CFD Investigation of 2D and 3D Dynamic Stall

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Abstract

The results of numerical simulation for 2D and 3D dynamic stall case are presented. Square wings of NACA 0012 and NACA 0015 sections were used and comparisons are made against experimental data from Wernert *et al.*for the 2D and Schreck and Helin for the 3D cases. The well-known 2D dynamic stall configuration is present on the symmetry plane of the 3D cases. Similarities between the 2D and 3D cases, however, are restricted upto the midspan and the flowfield is markedly different as the wing-tip is approached. Visualisation of the 3D simulation results revealed the same omega-shaped dynamic stall vortex which was observed in the experiments by Freymuth, Horner *et al.*and Schreck and Helin. Detailed comparison between experiments and simulation for the surface pressure distributions is also presented along with the time histories of the integrated loads. To our knowledge this is the first detailed study of 3D dynamic stall. ¹

Notation

α^+	Nondimensional	pitch	rate α^+	_	$\frac{d\alpha}{dt}$	<u>c</u>
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c Chord length of the aerofoil

$$C_P$$
 Pressure coefficient $C_P = \frac{1}{2\rho U^2} (P - P_{inf})$

- C_L Lift coefficient $C_L = \frac{1}{2c\rho U_{inf}^2}(L)$
- *d* Distance along the normal to chord direction
- *x* Chord-wise coordinate axis (CFD)
- y Normal coordinate axis (CFD)
- *z* Span-wise coordinate axis (CFD)
- k Reduced frequency of oscillation, $k = \frac{\omega c}{2U_{inf}}$
- L Lift force
- M Mach number
- *Re* Reynolds number, $Re = \rho U_{inf} c/\mu$
- U Local streamwise Velocity
- U_{inf} Free-stream streamwise velocity

Greek

- α Oscillatory incidence
- α_0 Mean incidence for oscillatory cases
- α_1 Amplitude of oscillation
- ρ Density
- ρ_{∞} Density at free-stream

Acronyms

- AoA Angle of Attack
- AR Aspect Ratio
- PA Pitch Axis
- CFD Computational Fluid Dynamics
- PIV Particle Image Velocimetry
- LDA Laser Doppler Anemometry

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Introduction

Unlike fixed-wing aerodynamic design which usually involves significant Computational Fluid Dynamics (CFD), rotarywing design utilises only a small fraction of the potential CFD has to offer. The main reason for this is the nature of the flow near the lifting surfaces which is complex, unsteady and turbulent. The numerical modelling of such flows encounters three main problems due to a) the lack of robust and realistic turbulence models for unsteady separated flows, b) the CPU time required for computing the temporal evolution and c) the lack of experimental data suitable for validation of the computations. This paper presents a fundamental study of the 3D dynamic stall of a finite wing which contains some of the important features encountered for helicopter rotors and aircraft during manoeuvers.

Dynamic stall (DS) is known to the aerodynamics community and is one of the most interesting phenomena found in unsteady aerodynamics. DS occurs when a lifting surface is rapidly pitched beyond its static stall angle, resulting in an initial lift augmentation and its subsequent loss in a highly non-linear manner. This lift augmentation is due to the formation of large vortical structures over the suction side of the wing. It has also been established that a predominant feature of dynamic stall is the shedding of vortical structures near the leading edge which pass over the upper surface of the aerofoil, distorting the chord-wise pressure distribution and producing transient forces that are fundamentally different from their static counterparts [1]. While the primary vortex is resident above the aerofoil, high values of lift are experienced which can be exploited for the design of highly maneuverable aircraft. The penalty however, is that this primary vortex eventually detaches from the surface and is shed downstream producing a sudden loss of lift and a consequent abrupt change in pitching moment [1]. The phenomenon continues either with the generation of weaker vortices if the body remains above its static angle of attack, or is terminated if the body returns to an angle sufficiently small for flow reattachment. During the DS the flow field includes boundary-layer growth, separation, unsteadiness, shock/boundary-layer and inviscid/viscous interactions, vortex/body and vortex/vortex interactions, transition to turbulence and flow re-laminarisation.

Rotor performance is limited by the effects of compressibility on the advancing blade and DS on the retreating blade. Effective stall control of the retreating blade of a helicopter rotor could increase the maximum flight speed by reducing rotor vibrations and power requirements. Consequently, the study and understanding of 3D DS flow phenomena would assist the rotorcraft industry in further pushing the design limits towards faster and more efficient rotors. In a similar way the maneuverability of fighters could be enhanced if the unsteady air-loads generated by dynamic stall were utilised in a controlled manner. Furthermore, improved understanding of wind turbine blade dynamic stall could enable more accurate engineering predictions, and appreciably reduce the cost of wind energy.

To date there have only been a very limited number of three-dimensional dynamic stall experiments and no detailed numerical studies. However, conclusions drawn from twodimensional numerical investigations and in particular regarding the turbulence modelling [2, 3] can be used as a guide for three-dimensional computations. Three-dimensional experiments have been undertaken by Piziali [4], Schreck and Helin [5], Tang and Dowell [6], Wernert et al. [7], Coton and Galbraith [8] and the Aerodynamics Laboratory of Marseilles (LABM) [9]. All the above works attempted to perform parametric investigations of the Reynolds number and reduced frequency effects on the dynamic stall of NACA 0012 and NACA 0015 wings. Flat or rounded wing tips were used along with splitter plates on the wing root. The Reynolds number was close to 5 $\times 10^5$ (with an exception for the case of Schreck and Helin) and the experiments include harmonically oscillating and ramping motions. Quasi-steady measurements were also taken as part of all the aforementioned experimental programmes which were conducted in the incompressible flow regime, with Mach number varying from 0.01 to 0.3.

Piziali [4] used a NACA 0015 finite wing of aspect ratio 10 and conducted experiments at various reduced pitch rates and angles of attack for a Reynolds number of 10^6 . A series of pressure transducers placed on the surface of the wing at various span-wise locations provided a comprehensive list of unsteady aerodynamic load measurements.

Tang and Dowell [6] used a NACA 0012 square wing oscillating in pitch and took measurements along three span-wise locations for various reduced pitch rates and angles of attack. The aspect ratio of their model was 1.5. Experiments were conducted below and above the static stall angle of the wing and used to identify the onset and evolution of the DSV.

Schreck and Helin [5] used a NACA 0015 profile on a wing of aspect ratio 2. The Reynolds number was 6.9×10^4 and pressure transducers were placed in eleven different span-wise locations. A ramping wing motion was employed for a variety of reduced ramp rates. They also carried out dye flow visualizations in a water tunnel in addition to providing detailed surface pressure measurements. However, it was Freymuth [10] the first to provide a visual representation of the DSV using titanium tetrachloride flow visualization in a wind tunnel and called the observed vortical structure the 'Omega Vortex', due to its shape.

Coton and Galbraith[8] used a NACA 0015 square wing of aspect ratio 3 in ramp-up, ramp-down and harmonic oscillation in pitch. A relatively high Reynolds number of 1.5×10^6 has been used for various angles of incidence and pitch rates. The DSV has been identified to form uniformly over the wing span, but shortly after the strong three dimensionality of the stall vortex in combination with the wing tip effects caused the DSV to distort to an 'Omega' shape.

Finally, the work undertaken by the Aerodynamics Laboratory of Marseilles (LABM) [9] employed an embedded Laser Doppler Anemometry (LDA) technique in order to provide detailed velocity measurements inside the boundary layer during DS and the experiment was designed to assist CFD practitioners with their efforts in turbulence modelling.

Amongst the plethora of 2D experimental investigations Wernert *et al.*[7] conducted a PIV study on a pitching NACA 0012 aerofoil for a mean angle of incidence of 15 degrees and oscillation of amplitude equal to 10 degrees. The wing had an aspect ratio of 2.8 and a reduced frequency of 0.15 has been used. The researchers used splitter plates on both ends of the wing to ensure 2D flow.

Based on the above summary and since this paper attempts to compare the 2D and 3D flow configurations during dynamic stall two experiments were selected for computations. In the absence of a 3D data set combining flow filed and surface pressure measurements, a combination of the PIV study of Wernert *et al.*[7] and the surface pressure survey experiment of Schreck and Helin [5] provides an adequate basis of comparison of the surface pressure loads on the maneuvering lifting surface and the velocity field and flow development around it. A summary of the flow conditions and measured quantities of all the above investigations is presented in Table 1.

In parallel to the experimental investigations, CFD studies have so far concentrated on 2D dynamic stall cases with the earliest efforts to simulate DS performed in the 1970s by Mc-Croskey et al.[1], Lorber and Carta [11] and Visbal [12]. Initially, compressibility effects were not taken into account due to the required CPU time for such calculations. However, in the late 1990s, the problem was revisited by many researchers [13, 14, 2] and issues like turbulence modelling and compressibility effects were assessed. Still, due to the lack of computing power and established CFD methods, most CFD work done until now focused on the validation of CFD codes rather than the understanding of the flow physics. Barakos and Drikakis [2] have assessed several turbulence models in their 2D study, stressing their importance in the realistic representation of the flow-field encountered during DS. More recently, the same researchers [15] presented results for a range of cases and have analysed the flow configuration in 2D. The only 3D CFD work done to date was by Ekaterinaris [13] who demonstrated that 3D computations are possible; comparison with experiments was very limited. Laminar 3D dynamic stall calculations were also presented by Newsome [16]. The present work, therefore, is to our knowledge the first systematic attempt to investigate the physics of the 3D DS phenomenon using CFD. Results are presented here for the cases by Wernert et al.[7] and Schreck and Helin [5] in order to highlight the differences between the 2D and 3D flow configurations.

Formulation

CFD solver

The CFD solver used for this study is the PMB code developed at the University of Glasgow [17]. The code is capable of solving flow conditions from inviscid to laminar to fully turbulent using the Reynolds Averaged Navier-Stokes (RANS) equations in three dimensions. The use of the RANS form of the equations allows for fully turbulent flow conditions to be calculated with an appropriate modelling of turbulence. Detached eddy simulation and large eddy simulation is also possible. The turbulence model used for this study has been the standard $k - \omega$ turbulence model [18] since for the selected cases turbulence is expected to have a secondary but significant role. To solve the RANS equations, a multi-block grid is generated around the wing geometry, and the equations are discretised using the cell-centred finite volume approach. Convective fluxes are discretised using Osher's upwind scheme and formal third order accuracy is achieved using a MUSCL interpolation technique and viscous fluxes are discretised using central differences. Boundary conditions are set using halo cells. The solution is marched implicitly in time using a secondorder scheme and the final system of algebraic equations is solved using a preconditioned Krylov subspace method.

Grid Generation

Meshing finite wings encounters a problem in the tip region as a single-block grid will (a) render flat tips topologically impossible and (b) lead to skewed cells in the case of rounded tips. To counter these problems, three different blocking strategies were implemented as shown in Figure 1. In a first attempt, shown in Figure 1(a), the tip end is formed by an array of collapsed cells resulting in a C-H single-block topology. Although this is adequate for thin, sharp tips it fails to satisfactorally represent the tip geometry of wings with thicker sections or flat tips. For wings with flat tips, like the ones used in this paper, good results can be obtained by using a true multiblock topology. As shown in Figure 1(b), the tip plane constitutes one of the six sides of a new block extending to the far-field. This topology is capable of describing both flat and rounded tips and has been found to be robust and accurate, with no collapsed cells in the vicinity of the tip region. A modification of this topology is shown in Figure 1(c) where 4 blocks were used next to the flat tip plane to promote cells with a better aspect ratio than in the previous case. Other approaches including H-H and C-O topologies have also been investigated. The latter is shown in Figure 1(d) and is suitable for truncated wings with rounded tips. In this case, the C- topology used around the leading edge curves around the tip resulting in a very smooth distribution of the radial mesh lines around the entire wing and in particular to the wing-tip interface, which is no longer treated as a block boundary. This blocking produces the smoothest mesh around the tip region as none of the emerging grid cells is skewed. Apart from the single-block C-H method all other topologies can be used for both rounded and flat wing tips. The details of all grids used in this study are presented in Table 2.

Results and Discussion

2D Dynamic Stall

The first target of the present work is to compare the 2D and 3D flowfields during dynamic stall and establish the differences between them. Starting with 2D cases, Figure 2 compares the flowfield measurements of Wernert et al.[7] along with the present CFD results. Three angles of attack were selected during the upstroke part of the oscillation cycle where the DSV is fully formed. The CFD calculations were at exactly the same conditions as the experiment, with a sinusoidal pitch on the blade of the form: $\alpha(t) = 15 - 10\cos(kt)$ at a reduced pitch rate of k = 0.15. The Re_c was 3.73×10^5 while the Mach number was set to M=0.1. The comparison between CFD and experiments is remarkably good, with the DSV predicted at almost the same position as in the measurements. The evolution of DS is similar to that described by previous authors [15]. A trailing edge vortex appears at high incidence angles and below the DSV a system of two secondary vortices is formed. Despite the lack of measurements of the surface pressure, the PIV study of Wernert et al.[7] provides the rare opportunity for validating the velocity field obtained against quantitative measurements. In this work velocity profiles were extracted at three chordwise stations corresponding to x/c = 0.25, 0.5and 0.75. With the exception of the work reported by Barakos & Drikakis in [9] this is the only other comparison of velocity profiles during DS appearing in the literature. As shown in Figure 3 the comparison between experiments and CFD is remarkably good at the lowest incidence angle (Figure 3(a)) and remains favourable even at higher incidence angles (Figure 3(b,c)). The experimentalists [7] have reported that at the angles of 23° and 24° the flowfield was no longer reproducible during the experiments, which explains the discepancies observed. The agreement is better closer to the wall while a constant shift appears towards the outter part of the boundary layer. Further comparisons of the turbulent flow quantities in this unsteady flow are not possible due to the lack of near-wall resolution of the PIV measurements.

3D Dynamic Stall - Grid and time convergence

et al.[5]. In contrast to the previous laminar study by Newsome [16] where rounded tips were used for a similar flow case, the present work preserves the real geometry of the wing using multi-block grids as explained in section 4.2. In Figures 4(a)

and 4(b), results from three different grids used are shown. The coarse grid is made out of 0.42 million cells, the medium 1.7 million cells and the fine 3.1 million cells. The medium grid has been considered as adequate following the C_L plots (Figure 4a) and was employed for the rest of the calculations. Even results of the 0.42 million cells grid are close to the ones obtained on finer meshes upto the near-stall angles. The reason for this is the highly impulsive nature of the flow which is predominantly driven by the dynamics of the fast moving surface.

A time-step sensitivity study was subsequently conducted by halving the original time-step (Figure 4(b)). The results of the two calculations were practically the same and therefore the original time-step was considered as adequate. This dimensionless time-step of 0.058 is adequate for resolving frequencies of about 20 Hz which is far higher than the range of oscillating frequencies employed during experiments.

The required CPU time for calculating the 2D and 3D flow cases is reported in Table 3. All calculations were performed on a Beowulf cluster with 2.5 GHz Pentium 4 nodes.

3D Dynamic Stall - Qualitative Comparison of the Flow field

Having established confidence in the employed grid and time steps used further computations were attempted. Figure 5 presents a comparison between experiments [5] and CFD flow visualisation for the fully formed omega-shaped vortex. The agreement in the overall shape is clear. The DS vortex is completely detached from the wing inboard and bends towards the surface of the wing outboard. An additional feature of the flow (not evident from the experimental flow visualisation) is the presence of the secondary vortices below the omega-shaped vortex. Furthermore the tip and the omega-shaped vortices start from the same point at the wing-tip. A set of snap-shots from the CFD calculations is presented in Figure 6. In this Figure the cores of the vortices are extracted from the CFD solutions using the vortex core detection toolbox in $FieldView^{TM}$ and are tracked in time. In addition, particles were used to highlight the size of the vortices and their sense of rotation. The phenomenon starts inboards with the formation of a vortex at the leading edge which is subsequently detached from the wing and grows in size. The growth is less as one moves towards the tip of the wing (Figure 6(a)) and the core of the vortex bends upstream towards the leading edge of the wing tip (Figure 6(b)). Further on during the cycle one cannot fail to notice that on the mid-span of the wing (Figure 6(c)) the flow looks like the 2D cases of Figure 2. However, as the DSV is formed, A second set of calculations simulated the experiment of Schreck the core of the vortex stays bound to the LE region of the wingtip while the main part of the DSV is convected downstream. As the DSV grows in size and its core moves above the surface of the wing, the omega-shape appears due to the fact that near the wing-tip the vortex is still bound. The phenomenon

becomes more and more interesting as the tip vortex is formed leading to a Π - Ω vortex configuration which is a combination of the two well-established vortical systems: the horse-shoe vortex and the dynamic stall vortex. The flow near the LE of the wing tip appears to split in two streams and is directed either towards the tip-vortex or the dynamic stall vortex. Apart from the main vortices all secondary vortices appearing during 2D dynamic stall are present in the 3D case. Interestingly, the secondary vortices formed below the DSV also appear to take the same omega shape and bend at the LE of the wing tip.

3D Dynamic Stall - Surface pressure history, effect of ramp rate

Further comparisons against measurements are presented in Figure 7 where Cp contours on the upperside of the wing are plotted. Measurements are available for only a fraction of the wing area, bounded by a solid box on the CFD plots. Overall, the shape and level of the contours corresponds with the measured data with the agreement getting better at higher incidence angles. The reason for any minor discrepancies towards the mid-span of the wing lies in the fact that the experiment used a splitter plate on the wing root with surface qualities that do not exactly match the idealisations made by either symmetry or viscous boundary conditions. The size of the plate is comparable with the DSV vortex size (the splitter plate diameter was equal to two cord lengths) and thus the effectiveness of the plate may not be good especially at high incidence angles. Further calculations were performed for different ramp rates and the pivot point of the wing was also changed from x/c = 0.33 to x/c = 0.25. The summary of the computed cases is presented in Table 4. The comparison between CFD and experiments for the surface pressure on the wing is shown in Figures 7, 8, 9. One can see that at the higher ramp rate the comparison between CFD and experiments is better since the character of the flow is more impulsive and driven predominantly by the imposed motion of the wing. Near the middle of the wing the footprint of each vortex is clearly visible (see Figures 8(b) and 9(b)) while closer to the wing tip, the surface pressure alone is not adequate for deducing conclusions for the flow configuration. For all cases the agreement between CFD and experiments is better when the splitter plate is modelled. Disparities in spatial resolution impacted agreement between the computed and measured data, as well. To further assist a quantitative comparison beween experiments and CFD results the Cp distribution at three spanwise stations (z/c = .5, 1.0, 1.6) and for two incidence angles (30 and 40) degrees) is extracted and the comparison is presented in Figures 10 and 11. The footprint of the DSV can be seen in all stations while the evolution of dynamic stall appears to be faster in the inboard stations and delayed near the tip. This can be seen from the comparison of the second peak of the Cp distributions which, at an incidence of 30°, is located between

x/c = 0.3 and x/c = 0.4 inboards (Figures 10(a) and (b)) while it just appears between x/c = 0.1 and x/c = 0.2 at the outboard station (Figure 10(c)). Finally, Figure 12 shows the Cp distributions for all experimental spanwise stations for the case 3 of Table 4 at an incidence of 40.3° . The location of the DSV in the experiments and CFD is identical near the wing root and as the tip is approached, the DSV in the CFD solution appears to be slightly aft in comparison with the experiment. The authors believe that this is a turbulence model issue and is a subject for further investigation.

2D/3D Dynamic Stall - Integral Loads

The ability to predict the integral loads of the wing during the unsteady manouevre is paramount for design. CFD results for the C_L , C_D And C_M coefficients are presented in Figure 13. For the sake of comparison 2D calculations have also been performed at the same conditions. As can be seen, results at higher ramp rate indicate a more impulsive behaviour and delayed stall in the 2D case. Overall the 3D calculations reveal a smoother variation of the integral loads with a more gradual stall in comparison to the 2D results. This is a direct effect of the interaction between the tip and the DS vortices. As the incidence increases the strength of the tip vortex also increases creating a second suction peak near the tip in addition to the suction created by the DS vortex, this has a strong effect especially for the moment and drag coefficients and this highlights the problem engineers have to face when scaling 2D measurements for use in 3D aerodynamic models.

Concluding Remarks

Numerical simulation of the 3D dynamic stall phenomenon has been undertaken and results have been compared against experimental data and 2D calculations. For all cases, CFD results compared favourably against experiments. The 3D structure of the DSV was revealed and was found to agree well with the only flow visualisation study available. The evolution of the 3D DS phenomenon was also presented. The main conclusion of this work is that similarity between 2D and 3D calculations is good only in the mid-span area of the wing while the outboard section is dominated by the omega-shaped vortex. The flow configuration near the wing tip is far more complex with the tip vortex and the DSV merged towards the wing tip. From this study it is evident that further experimental and numerical investigations of this complex flow phenomenon are necessary. In particular, combined efforts with well controlled experiments and measurements of both surface and boundary layer properties are essential to evaluate the predictive capabilities of CFD for unsteady, separated flows. This work is part of a wider effort undertaken by the authors in understanding, predicting and controlling unsteady aerodynamic flows.



Figure 1: Grid topologies employed for calculations: (a) 'collapsed' tip, (b,c,d) 'extruded' tips.



(c) AoA=24 deg upwards

Figure 2: Comparison between CFD (right) and experiments (left) by Wernert *et al.* for the flow field at: (a) 22° upstroke, (b) 23° upstroke and (c) 24° upstroke. The streamlines have been superimposed on colour maps of velocity magnitude and for the experimental cases, are based on PIV velocity data.



Figure 3: Comparison between CFD (solid line) and experiments by Wernert *et al.*(dashed line): for the streamwise velocity profile at three stations (x/c=0.25, 0.5, 0.75) along the aerofoil chord. (a) 22° upstroke, (b) 23° upstroke and (c) 24° upstroke.



Figure 4: (a) Grid and (b) time convergence studies for the ramping NACA0015 wing. Flow conditions correspond to case 1 of Table 4.



(b)

Figure 5: The 'Omega' vortex as shown from the visualisations performed (a) by Schreck & Helin [5] and (b,c) the CFD representation of the same structure (right). Flow conditions correspond to case 1 of Table 4.





(b) 20 deg



(d) 28 deg

Figure 6: Vortex cores (left) and streamtraces (right) for case 1 of Table 4. (a) 13° , (b) 20° , (c) 25° and (d) 28° .



Figure 7: Comparison between experiments by Schreck & Helin [5] and CFD results for the surface coefficient distribution on the suction side of the square NACA-0015 wing. Flow conditions correspond to case 1 of Table 4. (a) $\alpha = 30.0^{\circ}$ and (b) $\alpha = 40.9^{\circ}$. From top to bottom: CFD with splitter plate as symmetry plane, CFD with splitter plate as viscous wall and experimental values.



Figure 8: Comparison between experiments by Schreck & Helin [5] and CFD results for the surface coefficient distribution on the suction side of the square NACA-0015 wing. Flow conditions correspond to case 2 of Table 4. (a) $\alpha = 30.2^{\circ}$ and (b) $\alpha = 39.9^{\circ}$. From top to bottom: CFD with splitter plate as symmetry plane, CFD with splitter plate as viscous wall and experimental values.



Figure 9: Comparison between experiments by Schreck & Helin [5] and CFD results for the surface coefficient distribution on the suction side of the square NACA-0015 wing. Flow conditions correspond to case 3 of Table 4. (a) $\alpha = 29.5^{\circ}$ and (b) $\alpha = 40.3^{\circ}$. From top to bottom: CFD with splitter plate as symmetry plane, CFD with splitter plate as viscous wall and experimental values.



Figure 10: Comparison between experiments and simulation for the surface pressure coefficient distribution at an incidence angle of 30° . Three spanwise stations were considered (z/c = 0.5(top), 1.0(middle), 1.6(bottom)) and the flow conditions correspond to cases 1 (left) and 3 (right) respectively of Table 4.



Figure 11: Comparison between experiments and simulation for the surface pressure coefficient distribution at an incidence angle of 40° . Three spanwise stations were considered (z/c = 0.5(top), 1.0(middle), 1.6(bottom)) and the flow conditions correspond to cases 1 (left) and 3 (right) respectively of Table 4.



Figure 12: Comparison between experiments and simulation for the surface pressure coefficient distribution at an incidence angle of 40.3°. All experimental spanwise stations were considered: z/c = 0.0(a), 0.1(b), 0.2(c), 0.3(d), 0.5(e), 0.75(f), 1.0(g), 1.25(h), 1.4(i), 1.5(j) and 1.6(k) and the flow conditions correspond to cases 5 of Table 4. The splitter plate has been modelled as a viscous wall.



Figure 13: Comparison between 2D and 3D simulation results for the lift, drag and quarter-chord moment coefficient for cases 1 (a) and 3 (b) of Table 4. (a) $\alpha^+ = 0.1$ and (b) $\alpha^+ = 0.2$.

Conditions	Measurements	
Ramping motion	Surface pressure	
$Re = 6.9 \times 10^4, M = 0.03$	Flow visualisation (dye injection)	
NACA0015, AR=2		
Ramping and oscillatory motion	Surface pressure	
$Re = 2.0 \times 10^6, M = 0.278$	Flow visualisation (micro-tufts)	
NACA0015, AR=10		
Ramping and oscillatory motion	Surface pressure	
$Re = 1.5 \times 10^{6}, M = 0.1$		
NACA0015, AR=3		
Oscillatory motion	Surface pressure	
$Re = 0.52 \times 10^6, M \ 0.1$		
Oscillatory motion	Boundary layers	
$Re = 3 - 6 \times 10^6, M = 0.01 - 0.3$	Velocity profiles	
NACA0012	Turbulence quantities	
Oscillatory motion	PIV & LSV	
$Re = 3.73 \times 10^5, M \ 0.1$		
NACA0012, AR=2.8		
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Table 1. Summary of validation cases for dynamic stall

The first five cases concentrate on 3D dynamic stall while the last one employed PIV for the study of the 2D configuration.

Table 2. Details of the employed CFD grids.

Grid	Blocks	Points on wing	Points on tip	Farfield	Wall distance	Topology
1	13	6222	820	8 chords	10^{-4} chords	3D C extruded
2	20	7100	900	8 chords	10^{-4} chords	3D C-extruded double
3	44	8400	900	8 chords	10^{-4} chords	3D C-extruded double
4	6	240	n/a	8 chords	$10^{-5} {\rm ~chords}$	2D C-type

Table 3. Details of the	CPU time r	equired for	calculations.
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Grid	Size (nodes)	No of processors	CPU time (s)
1	420,000	1	$9.2 imes 10^5$
2	729,000	8	1.12×10^5
3	1,728,000	8	$4 imes 10^5$
4	28,800	1	$6.8 imes 10^4$

All calculations were performed on a Linux Beowulf cluster with 2.5GHz Pentium 4 nodes.

Table 4. Summary of conditions for CFD calculations.

Case	$2D \setminus 3D$	α^+	$(x/c)_{rot}$	Re	М	Motion
1	2D&3D	0.10	0.33	6.9×10^{-4}	0.2	ramping 0-60
2	2D&3D	0.10	0.25	$6.9 imes 10^{-4}$	0.2	ramping 0-60
3	2D&3D	0.20	0.25	$6.9 imes 10^{-4}$	0.2	ramping 0-60

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