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LES simulation and experimental validation of the unsteady aerodynamics of blunt wind turbine airfoils

Guofu Wang¹, Lei Zhang^{2*}, Wen Zhong Shen³

¹ Liao Ning Shihua University, Fushun 113001, China

² Institute of Engineering Thermophysics, Chinese Academy of Sciences, Beijing 100190, China

³ Department of Wind Energy, Technical University of Denmark, 2800 Lyngby, Denmark

* Correspondence: zhanglei@iet.cn; Tel.: +86-10-82543132

Abstract:

In order to investigate the unsteady performance of blunt wind turbine airfoils caused by boundary layer separation and wake eddies, this paper studies the aerodynamic performance by large eddy simulation (LES) and wind tunnel experiment at a Reynolds number of 2.62×10^5 . The blunt airfoils are obtained by enlarging the trailing edge of the DU 91-W2-250 airfoil to 6% and 10% chords symmetrically on both pressure and suction sides of the airfoil. The simulation was carried out with the incompressible finite-volume Navier-Stokes code EllipSys3D; and, the experiment was done in a wind tunnel with a cross-section of $0.5\text{m} \times 0.5\text{m}$ by measuring the surface pressure and wake velocities using ESP-64HD pressure scanner and TSI hot-wire anemometer. The unsteady wake was captured by hot-wire in the wind tunnel, and LES with EllipSys3D. Both experiment and LES show that the spectrum of aerodynamic forces has a broadband nature which is in coincidence with the wake eddies, implying that the unsteady Kármán vortex sheet is the driving mechanism of the force fluctuation. Moreover, the trailing edge size affects the separation bubbles and transition process in the boundary layer. It shows that the boundary layer near the leading edge is unstable in the spanwise direction, which is characterized by low frequency waves.

24 **Keywords:** blunt airfoil; wake; unsteady; transition; large eddy simulation; experiment

25 Nomenclature

26 C_d = drag coefficient of airfoil

27 C_l = lift coefficient of airfoil

28 C_p = pressure coefficient

29 Re = Reynolds number

30 c = chord length

31 d = reference length

32 f = frequency

33 h' = wake width

34 $\bar{p}$ = average pressure

35 s = spanwise length of airfoil model

36 t = time

37 $\bar{u}_i$ or $\bar{u}_j$ = time-averaged velocity component

38 u_s = velocity at separation

39 u_∞ = free stream velocity

40 u = velocity in x direction

41 v = velocity in y direction

42 w = velocity in z direction

43 x = spanwise coordinate

44 x_i or x_j = coordinates in the NS equations, typically referring to x , y , and z in Cartesian

45 coordinates

46 y = cross stream coordinate

47 z = streamwise coordinate

48 α = angle of attack

49 ρ = density of fluid

50 ν = kinematic viscosity of fluid

51 τ_{ij} = shear stress

52 1. Introduction

53 With the development of wind energy industry, the size of wind turbine rotors increases
 54 gradually and the maximum capacity of a single commercial wind turbine has gone up to 8 MW^[1].
 55 Large scale rotor blades should be manufactured strongly enough to bear both aerodynamic and
 56 structural loads. Blunt-trailing-edge airfoils have potential advantages due to their larger sectional
 57 area and high lift coefficient, especially when they are used at the inboard section of blade^[2,3].
 58 However, the shedding vortices in the wake may cause a base drag, unsteadiness and vortex
 59 shedding noise^[4, 5]. The present study mainly focused on the unsteady aerodynamics due to the
 60 vortex street in the wake, which was primarily studied in aeronautic wing applications^[6]. In 1987,
 61 Law and Gregorek evaluated a truncated airfoil for the application on large wind turbines and
 62 concluded that the truncated airfoil had a higher maximum lift, higher lift curve slope, lower drag
 63 at higher lift coefficient, and higher chordwise force coefficient than sharp-trailing-edge airfoils
 64 with a similar thickness^[7]. Additionally, Sato and Sunada (1995) found that the maximum lift-to-
 65 drag ratio could be increased by cutting off the trailing edge of airfoils^[8], while Iwasaki et al. (1997)
 66 provided further verification on the lift curve slope which is higher than both the original airfoil
 67 and sharp airfoils with the same thickness^[9]. In 2003, Standish and van Dam investigated the
 68 aerodynamic performance of blunt-trailing-edge airfoils and indicated that the lower adverse
 69 pressure gradient could decrease the sensitivity to premature boundary-layer transition, i.e.

insensitive to leading edge contamination^[10]. Following those topics, Baker et. al. conducted experimental studies and verified that blunt-trailing-edge airfoils were insensitive to leading edge contaminations and the drag increase was caused by the base drag in the wake^[11]. The detailed researches on blunt airfoils and blades by van Dan team (at University of California Davis) were referred to references [12-18]. Moreover, similar work was done by T. Zhang et al. based on computational fluid dynamics method^[19].

On the blunt airfoils for wind turbine application, the research topics are strongly diversity, including such as aerodynamic performance, simulation methods, design methods, application on blades, and so on. In 2006, Christopher et. al. studied the performance of blunt airfoils with computational fluid dynamics and showed that the DES method did not predict the transition and separation along the airfoil surface while the HRLES method did^[20]. The authors investigated the aerodynamic performance of blunt airfoils affected by vortex generators^[21]. As for design methods, in 2003, Standish and van Dam found that symmetrically adding thickness along the baseline airfoils is better than the cutting off method^[10], which was also proofed by Murcia and Pinilla in 2011^[22]. Moreover, the optimization of blunt airfoil shape using a genetic algorithm should produce even better airfoils meeting specific objectives^[23]. Except researching the aerodynamic performance of airfoils, the influence of blunt airfoils on rotor blade was studied by Lee et. al. based on the benchmark blade, i.e., NREL Phase VI blade^[24]. Additionally, it has been studied that flatback, i.e., blunt, design does not impede rotational augmentation of aerodynamic performance on rotor blades^[25]. The results showed that the blunt trailing edge airfoil helps to improve both structural and aerodynamic performance of the blade. Conclusively, in wind energy applications studies were mainly focused on the steady aerodynamics and the overall characteristics.

Due to shedding vortices from the blunt trailing edge, the aerodynamic performance of blunt airfoils is unsteady. The shedding vortices adversely increase the base drag and lead to unfavorable noise and aeroelastic problems^[26]. In 2011, the authors primarily studied the unsteady characteristics of blunt airfoils and described the unsteady characteristics of lift and drag

coefficients of blunt airfoils^[27]. Similar conclusions were put up by Xu et. al.^[28]. In 2015, Bangga et. al. investigated the unsteady characteristics of flow over thick blunt airfoils and suggested a new definition of Strouhal number^[29]. Because the rotor operates in unsteady conditions and the fatigue loads are caused by unsteady forces, more efforts should be paid on unsteady aerodynamics of blunt airfoils and blunt blades, which is the motivation of the present study.

In this paper, the unsteady aerodynamics and boundary layer instability of blunt airfoils were studied by three dimensional LES based on the CFD solver EllipSys3D^[30]. Meanwhile, measurements with ESP-64HD pressure scanner and TSI hot-wire anemometer were carried out to compare and verify the numerical method and shedding frequency of vortex sheet in the wake.

2. Methods

2.1 Large eddy simulation

Due to the blunt trailing edge, the vortex shedding downstream and in the boundary layer is expected to dominate the flow characteristics. Since Reynolds Averaged Navier-Stokes (RANS) simulation is not accurate enough to capture broad scales of eddies in the flow field, large eddy simulation is applied on this research issue. These simulations are based on a CFD solver, EllipSys3D, developed at Wind Energy Department of Technical University of Denmark. For more details about the numerical techniques, the reader is referred to references [30-33].

The code is based on a multi-block/cell-centered finite volume discretization of steady/unsteady incompressible Navier-Stokes equations^[34-36] in primitive variables (pressure and velocity). In this code, the momentum equations are discretized by a second-order backward-differentiation scheme in time and second-order central differences in space, while the convective terms are discretized by the QUICK upwind scheme^[37]. The code is programmed using the multi-block topology; therefore, it is easily parallelized using message-passing interface (MPI). The filtered incompressible Navier-Stokes equations, turbulent stresses and eddy viscosity equations are solved to obtain the turbulent flow data.

$$\frac{\partial \bar{u}_i}{\partial x_i} = 0 \quad (1)$$

$$\frac{\partial \bar{u}_i}{\partial t} + \bar{u}_j \frac{\partial \bar{u}_i}{\partial x_j} = -\frac{1}{\rho} \frac{\partial \bar{p}}{\partial x_i} + \frac{\partial}{\partial x_j} \left(\nu \frac{\partial \bar{u}_i}{\partial x_j} \right) + \frac{1}{\rho} \frac{\partial \tau_{ij}}{\partial x_j} \quad (2)$$

where $\tau_{ij} = \bar{u}_i \bar{u}_j - \overline{u_i u_j}$ is the sub-grid scale (SGS) stress tensor. The SGS turbulence model used in this paper is the mixed-scale turbulence model^[37].

Figure 1 Profiles of airfoil models

The baseline airfoil examined in this study is DU 91-W2-250 with a maximum thickness to chord ratio of 25%^[38]. The trailing edge of this airfoil is enlarged from the original thickness of 0.65%*c* to 6%*c* and 10%*c* by symmetrically adding thickness on both sides. They are denoted as DU 91-W2-250_6 and DU 91-W2-250_10 respectively, as shown in Figure 1.

The computational domain is shown in Figure 2, which is the same as the wind tunnel testing section to verify LES results against measurements directly. A multi-block structured conformal mesh is used in the simulations with 60 blocks of 64×64×64. In the flow region with airfoil, the O-mesh topology is employed with 512 grid points on the airfoil, i.e., the averaged spacing in the chordwise direction equaling 0.0041*c*. The spanwise grid number is 128 based on previous experiences^[37], and the averaged spacing in spanwise equals 0.026*c*. The maximum Equiangle Skew of grid cells is 0.57 which appears at the trailing edge. The smallest wall normal distance is 10⁻⁵ so that *y*⁺ is less than 1 at a Reynolds number of 3×10⁵.

In simulations, velocity inlet with no perturbations and outlet boundary conditions are applied, as well as the slip wall boundary condition on the tunnel walls. The simulations were

conducted in dimensional system and the inflow velocity was set as 30 m/s. The simulation period lasted at least 5 s with a time step of 0.0001 s. Instantaneous flow fields are saved synchronically with the averaged flow field; and moreover, typical time history data are recorded ahead the leading edge, as well as inside the boundary layer on the suction surface and in the wake.

Figure 2 Block topology of the flow domain (the mesh is shown with every 4 grid points)

2.2 Experiment setup

The experiment was carried out in the IET (Institute of Engineering Thermophysics, Chinese Academy of Sciences) wind tunnel with a cross-sectional dimension of 0.5 m × 0.5 m and a tunnel length of 2.0 m. The setup of airfoil model in this wind tunnel is shown in Figure 3. The ceiling and bottom walls of the test section are parallel but the side walls are tapered with an angle of 0.8 degrees to compensate for boundary layer growth. The wind tunnel was calibrated before measurements using a hot-wire anemometer and other necessary apparatus. The slope of static pressure drop and turbulence intensities depending on wind speeds are shown in Table 1. The turbulence intensity at 30 m/s is less than 1.6% and the static pressure coefficient drop is less than 1.8% per meter along the axis of wind tunnel.

Figure 3 Setup of an airfoil model in the IET low speed wind tunnel

Table 1 parameters of the wind tunnel

Wind speed (m/s)	8.4	21.8	34.8
Slope of static pressure coefficient drop (1/m)	-	0.0180	0.0134
Turbulence intensities (%)	0.133	0.141	0.156

The airfoil model was mounted vertically bisecting the test section on a turntable driven by a stepping motor. The angle of attack can be controlled with a precision of ± 0.05 degrees. The spanwise width of the model is 0.496 m, leaving a gap of 0.004 m from the ceiling wall. The chord length of the test models is 0.15 m producing a minimum solid blockage of 7.5%. As an example, reliable results were obtained with the minimum blockage of 8.3% in the similar test [39]. Because the same airfoil models with tunnel walls were applied in LES, the obtained data can be compared directly without corrections of tunnel walls and blockage.

The experiment was carried out at a Reynolds number of $Re = 2.62 \times 10^5$. The corresponding velocity is 30 m/s, measured by Pitot tubes at 0.5 m upstream from the turning axes of the testing model. Pressure taps (48~50 depending on specific models) were mounted in the middle section of testing models, whose distribution was pre-optimized based on the pressure distribution calculated by XFOIL [40]. The maximum measuring range of pressure sensors is ± 1.0 PSI with a tolerance of $\pm 0.05\%$ FS. On the other hand, the hot-wire sensor is calibrated before the measurement with an error of less than $\pm 0.3\%$.

3. Validation and comparison of the two methods

The experimental data should be verified itself and then used for validation of numerical simulations. First, the repeated measurements are compared for the DU91-W2-250 and DU91-W2-250_6 at an angle of attack of 0° . Second, the lift and drag coefficients are compared between numerical simulations and measurements; and, the effect of spanwise grid density on drag coefficient is also examined in this part. Third, pressure distributions of all the three airfoil models are compared. Finally, the simulated frequency spectrum of velocity in the wake is compared to measurement; and meanwhile, the grid stretching effect on spectral characteristics is discussed.

3.1 Validation of experiment by repeats

The distributions of surface pressure are compared in Figure 4 a) and b) where the repeated measurements were carried out at an airspeed of 20 m/s. Different tests are numbered as “test1”, “test2”, and “test3” in the figure. It shows that the pressure distributions basically match each other with only very small differences. The differences are less than 3.5% for the DU 91-W2-250 airfoil. A similar validation is carried out for the DU 91-W2-250_6 airfoil, showing that maximum differences are less than 2.2%.

Figure 4 Comparison of pressure distribution from the repeated measurements. a) Pressure distribution of DU 91-W2-250; b) Difference between tests and their average values

3.2 Validation of LES

194 3.2.1 Comparison of lift and drag coefficients

197 **Figure 5 Comparisons of force coefficients at $Re=2.62 \times 10^5$. a) Lift coefficient of DU 91-W2-250, b) Drag**
 198 **coefficient of DU 91-W2-250.**

199 Figure 5 shows the comparisons of lift and drag coefficients of DU 91-W2-250 predicted by LES
 200 and measured by pressure scanner. It indicates that the lift from LES is in better agreement with
 201 that from experiment than drag before stall. When the airfoil undergoes stall, the present LES setup
 202 overestimates the lift. Drag coefficient in simulation is strongly affected by spanwise grid density,
 203 but the lift coefficient varies slightly with less difference. Refinement of the spanwise grid is
 204 necessary if the drag simulation is the key concern^[37]. To test the influence, the spanwise grid
 205 number was refined by two times and three times for the airfoil model DU 91-W2-250. It was found
 206 that the drag coefficient firstly dropped to 0.0304 (two-times refined) and then to 0.0154 (three-times
 207 refined), much close to the experimental value of 0.017.

208 3.2.2 Comparison of pressure distributions

209 Figure 6 shows comparisons of time-averaged surface pressure distributions at the mid-span of
 210 airfoils at an angle of attack of 10° . The pressure distributions on the pressure side match together
 211 as well as the suction peaks. And, LES captures the pressure platform and the followed pressure
 212 drop on both sides of airfoil surface. In addition, the pressure distribution at other angles of attack
 213 is also compared and the difference is that the area enclosed by pressure curves from LES is slightly

larger than that from experiment especially at large angles of attack, implying that force coefficients are slightly larger than that from experiment.

Figure 6 Comparison of time-averaged pressure distribution at $\alpha = 10^\circ$ and $Re = 2.62 \times 10^5$. a) DU 91-W2-250, b) DU 91-W2-250_6, c) DU 91-W2-250_10

3.2.3 Comparison of velocity frequency in the wake

In numerical simulations, the grid stretching has an important effect on the maximum calculated frequency because a coarse grid will smooth out small-scale vortices. Figure 7 shows the effect of grid stretching on the highest frequency that can be captured by the mesh. The spectra are smoothed by a filter to clearly show the trend. The circles in the figure show the spectral energy near the leading edge where a fine mesh is used. The stars show the spectral energy in the wake where a coarse mesh is used. It shows that in the boundary layer with a fine mesh we can capture small scale vortices in the inertial sub-range. The highest frequency that can be distinguished is as high as 5000 Hz. On the other hand, the coarse mesh in the wake can only capture vortices with frequencies no more than 600 Hz.

In the experiment, the velocity frequency in the wake was acquired by a hot-wire probe with a sample frequency of 1000 Hz. The measured peak value of velocity frequency in the wake of DU 91-W2-250_6 is about 439.5 Hz, while it is 308.4 Hz for DU 91-W2-250_10. The simulated values in LES are 440.1 Hz and 323.2 Hz with errors of 0.1% and 4.7%, respectively. As an example, the comparison of velocity spectra of DU 91-W2-250_10 at $\alpha=0^\circ$ is shown in Figure 8. It shows that two

234 methods get the similar spectrum characteristics of velocity in the wake and the bands of shedding
 235 frequencies are basically in good agreement.

236

237

Figure 7 Effect of grid density on spectral energy of velocities

238

239

Figure 8 Comparison of velocity spectra of DU 91-W2-250_10 at $\alpha=0^\circ$, a) Measurement, b) LES

240

4. Results and discussion

The main objective of this study is to examine the unsteady dynamics of blunt airfoils by LES and experimental methods because of the lack of data in literature. The first issue is the spectrum characteristics of forces and velocities. Time signals of force coefficients are analyzed by spectral analysis to overlook the unsteady characteristics of blunt airfoils. Then the dominant frequencies at different angles of attack are compared. The relationship between the vibration of velocities and the fluctuation of airfoil forces is illustrated by comparing their spectra. The second issue is about the instability of boundary layer on blunt airfoils. Boundary layer separation bubbles are examined as well as the influence of trailing edge thickness and angle of attack on them. By comparing the velocity frequency in the boundary layer the development of T-S waves is illustrated. Additionally, the spanwise modes of velocity are analyzed at the end of this issue.

4.1 Spectral analysis

4.1.1 Fourier analysis of lift coefficients

Large eddy simulations get vibrating forces, as well as velocities in the flow field, containing many small fluctuations, which are different from URANS simulations containing a single frequency fluctuation^[27 28]. The dominant frequency can be worked out by applying the fast Fourier transform (FFT) on time signals. Taking the lift signal of DU 91-W2-250_6 at $Re=2.62 \times 10^5$ and $\alpha=0^\circ$ as an example, the time signal shows no regulation but the lift spectrum shows obviously the dominant frequency appearing at about 451 Hz, as shown in Figure 9. By a similar analysis, the dominant frequency of DU 91-W2-250_10 at the same condition is 308 Hz. The peak frequency observed for the lift coefficient is relative with the vortex shedding in the wake because of the absence of external perturbation. However, no characteristic frequency is captured for DU 91-W2-250 because the trailing thickness is small and the mesh in the wake region is not fine enough.

Figure 9 Fourier analysis on the lift coefficient of DU 91-W2-250_6 at $Re=2.62 \times 10^5$ and $\alpha=0^\circ$.

Specific characteristic frequencies of the lift coefficient at various angles of attack are shown in Table 2. The dominant frequencies for DU 91-W2-250_6 and DU 91-W2-250_10 before stall are about 450 Hz and 330 Hz, respectively. After stall, there are large separated vortices from the airfoil body and the characteristic frequency decreases a lot, which is in agreement of the previous researches^[28].

Table 2 Characteristic frequency of C_l at different angles of attack and $Re=2.62 \times 10^5$

Angles of attack (deg)	0	5	10	15	20
DU 91-W2-250_6	451	447	457	-	-
DU 91-W2-250_10	308	336	340	30	31

4.1.2 Effects of vortex shedding in the wake on unsteadiness of aerodynamic forces

Eddies shed from the trailing ledge of a blunt airfoil, induce vibrations on aerodynamic forces. Figure 10 shows the spectra of the y -direction velocity at different locations in the wake of the DU 91-W2-250_6 airfoil at $\alpha=0^\circ$. The peak frequency of the velocity is about 440 Hz, smaller than the peak frequency of the corresponding lift coefficient as shown in Table 2 and Figure 10. The reason is

that the lift contains all the effects of three-dimensional vortices in the wake, but the velocity spectrum represents only one point at middle span in the wake. The similar phenomenon was found by Huang and Lin, whose results showed that the frequency of velocity varies with the span location in the wake^[42]. Therefore, the difference between the velocity and lift frequencies is reasonable. As the monitoring point moves downstream, the amplitude of vibration decreases and the peak frequency disappears in the far wake region. The turbulence dissipation plays an important part in decrease of vortices, as well as the cut-off effect of the coarse grid in the far wake region.

Further examinations on another blunt airfoil DU 91-W2-250_10 show that large shedding vortices spread farther downstream than small vortices in the wake. The peak frequency can also be found at $3.4 c$. The peak frequency of the vortex shedding at $\alpha=0^\circ$ is about 313 Hz which is slightly larger than that of the lift coefficient in Table 2, providing an additional evidence that the frequency of velocity in wake doesn't equal to that of the lift coefficient. Furthermore, it can be found that the peak frequency of shedding vortices in the wake of DU 91-W2-250_10 is smaller than that of DU 91-W2-250_6 due to the increase in trailing edge thickness. Therefore, the dominant frequency of the lift coefficient of DU 91-W2-250_10 is correspondingly smaller than that of DU 91-W2-250_6.

Figure 10 Velocity spectra in the wake of the airfoil model DU 91-W2-250_6 at $Re=2.26 \times 10^5$ and $\alpha=0^\circ$, a) at 0.08c, b) at 0.7c, c) at 1.3c, and d) at 2.3c downstream from the trailing edge.

The above results show that the fluctuation of aerodynamic forces depends on the vortex shedding in the wake region but the peak frequencies may be slightly different between forces and velocities at a specific point downstream.

4.2 Boundary layer instability

4.2.1 Boundary layer separation

In Figure 11 the instantaneous streamlines at mid span of airfoil models are plotted at two different angles of attack, i.e., $\alpha=0^\circ$ and $\alpha=10^\circ$. Because the Reynolds number is low, the boundary layer on airfoil models separates even at $\alpha=0^\circ$. For the DU 91-W2-250 airfoil, small separation

bubbles are produced in the boundary layer on both sides near the mid chord position. The separation bubble on the suction surface approaches the leading edge when increasing the angle of

Figure 11 Instantaneous flow field at middle span shown by streamlines at $Re=2.62 \times 10^5$

attack, while it moves towards the trailing edge on the pressure surface. At large angles of attack the entire boundary layer separates from the leading edge on the suction surface indicating that the airfoil is undergoing stall. Compared with instantaneous results, the time-averaged flow structures are shown in Figure 12, which are much more different from the instantaneous flow field. The transient discrete shedding vortices in the boundary layer merge together into a long and thin

311 separation bubble in the averaged flow field, and it exists at all angles of attack simulated. The
 312 separation flow reattaches to the surface of airfoil again when the laminar-turbulent transition
 313 manages to increase the momentum into the boundary layer that overcomes the adverse pressure
 314 gradient.

Figure 12 Time-averaged flow field at mid span shown by streamlines at $Re=2.62 \times 10^5$

315
 316 The trailing edge thickness strongly affects the flow structure and correspondingly the
 317 aerodynamic performance. For DU 91-W2-250_6 shown in Figure 11, the boundary layer flow is
 318

more stable as compared to the original airfoil DU 91-W2-250. Enlarging the trailing edge thickness reduces the slope of airfoil profile at both sides and then helps to reduce the adverse pressure gradient^[4, 7, 11]. Therefore, eddies in the boundary layer of DU 91-W2-250_6 are smaller than that of DU 91-W2-250 at the same angle of attack. For blunt airfoils at small angles of attack, e.g., $\alpha=0^\circ$, a Kármán vortex shedding is caused by flow separation at the blunt trailing edge. This separation is gradually merged into the boundary layer separation with the increase of angle of attack until the airfoil stalls when a large boundary layer separation dominates the flow in the wake. Different from the transient flow field, the averaged flow field in Figure 12 shows that there is no obvious separation at $\alpha=0^\circ$ and at $\alpha=10^\circ$ on the suction surface of DU 91-W2-250_6.

As for the airfoil model DU 91-W2-250_10, the instantaneous flow structure in Figure 11 shows that separation bubbles return to the boundary layer at $\alpha=0^\circ$ as compared to DU 91-W2-250_6 although the trailing edge thickness of DU 91-W2-250_10 is larger. The averaged flow field in Figure 12 shows the same phenomenon with the instantaneous flow pattern for this airfoil model. It shows that too large trailing edge thickness may play a negative role which brings the separation bubble back to the boundary layer.

For higher angles of attack, as shown in Figure 13, the flow pattern in instantaneous flow field is significantly different from in time-averaged flow field. The averaging smoothes out small separation eddies or merges them into a thin separation bubble. Even after stall the separation vortex seems smaller than in the instantaneous flow pattern.

Figure 13 Flow pattern at a high angle of attack of 20°, DU 91-W2-250_6 at $Re=10^5$

4.2.2 Spectrum characteristics of the velocity in boundary layer

Figure 14 Spectra of velocity components at 0.24c ahead of the leading edge, DU 91-W2-250_6

Figure 15 Spectra of y -velocities in the boundary layer, DU 91-W2-250_6

Ahead of the leading edge, the fluctuation of velocities is dominated by the low frequency vibration, which is less than 50 Hz, as shown in Figure 14. It shows that the vibration in the y -direction (v) is stronger than that in the other two directions. There is a peak (about 25 Hz) in the x -direction (u) and z -direction (w). This fluctuation presents in the boundary layer nearby the leading

edge and will develop into larger vibrations with higher frequencies, as shown in Figure 15. The peak frequency increases to about 1000 Hz at $0.313 c$, $0.5 c$, and $0.035 c$ from the model surface. Meanwhile, the PSD of v increases from 10^{-7} to 10^{-4} in magnitude. The unsteadiness of the vibrations increases greatly with the development of boundary layer implying the evolution of waves in the boundary layer. The development of these waves agrees well with the description of T-S instability when the boundary layer develops to turbulent flow from laminar flow^[43, 44]. Moreover, the friction factor on the upper airfoil surface abruptly increases at $0.32\sim 0.33 c$, indicating that the transition presents near the monitored point at $0.313 c$.

4.2.3 Spanwise modes of velocity in boundary layer

In order to examine the velocity distributions near the leading edge in the boundary layer, the flow field is simulated based on DU 91-W2-250_6 after 6 seconds of calculation. Instantaneous velocities on a spanwise line at $5.3\%c$ near the leading edge on the suction surface were recorded every 0.004 s, i.e., 40 iterations. Figure 16 shows the velocity distributions at 6.004 s, in which the velocities in the z -direction and y -direction are transformed into normal and tangential velocities respect to the airfoil surface because they represent the boundary layer flow better than v and w . It shows that the velocity distributions are not uniform but vary in the spanwise direction. The maximum amplitude of normal velocity is about 0.5% of the average velocity; and, the maximum amplitude of tangential velocity is about 0.06% of the average velocity. The spanwise velocity shows like a sinusoidal distribution with small fluctuations. The results show that in the viscous boundary layer the velocity basically keeps two-dimensional characteristics with very small variations.

Figure 16 Velocity distributions at 6.004s and $x/c=5.3\%c$ near the leading edge, the span dimension is normalized by the width of span: a) Normal velocity; b) Tangential velocity; c) Spanwise velocity

Figure 17 Spanwise velocity distributions along the span, at 5.3%c from the leading edge

To further study the variation of flow structures, the x -velocity distributions at different times are fitted and plotted in Figure 17. The integer numbers in the legend are the iteration steps. For example, $u-40$ is representative of velocities recorded at iteration 40, i.e., after 0.004 s. The variation of peaks and valleys shows that there are stages in the simulation period of 0.04 s. At the first stage

from u-40 to u-200, the peak of fitting curves varies slightly but the valley increases a lot. From u-200 to u-280 the valley returns to decrease on contrary to the peak at this time. With time going on the valley keeps decreasing as well as the peak at u-360. At u-400 the velocity distribution goes back again to increase the valley and the peak as before. The velocity in the x -direction is not uniform in space and varies with time, indicating that boundary layer is unstable. The authors consider it as a low frequency variation because in the simulation period the variation of direction only changes two times oppositely. The frequency may be around 25 Hz estimated from the total simulation time. On the other hand, a low characteristic frequency is found in the spectra of forces and velocities near the leading edge in Figure 9 and Figure 15. That should be the reflection of non-uniformity and instability of the boundary layer.

Figure 18 Power spectral density in wavenumber domain, the left column is at time iteration 200 and the right column is at time iteration 240.

By making a spatial FFT we get a power spectral density in wavenumber domain, as shown in Figure 18. The spatial sample step is calculated by the spanwise grid step, which is in averaged distribution. Figure 18 shows that the boundary layer instability is dominated by long waves with a small wavenumber. The dominant wavenumbers in the spanwise, normal, and tangential directions are 2 (6 at some time), 14, and 2, respectively. Because the span of the airfoil model is 0.5 m, the minimum wavenumber that can be recognized is 2 m^{-1} . As shown in Figure 16 b), the longest wave in the tangential velocity is not intact, but it is still taken as an intact wave with wavenumber 2. On the other hand, the spanwise velocity distribution contains an intact wave of wavenumber 2. This result provides an additional evidence that the spanwise variation of the boundary layer contains a dominant frequency of about 25 Hz.

Due to the effect of the boundary layer instability, the dominant wave of the spanwise velocity distribution varies with time. At some specific time, the wave of wavenumber 6 has a larger energy than the wave of wavenumber 2 and then dominates the spanwise flow. Conclusively, long waves of velocity distribution show a low frequency fluctuation of velocity in the boundary layer and dominate the instability in the boundary layer.

5. Conclusions

The LES method has been applied to investigate unsteady aerodynamic performance of blunt wind turbine airfoils. The experimental data at the same Reynolds number is used to validate the LES method including comparisons of aerodynamic forces, pressure distributions, and dominate frequency in the wake. The key points of this paper are: 1) comparison of aerodynamic unsteadiness between the Exp. and LES simulation; 2) presenting separation bubbles in the boundary layer affected by the trailing edge thickness; 3) presenting the boundary layer transition

process on blunt airfoils; 4) showing the unsteadiness of boundary layer of blunt airfoils in the spanwise direction.

The aerodynamic performance of blunt airfoils is highly unsteady as well as velocities in the wake region. FFT on forces and velocities shows that the higher characteristic frequency is caused by vortex shedding, which is broadband as shown in the results of both LES and experiment. The peak frequencies of lifts in the band are about 450 Hz and 318 Hz, respectively, for DU 91-W2-250_6 and DU 91-W2-250_10 at $\alpha=0^\circ$ and $v=30$ m/s when the chord length is 0.15 m. Moreover, the results show that the frequency of aerodynamic forces depends on the vortex shedding but the peak frequencies may be slightly different between forces and monitored velocities. The trailing edge thickness has a strong effect on the spreading of vortex in the wake. The peak frequency can be found at $3.4 c$ in the wake of DU 91-W2-250_10 but disappears at $2.3 c$ in the wake of DU 91-W2-250_6. Therefore, large shedding vortices spread farther downstream than small vortices in the wake.

Boundary layer instability has been investigated in three aspects including boundary layer separation, spectrum characteristics of velocity, and spanwise modes of velocity. First, the boundary layer of the original airfoil undergoes a serious separation even at zero angle of attack due to the low Reynolds number. The separation bubble moves towards the leading edge with the increase of angle of attack on the suction surface, while opposite on the pressure surface. Enlarging the trailing edge thickness can smooth the separation but a too much trailing edge gap plays weaker effects on smoothing the separation due to the increase of base pressure. Second, the spectral characteristics of velocity at special locations ahead of the leading edge and in the boundary layer show the instability of the boundary layer. The characteristic frequencies are about 25 Hz at the position ahead of the leading edge and in the boundary layer near the leading edge. With the development of boundary layer, the low frequency unsteady flow develops into a high frequency unsteady flow, which is in agreement with the development of T-S waves that lead to laminar-turbulent transition in boundary layer flows. Finally, an examination of spanwise modes of

velocity shows an instability of boundary layer in the spatial domain. Velocity distributions along the span are not uniform and vary with time, which shows the three-dimensional characteristics of boundary layer. The boundary layer instability is from the low frequency unsteadiness judging from the development of spanwise distributions of velocities and dominant wavenumbers.

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1. Comparison of unsteady aerodynamics of blunt airfoils Between Exp. and LES
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3. Presenting the effect of trailing edge size on separation bubbles in boundary layer
4. Presenting the boundary layer transition process on blunt airfoils
5. Unstable boundary layer in spanwise is characterized by low frequent wave