rolling moment vs. time for half aircraft calculation (no tails).



Fig. 17 Rolling moment vs. time for full aircraft calculation (no tails).







Fig. 19 Plot of Rolling moment vs. time and flow visualizations at specific times – whole aircraft. Flow visualizations are isosurfaces of vorticity colored by pressure (no tails).

In order to provide conclusive evidence that the low frequency shock motion could lead to large low frequency rolling moments on a full aircraft, the halfaircraft tail-less adapted grid was mirrored around the plane of symmetry leading to an 18.2×10^6 cell grid. This grid was then run in the same manner as previous calculations, at 9° angle of attack. The resulting rolling moment is shown in Figure 17, with the same scale as Figure 16, but centered on zero. The magnitude and frequency of the full aircraft calculation seems to match up fairly closely to the half aircraft calculation. The first 1/3 of the time represents the initial start up of the flow solution and would normally be discarded. However, it is interesting to see that there is a growth of lateral instabilities despite the grid and initial flowfield symmetry. This initial asymmetry must come from slight asymmetries in the flow solver (asymmetries in the grid partitioning, ordering of the grid, machine roundoff, etc) that are then amplified by the unstable nature of the flow. If these initial asymmetries were not present then it would be necessary to provide some flow field asymmetry in the initial conditions. Comparisons are not made to unsteady rolling moments from the experiments since they were believed to be polluted by aeroelastic effects since the frequency of rolling moment oscillation correlated with one of the aeroelastic modes of the model, rather than frequencies from the surface pressures. However, the magnitude of the maximum rolling moment of the CFD calculation was similar to that seen in the wind tunnel.

Flow visualizations are also provided for this calculation in Figure 19, with a zoomed in region of the rolling moment plot. These isosurfaces of zero streamwise velocity are an indicator of the separated region. The shock on the left side starts further back in Figure 19a, giving a large positive right rolling moment. As this shock moves forward, the rolling moment moves towards zero in Figure 19b. Then the right shock moves aft in Figure 19c, giving a large negative rolling moment.

Free-to-roll CFD - a preview

The DES calculations outlined above have built confidence in DES as a baseline tool for prediction of the unsteady effects associated with abrupt wing stall. Although further testing of the method is suggested, such as longer time averaging, and mean force calculations with various roll angles, it was decided to begin an attempt to reproduce wind tunnel free-to-roll tests[9] by using a single degree of freedom model and rigid body motion. Rigid motion has been used with *Cobalt* and DES for prediction of spin characteristics of forebodies[23], and the F-15E[24]. A one degree of freedom model (1-DOF) was implemented by Cobalt Solutions to support this project. The grid used for the 1-DOF simulation was provided by NASA Langley, and was a F/A-18E with tails and $10^{\circ}/10^{\circ}/5^{\circ}$ flaps. Having a grid with tails is crucial for the 1-DOF simulation so the model has the correct lateral stability characteristics. A flow solution for a half aircraft was first run at 9° angle of attack, and the solution used for adaption. The adapted grid was then mirrored along the plane of symmetry to provide a full aircraft grid with a total of 8.36×10^6 cells. The grid spacing in the separation region is slightly finer than the no tails baseline grid, but not as fine as the no tails adapted grid.

The angle chosen for examination was 8.5° at M=0.9, since it had a lot of lateral oscillations in the wind tunnel tests[9]. The dominant frequency of rolling motion of the wind tunnel tests was around 0.6 Hz. At the current CFD timestep, this means that around 50,000 iterations will be required to capture a single cycle! However, the 16TT wind tunnel model had a moment of inertia 15-20 times larger than a dynamically scaled model[25]. This would naturally lead to lower frequency content than a full scale model (in non-dimensional terms). This fact was confirmed by a FTR test in R134a gas where the model's moment of inertia was only 2-3 times larger than a dynamically scaled model. The non-dimensionalized frequencies were about five times higher than the 16TT tests. However the fidelity of these tests is not expected to be as good as the 16TT tests, so CFD calculations will be performed to match the 16TT tests. Although these calculations will be extremely costly, they are necessary to build confidence in the method. Once that confidence is built, it could be used to predict the full scale vehicle at a much lower cost.

Currently a 1-DOF simulation has been performed with an artificially low moment of inertia. The free axis was set to the longitudinal axis of the model through the moment reference point. The resulting bank angle and roll rate vs. time is shown in Figure 20. The characteristic frequency is around 25Hz – far higher than the experiments due to the lowered moment of inertia. A visualization of the aircraft at the maximum bank is shown in Figure 21. The resolution of turbulent structures near separation is poorer than the no tails adapted grid. However, prior to releasing the model into one degree of freedom, the lift coefficient was compared to experiments and was a good match. Also unsteady rolling moments were observed that were similar in magnitude and frequency as those presented for the no tails adapted grid.

A run matching the wind tunnel inertia is currently underway but will take a couple of months of run time to collect several cycles of data. The current calculations although preliminary, however, do show that resolving the unsteady shock motion using DES can provide a trigger event to induce lateral motions. Whether are not these motions can then be predicted accurately using a 1-DOF model remains to be seen.



Fig. 20 CFD Single degree of freedom bank angle and roll rate vs. time - F/A-18E with tails



Fig. 21 1-DOF calculation of the F/A-18E with tails. Bank angle is 6° . Isosurface of vorticity colored by pressure.

Concluding Remarks

In summary, Detached-Eddy Simulation has been applied to the pre-production F/A-18E with $10^{\circ}/10^{\circ}/5^{\circ}$ flap set with comparison to steady and unsteady experimental measurements and leading RANS models. Comparisons were made to experimental surface pressures (both time-averaged and steady) and mean force coefficients. Solution based adaptation was used to improve the simulations. Unsteady rolling moments were observed on both half and full aircraft simulations due to unsteady shock motions.

The mean flow predictions on the adapted grid were seen to be in excellent agreement to the experiments, showing a slight improvement over the SST RANS model, and a larger improvement over the SA RANS model. In deciding whether to use DES or RANS to provide mean flow predictions to look for susceptibility to AWS, the cost must be considered. The DES calculations performed here were about 5-10 times more expensive than a steady RANS calculation since twice the number of Newton subiterations were used, time averages were taken over twice as many iterations, and an adapted grid with about 30% more points was needed. Note that the cost for an unsteady RANS calculation would have been the same as a DES calculation, however unsteady RANS failed to go unsteady. At this cost, it may be impracticable to do DES calculations throughout the entire flight envelope. This is especially true since the RANS models were not as accurate, but did at least predict the correct trends in the lift curve break. However, a potential use of DES would be to use RANS to first find the Mach numbers and angles where AWS is suspected. Then DES could be used in focused calculations in these areas. As computer speeds increase the flight envelope where DES is used could be expanded as its cost is decreased.

Comparisons to unsteady pressures built confidence in the accuracy of DES for this class of flows, but highlighted a need for longer time-averaging. Current computational speeds put the wind tunnels far ahead in their ability to look at the relatively low frequency shock oscillation associated with this flow. As computer speeds increase this gap will naturally narrow. Calculations on both half and full span aircraft showed large oscillations in rolling moment at low frequency (close to 2Hz when scaled to full scale). This supports the conclusion from the unsteady experiments [2]: "This is significant since the combination of large-scale shock motion and low frequency provide a potential triggering mechanism for lateral instabilities, such as wing drop, which probably could not be effectively damped by the automatic flight control system." Because the CFD calculations were for a completely rigid aircraft, there is strong support of the conclusion that "the unsteady aerodynamics experienced on the F/A-18E model at AWS conditions are not a direct result of the structural vibrations encountered in the wind tunnel."[2]

One degree of freedom calculations to match free-toroll tests are highly preliminary, but underway. The ability of the unsteady shock motion resolved by DES to trigger lateral motions was demonstrated. RANS has thus far been unable to resolve the unsteady shock motion and therefore would not be useable for a 1-DOF calculation unless another trigger mechanism were used. The 1-DOF calculation will be costly at wind tunnel conditions, but more affordable at full flight scales. Calculations at wind tunnel conditions are needed, however, to build confidence in the method. Since many of the FTR tests included guite long periods with no motion, it is unlikely that 1-DOF CFD calculations will be able to replace wind tunnel FTR tests any time soon due to the long integration times needed to capture such a low frequency phenomena. 1-DOF calculations could prove useful, however,

to help quantify the effects of not having a dynamically scaled model, which is in general difficult. Also the CFD provides more off body flow data that could be used to gain insight into the lateral motions.

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