

**Computational investigation of  
installation noise reduction in  
urban air mobility vehicles using  
structured porous surface  
treatments**

**A Thesis Submitted for the Degree  
of Doctor of Philosophy**

**By**

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## DECLARATION

I declare that this thesis is my own work, undertaken at Brunel University of London between January 2022 and May 2026, and has not been submitted for any other degree or professional qualification. Where the work of others has been used, it has been appropriately acknowledged and referenced.

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# CONTRIBUTIONS

## Journal Papers

1. Naicker, C. J., Tyacke, J. C., 2026, *Large-Eddy Simulation of Urban Air Mobility Vehicle Installation Noise and Its Reduction*, Aerospace Science and Technology (draft).
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## Software Development

1. Developed and implemented a Zonal-LES model within OpenFOAM.
2. Beta-tested the upSuite parallel HDF5 I/O library.
3. Developed a Python code to mitigate Pointwise periodic boundary condition export issues.

## ABSTRACT

Urban Air Mobility Vehicles (UAMVs) offer a possible solution to the ever-increasing ground-based congestion by utilising the free airspace. However, since these vehicles are expected to operate close to densely populated areas, the noise generated during take-off, landing and low-altitude flight is a major concern, both with respect to certification and public perception. Due to the use of electric propulsion, the contribution of engine noise is reduced, and therefore aerodynamic noise from the rotors, airframe and their interactions becomes increasingly important. This thesis investigates the use of turbulence-resolving Computational Fluid Dynamics (CFD) and computational aeroacoustics to predict the flow and noise associated with UAMVs, alongside the impact of porous coatings for installation noise reduction.

The simulations were carried out using OpenFOAM. The computational setup was first assessed using high-performance computing scaling tests, followed by 2-D and 3-D NACA 0012 validation cases. The 2-D Reynolds-Averaged Navier–Stokes (RANS) validation showed good agreement with NASA reference data, with maximum deviations of 2% in lift coefficient and 3.5% in drag coefficient when compared to computational reference codes. The 3-D Large-Eddy Simulation (LES) validation captured the main flow features of the blunt trailing-edge case, including separation, reattachment, transition and vortex shedding.

The validated setup was then applied to low-Mach-number aeroacoustic simulations of a tripped NACA 0012 airfoil. The results showed that tripping was required to reproduce the experimental boundary-layer development, and that LES and Zonal LES were able to predict the main flow and acoustic behaviour. In contrast, the Improved Delayed Detached Eddy Simulation (IDDES) produced limited resolved turbulence in the near-wall region and significantly underpredicted the acoustic levels. The acoustic predictions also showed that Ffowcs Williams–Hawkins (FW-H) surface and closing-disc placement had a strong influence on the predicted acoustic spectra. The best placement and closing disc location were identified providing future guidance.

Structured porous coatings were then investigated using both fully resolved geometry and reduced-order approaches. Fully resolved LES of a structured porous coated cylinder showed that the coating modified the near-cylinder flow through internal separation, circumferential pore flow, shear-layer diffusion and boundary-layer development on the inner cylinder. A Darcy–Forchheimer approximation was then developed to represent the porous coating at lower computational cost, and was shown to reproduce several of the main wake trends while also highlighting the sensitivity of the results to the selected model coefficients.

Finally, the porous coating was assessed for modelled vortex impingement on both a cylinder and an airfoil leading edge. For the cylinder case, the coating diffused the incoming vortex, reduced near-wake turbulence intensity and reduced the Overall Sound Pressure Level by up to 5.8 dB. For the airfoil leading-edge case, the coating weakened the downstream vortex imprint, reduced surface-pressure fluctuations near the leading edge and reduced the Overall Sound Pressure Level at all observer angles, with a maximum reduction of 8.1 dB.



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# NOMENCLATURE

## Symbols

### Latin and Other Symbols

| Symbol              | Description                                             | Unit                       |
|---------------------|---------------------------------------------------------|----------------------------|
| $C$                 | Airfoil chord length                                    | m                          |
| $C_D$               | Drag coefficient                                        | –                          |
| $C_f$               | Skin-friction coefficient                               | –                          |
| $C_L$               | Lift coefficient                                        | –                          |
| $C_T$               | Propeller thrust coefficient                            | –                          |
| $D$                 | Outer diameter of the coated cylinder                   | m                          |
| $D_m$               | Darcy resistance coefficient                            | $\text{m}^{-2}$            |
| $d$                 | Bare / inner-cylinder diameter                          | m                          |
| $F$                 | Forchheimer inertial resistance coefficient             | $\text{m}^{-1}$            |
| $f$                 | Frequency                                               | Hz                         |
| $\tilde{f}_d$       | Empirical IDDES/Zonal-LES blending function             | –                          |
| $H$                 | Boundary-layer shape factor, $H = \delta^*/\theta_{BL}$ | –                          |
| $k$                 | Turbulent kinetic energy                                | $\text{m}^2 \text{s}^{-2}$ |
| $k_i$               | Induced-power factor used in the rotor inflow model     | –                          |
| $L_c$               | Spanwise coherence length                               | m                          |
| $L_{\text{exp}}$    | Experimental span used in spanwise acoustic correction  | m                          |
| $l_{\text{IDDES}}$  | IDDES length scale                                      | m                          |
| $l_{\text{LES}}$    | LES-based length scale                                  | m                          |
| $l_{\text{RANS}}$   | RANS-based length scale                                 | m                          |
| $L_{\text{sim}}$    | Simulation span used in spanwise acoustic correction    | m                          |
| $M$                 | Mach number                                             | –                          |
| $p$                 | Pressure                                                | Pa                         |
| $P_{\text{ref}}$    | Reference acoustic pressure                             | Pa                         |
| $p_{\text{RMS}}$    | Root-mean-square surface pressure                       | Pa                         |
| $Q_{\text{norm}}$   | Normalised $Q$ -criterion                               | –                          |
| $R$                 | Propeller radius                                        | m                          |
| $r$                 | Radial distance from the vortex centre                  | m                          |
| $r_c$               | Vortex core radius                                      | m                          |
| $r_{\text{cutoff}}$ | Cut-off radius for the modified Kaufmann vortex         | m                          |
| $Re$                | Reynolds number                                         | –                          |
| $St$                | Strouhal number                                         | –                          |
| $T^*$               | Characteristic time scale                               | s                          |
| $U$                 | Streamwise velocity                                     | $\text{m s}^{-1}$          |
| $u^*$               | Friction velocity                                       | $\text{m s}^{-1}$          |
| $U_{\infty}$        | Freestream velocity                                     | $\text{m s}^{-1}$          |
| $v$                 | Induced velocity in hover                               | $\text{m s}^{-1}$          |
| $V_T$               | Rotor tip speed                                         | $\text{m s}^{-1}$          |

| Symbol              | Description                                                                                    | Unit              |
|---------------------|------------------------------------------------------------------------------------------------|-------------------|
| $V_\theta$          | Tangential vortex velocity                                                                     | $\text{m s}^{-1}$ |
| $x, y, z$           | Cartesian coordinates (streamwise, wall-normal/vertical, and spanwise directions respectively) | m                 |
| $y^+$               | Dimensionless wall distance                                                                    | –                 |
| $y_{\text{LES}}^+$  | Upper $y^+$ limit above which the LES length scale is used                                     | –                 |
| $y_{\text{RANS}}^+$ | Lower $y^+$ limit below which the RANS length scale is used                                    | –                 |

## Greek Symbols

| Symbol                               | Description                                                                         | Unit                             |
|--------------------------------------|-------------------------------------------------------------------------------------|----------------------------------|
| $\alpha$                             | Angle of attack                                                                     | $^\circ$                         |
| $\Delta$                             | LES filter width                                                                    | m                                |
| $\Delta x^+, \Delta y^+, \Delta z^+$ | Streamwise, wall-normal and spanwise grid spacing in wall units                     | –                                |
| $\Delta z_{ij}$                      | Spanwise separation between the two pressure signals used in the coherence function | m                                |
| $\delta^*$                           | Boundary-layer displacement thickness                                               | m                                |
| $\delta_k$                           | Boundary-layer height at the trip location                                          | m                                |
| $\gamma$                             | Heat-capacity ratio                                                                 | –                                |
| $\Gamma$                             | Vortex circulation strength                                                         | $\text{m}^2 \text{s}^{-1}$       |
| $\Gamma_{ij}^2$                      | Magnitude-squared coherence between two surface-pressure signals                    | –                                |
| $\lambda$                            | Rotor inflow ratio                                                                  | –                                |
| $\mu$                                | Dynamic viscosity                                                                   | $\text{kg m}^{-1} \text{s}^{-1}$ |
| $\nu$                                | Kinematic viscosity                                                                 | $\text{m}^2 \text{s}^{-1}$       |
| $\nu_t$                              | Turbulent/subgrid kinematic eddy viscosity                                          | $\text{m}^2 \text{s}^{-1}$       |
| $\nu_\infty$                         | Freestream kinematic viscosity                                                      | $\text{m}^2 \text{s}^{-1}$       |
| $\omega$                             | Specific dissipation rate                                                           | $\text{s}^{-1}$                  |
| $\Omega$                             | Propeller angular velocity                                                          | $\text{rad s}^{-1}$              |
| $\omega_y$                           | Wall-normal vorticity                                                               | $\text{s}^{-1}$                  |
| $\omega_z$                           | Spanwise vorticity                                                                  | $\text{s}^{-1}$                  |
| $\phi$                               | Porosity                                                                            | –                                |
| $\phi_{pp}$                          | Acoustic pressure power spectral density                                            | $\text{dB Hz}^{-1}$              |
| $\phi_{uu}$                          | Velocity power spectral density                                                     | $\text{dB Hz}^{-1}$              |
| $\rho$                               | Fluid density                                                                       | $\text{kg m}^{-3}$               |
| $\tau_w$                             | Wall shear stress                                                                   | Pa                               |
| $\theta$                             | Observer polar angle                                                                | $^\circ$                         |
| $\theta_{BL}$                        | Boundary-layer momentum thickness                                                   | m                                |

## Acronyms and Abbreviations

| Acronym | Definition                                         |
|---------|----------------------------------------------------|
| AoA     | Angle of Attack                                    |
| BOFFS   | Block Overset Fast Flow Solver                     |
| BPM     | Brooks, Pope and Marcolini                         |
| CFD     | Computational Fluid Dynamics                       |
| CPU     | Central Processing Unit                            |
| DNS     | Direct Numerical Simulation                        |
| EUAMV   | Electric Urban Air Mobility Vehicle                |
| eVTOL   | Electric Vertical Take-off and Landing             |
| FFT     | Fast Fourier Transform                             |
| FW-H    | Ffowcs Williams-Hawkings                           |
| GT      | Garrick Triangle                                   |
| HDF5    | Hierarchical Data Format, version 5                |
| HPC     | High Performance Computing                         |
| IDDES   | Improved Delayed Detached Eddy Simulation          |
| I/O     | Input/Output                                       |
| LES     | Large Eddy Simulation                              |
| LUST    | Linear Upwind Stabilized Transport                 |
| NACA    | National Advisory Committee for Aeronautics        |
| NASA    | National Aeronautics and Space Administration      |
| OASPL   | Overall Sound Pressure Level                       |
| PIMPLE  | Hybrid PISO–SIMPLE algorithm                       |
| PISO    | Pressure-Implicit with Splitting of Operators      |
| PSD     | Power Spectral Density                             |
| RANS    | Reynolds-Averaged Navier–Stokes                    |
| RMS     | Root Mean Square                                   |
| RPM     | Revolutions Per Minute                             |
| SGS     | Subgrid-Scale                                      |
| SIMPLE  | Semi-Implicit Method for Pressure Linked Equations |
| SPCC    | Structured Porous Coated Cylinder                  |
| SPL     | Sound Pressure Level                               |
| SST     | Shear Stress Transport                             |
| TKE     | Turbulent Kinetic Energy                           |
| TI      | Turbulence Intensity                               |
| UAM     | Urban Air Mobility                                 |
| UAMV    | Urban Air Mobility Vehicle                         |
| VTOL    | Vertical Take-off and Landing                      |
| WALE    | Wall-Adapting Local Eddy-viscosity                 |
| ZLES    | Zonal Large-Eddy Simulation                        |

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# 1 INTRODUCTION

The following chapter provides an insight into the Background and Motivation for the project.

## 1.1 Background and Motivation

Urbanization is defined as “the migration from rural areas into towns and cities” [1]. The United Nations has developed standardized definitions for cities, towns and suburbs, and rural areas. They are defined as:

- **Cities:** A minimum of 1,500 people per km<sup>2</sup>, and at least 50,000 people in total.
- **Towns and suburbs:** A minimum of 300 people per km<sup>2</sup>, and at least 5,000 people in total.
- **Rural areas:** sparsely populated areas such as settlements and small villages. [1]

Based on these definitions, the share of the world’s population living in cities, towns and suburbs, and rural areas is shown in Figure 1.1. From Figure 1.1, it can be seen that approximately 80% of the world’s population live in cities, towns and suburbs, with the population in cities steadily increasing.

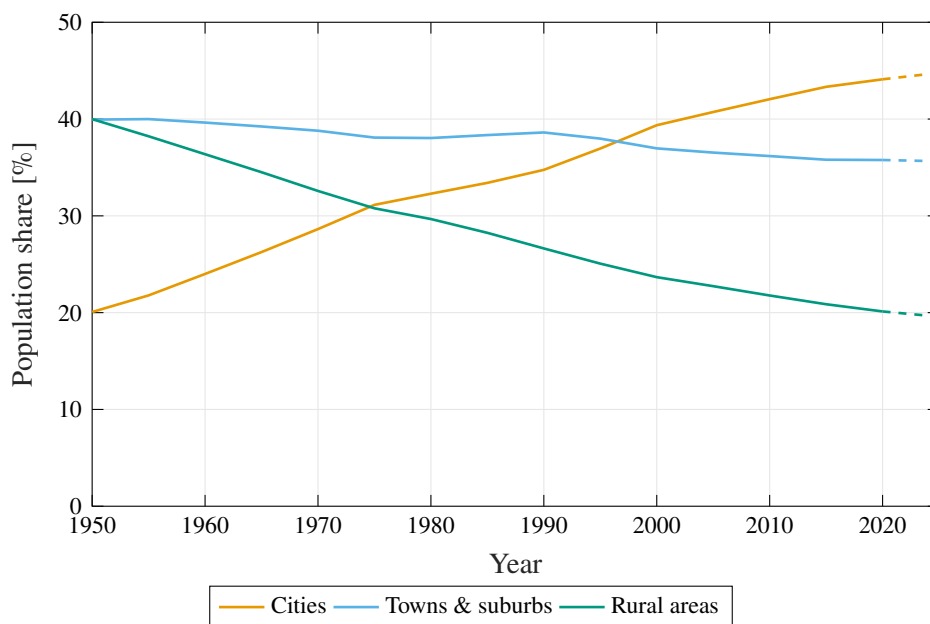


Figure 1.1: World population distribution by settlement type (dashed lines are projected). Adapted from Ortiz-Ospina et al. [1].

This continuous increase in urbanisation results in increased congestion within the cities which not only negatively impacts the lives of the urban citizens with respect to traffic, queuing and overcrowded public transport, but also increases emissions and puts excess stress on existing infrastructure. Electric Urban Air Mobility Vehicles (EUAMVs) offer a solution to this problem by utilising airspace, thereby reducing congestion on the ground, as well as offering zero operational emissions as a result of using battery power. Furthermore, a Life Cycle Assessment (LCA) conducted on Electric Vertical Take-off and Landing (eVTOL) concepts by André and Hajek [10], found that with respect to greenhouse gas emissions, eVTOL

vehicles can environmentally compete with internal combustion engine vehicles, provided that the relative time spent in hover during a mission is low and the energy required to charge the batteries is supplied by a sufficiently clean energy grid. Under certain circumstances, it was found that eVTOL vehicles can even environmentally compete with electric cars. The conditions are as follows:

- The required energy is mainly provided by renewable energy sources;
- The available seats on the aircraft are fully utilised;
- The mission results in an overall distance and time saving compared to an on-the-ground mission (a car travelling on the road with the usual traffic);
- The batteries must be rechargeable a minimum of 1000 times [10].

Figure 1.2 shows the Vertical Aerospace EUAMV in both hover/take-off and forward flight configurations. In hover/take-off configuration (Figure 1.2a), it can be seen that the front rotors tilt upwards, and the rear rotors are active in order to generate the vertical thrust required for this flight phase. For forward flight (Figure 1.2b), the front rotors tilt forward and the rear rotors are stowed. The tilt-rotor configuration is also utilised by other EUAMV manufacturers such as Joby and Archer.



(a) Hover/take-off (reproduced from Vertical Magazine [11])



(b) Forward flight (reproduced from Vertical Magazine [12])

Figure 1.2: Vertical Aerospace VX4 aircraft in hover and forward flight phase.

EUAMVs are expected to operate between a network of “vertiports”, which are hubs that feature take-off and landing pads and charging facilities. It is claimed that the development of vertiports will be much cheaper than the development of other ground infrastructure such as tunnels, railways and roads since many already existing rooftop garages and carparks can be repurposed to be used as vertiports [2]. A vertiport, as envisioned by UBER Elevate, can be seen in Figure 1.3.



Figure 1.3: UBER Elevate vertiport vision (reproduced from UBER Elevate [2])

The possible benefits of Urban Air Mobility Vehicles (UAMVs) are therefore evident. In addition, there are currently more than 150 UAM concepts being developed worldwide, in a range of different configurations [10], which supports a promising future for UAMVs.



All the proposed eVTOL designs make use of multirotor configurations, with either ducted or unducted rotors, for both efficiency and the VTOL requirements. Due to the electric propulsion of UAMVs, the noise of these electrical systems can be considered to be negligible, which leaves rotor noise and their mutual interaction noise, as the dominant emission [13]. The “humming noise” of several propellers during take-off has been described as “unbearable” by Filippone and Barakos [13]. In addition to this, airframe interaction noise will also play a significant role, however, this is also directly related to rotor flow.

Rotor noise is a cause for concern, not just in the UAMVs field, but in many other applications such as:

- Ship propellers - Propellers are the dominant noise source for most vessels. The sound generated by the propellers is radiated to the far-field, and can possibly interfere with the critical survival activities (communication, location of prey and navigation) of sea mammals such as whales and dolphins [14].
- Airboat propellers - Airboats are especially common in lakefront communities, and can be a nuisance to the community and also affect fish and other wildlife. An analysis conducted by Perez et al. [15] found that rotor noise, specifically from rotors with a high tip Mach number, i.e., a tip Mach number of  $M > 0.7$ , is the greatest contributor of noise generated by an airboat.
- Wind turbine rotors - Since wind constantly changes velocity and direction, and also increases with height, the inflow of air to wind turbines is rarely stable. This, coupled with inflow turbulence from other wind turbines results in the generation of impulsive noise, which can be described as an annoying “swishing” or “thumping” sound [16].
- Propeller aircraft - Although highly efficient, propeller-driven aircraft are plagued by high levels of cabin noise, resulting in its commercial usage being limited [3].

### 1.1.1 Health Effects and Social Impacts

Unlike the examples mentioned above, UAMVs face a greater challenge in the sense that they have not yet been commercialised, therefore, the issue of noise will need to be addressed before the concepts achieve certification and community acceptance. Environmental noise can be defined as a type of occupational noise that includes noise from road, rail and air traffic as well as noise from industrial construction, amongst others [17]. This type of noise is more prevalent in urban areas. Research shows that a forty-year exposure to a 75 A-weighted Decibels (dBA) sound level for eight hours a day, can result in permanent hearing loss [18]. With respect to non-auditory effects, environmental noise has been linked to a range of adverse health effects such as sleep disturbances, annoyance, impaired cognitive performance in children and even cardiovascular disease. An exposure to a Day-Evening-Night-Level ( $L_{DEN}$ ) of 55 dB per year is considered a health risk [17].

Given that a typical mission for an Urban Air Mobility Vehicle (UAMV) will involve taking off and landing in densely populated areas, i.e., urban areas, and relatively low altitude flight over these areas, the urban citizens will be directly affected by the noise emitted by these vehicles. An Automated Flight Planning System (AFPS) proposed by Tang et al. [19] used an estimate of just 1400 ft. for the maximum flight altitude of a UAMV, which is significantly lower than the flight altitude of a commercial aircraft, which is typically 30,000 to 38,000 ft. according to Sforza [20].

There have been UAMV community perception studies carried out by the likes of European Union Aviation Safety Agency (EASA) and Airbus. The EASA study conducted in 2021 polled 4,000 participants from six different European urban areas. The top three concerns of the participants for UAMVs being used as air taxis were environmental concerns, noise related concerns and safety [21]. The percentage of participants who ranked the combination of “Noise related to flying aircraft” and “Noise related to vertiports” as either their first, second or third biggest concern, was 38%, ranking even higher than safety at 37% [21].

The Airbus community perception study polled 1,540 participants, equally divided among four regions, namely, New Mexico, Los Angeles, New Zealand and Switzerland. The parameters investigated are summarised in Table 1.1. The participants were asked to rate how they feel using a Likert Scale, based on a combination of parameters listed in the table. The study found that the biggest concern of the participants was safety, with 56% of participants being concerned about safety, whilst the type of noise generated and the volume of noise generated by UAMVs were tied as the second biggest concern, with 49% of participants being concerned about these two factors [9].

*Table 1.1: Parameters investigated in the Airbus community perception study (adapted from Yedavalli and Mooberry [9])*

|                                      |                                                           |
|--------------------------------------|-----------------------------------------------------------|
| <b>Altitude</b>                      | Low, medium, high, very high                              |
| <b>Frequency of sound every hour</b> | 1, 2, 5, 10, 100                                          |
| <b>Landing spot</b>                  | Far away, near                                            |
| <b>Number of passengers</b>          | 2, 4, 6, 12, 20                                           |
| <b>Type of passenger</b>             | Diverse, students, family, executive, missing             |
| <b>Time of day</b>                   | Morning, afternoon, night, middle of the night            |
| <b>Type of sounds</b>                | Car passing, bee buzzing, truck passing, helicopter above |

### 1.1.2 Rotor Acoustic Traits

There are two noticeably different types of acoustic signatures emitted by rotating blades, namely:

- Tonal or harmonic noise - Caused by a periodic repetition of sources during each rotation.
- Broadband noise - Caused by turbulent flow over the blades which results in the generation of a chaotic, non-periodic signal [3].

Figure 1.4 shows a 3 Hz bandwidth rotor noise spectrum for a rotor with 3 blades rotating at 600 RPM. From Figure 1.4, the two distinctly different acoustic signatures can be seen.

It is worthwhile noting that many of the proposed UAMV designs have relatively small rotors compared to typical helicopter rotors or aircraft propellers, which results in a lower blade tip Mach number (typically  $0.2 \leq M_{tip} \leq 0.5$ ) and as such, the relative importance of broadband noise increases.

When addressing broadband noise, it is important to consider the complex psychological phenomenon of perceived loudness and perceived annoyance. In this phenomenon, people may perceive a sound emitted by a source to be louder and more annoying than the same level of sound being emitted by a different source [22]. For example, Steele and Chon [23] carried out a noise perception study on 16 participants. For this study, six different types of sounds were used, namely, (a) pink noise; (b) two-sinusoids; (c) narrowband noise; (d) broadband noise; (e) narrowband tone cluster; and (f) broadband tone cluster. The

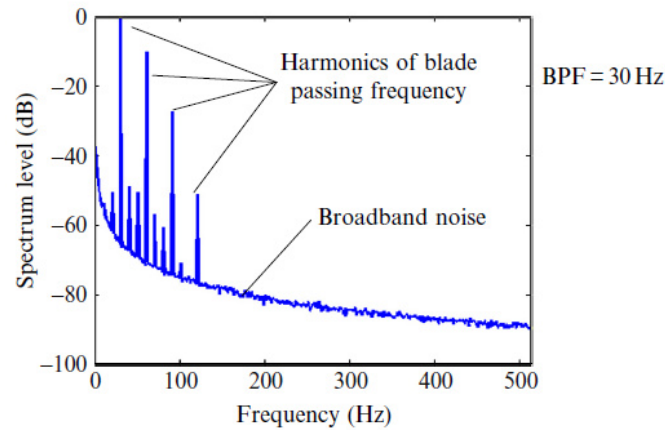


Figure 1.4: Rotor noise spectrum showing harmonic and broadband noise (reproduced from Glegg and Devenport [3]).

experiment was carried out at four different Sound Pressure Levels (SPLs), namely, 50; 60; 70; and 80 dB.

By averaging the stimulus ranking for each participant over the four intensity levels, and then averaging the averaged stimulus ranking for all sixteen participants, the absolute stimulus annoyance per stimulus type was found. Due to the averaging over the four intensity levels, the absolute annoyance was independent of loudness. To determine the loudness influence of each stimulus type, the averaging over the four intensity levels was removed, and the difference between each participant's highest annoyance ranking and lowest annoyance ranking was taken, and then averaged over the sixteen participants. The logic behind this was that the greatest difference corresponds with the greatest ability of the intensity level to influence annoyance. It was found that broadband noise ranked the highest, meaning that if the Sound Pressure Level (SPL) were to be increased for all stimuli by a fixed amount, there will be a more drastic increase in the perceived annoyance of broadband noise compared to the other noise sources.

Based on the possible health effects of noise from UAMVs and the public perception towards both the type of noise and level of noise generated by UAMVs, the necessity of addressing the issue of rotor noise in these vehicles is evident.

# 2 LITERATURE REVIEW

The following chapter focuses on UAMV noise sources; a background on aeroacoustics; experimental and computational aeroacoustic investigations; and noise mitigation strategies.

## 2.1 Noise Sources Definitions

Figure 2.1 shows the community noise sources for VTOL/UAM vehicles, according to Li and Lee [4]. From the figure, it can be seen that blade flow related noises make up the bulk of the noise generated by VTOL/UAM vehicles. Furthermore, these noise sources are not specific to EUAMVs, in which case, even the motor noise can be considered negligible. Note that BWI represents Blade Wake Interaction Noise.

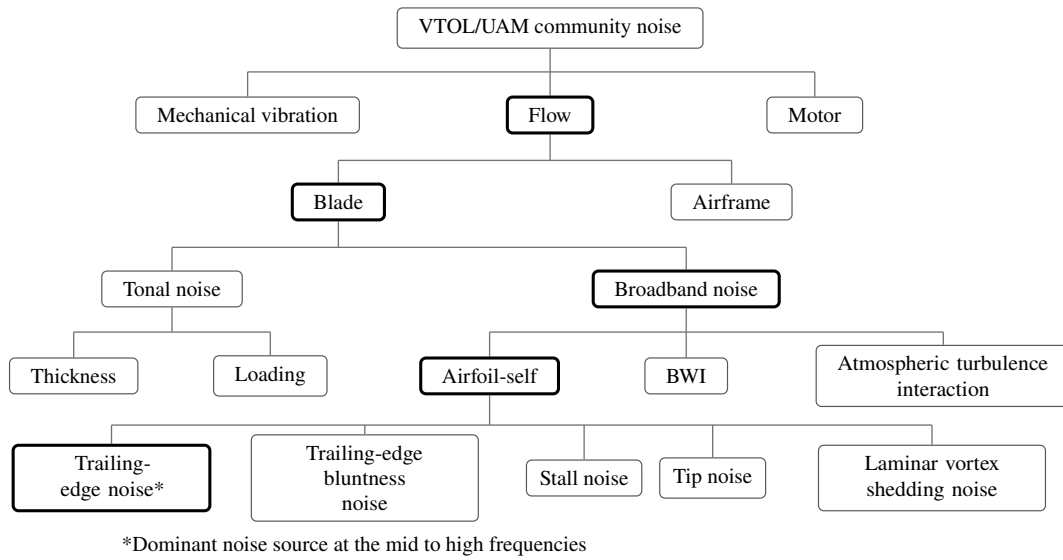


Figure 2.1: UAM VTOL community noise sources and classifications (adapted from Li and Lee [4])

### 2.1.1 Thickness and Loading Noise

Thickness noise refers to noise generated by a moving blade displacing a volume of air, where the volume displaced is dependent on the blade thickness [24]. Loading noise refers to the acoustic disturbance caused by the accelerating force generated by the moving blade [25]. Both thickness noise and loading noise are “discrete frequency” or “tonal” noise sources generated by rotating blades.

### 2.1.2 Airfoil Self-Noise

There are five mechanisms for airfoil self-noise generation [5], namely:

- Turbulent boundary layer trailing-edge noise (Figure 2.2a), characteristic of high Reynolds number flows in which turbulent boundary layers develop over a large part of the airfoil surface, causing noise as this turbulence passes over the airfoil trailing-edge
- Laminar boundary layer vortex shedding noise (Figure 2.2b), characteristic of low Reynolds number flows, in which noise is created by instabilities within the laminar boundary layers which cause vortex shedding, and corresponding trailing edge noise.

- Separation stall at non-zero angles of attack in which trailing-edge noise is generated by turbulent vorticity. In cases of deep stall, noise is radiated over the entire chord of the airfoil. (Figure 2.2c)
- Trailing-edge bluntness vortex shedding noise caused by the flow separation aft of the trailing-edge. (Figure 2.2d)
- Blade tip vortex formation noise caused by the vortex flowing over the blade tip. (Figure 2.2e)

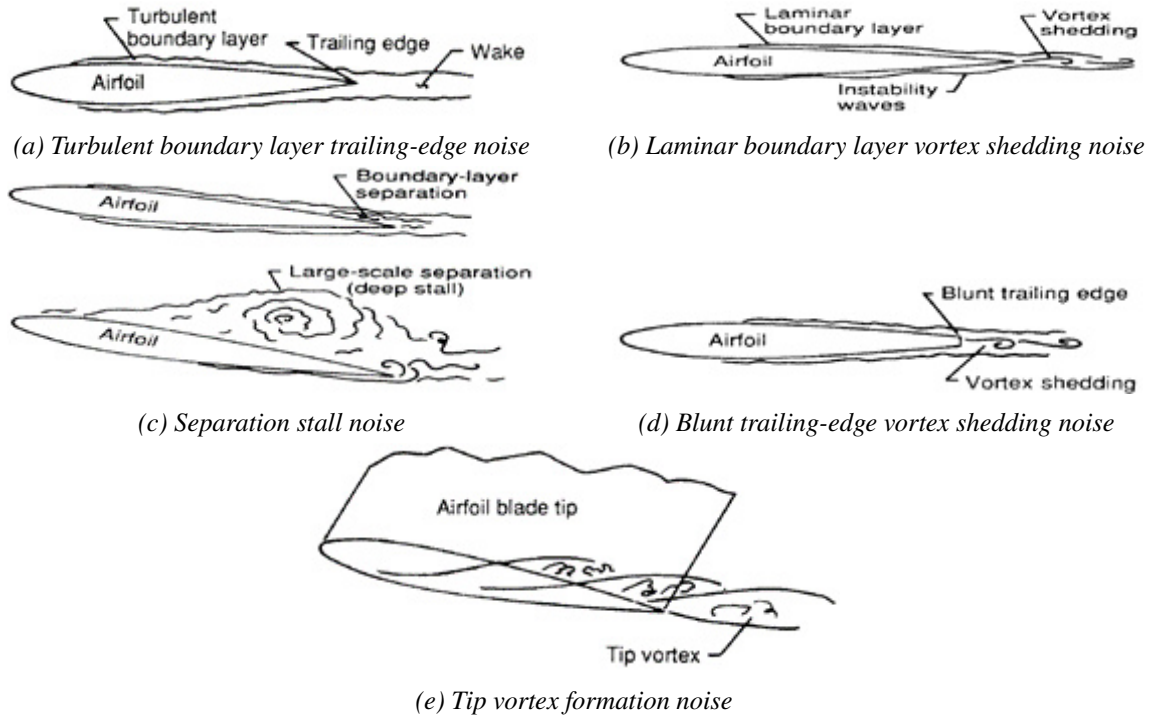


Figure 2.2: Mechanisms of airfoil self-noise generation (reproduced from Brooks et al. [5])

### 2.1.3 Blade Wake Interaction Noise

Rotor blade wake interaction noise is caused by a rotor blade interacting with vortices generated by the “previous” blades, resulting in pressure fluctuations on the rotor blade [26], as shown in Figure 2.3.

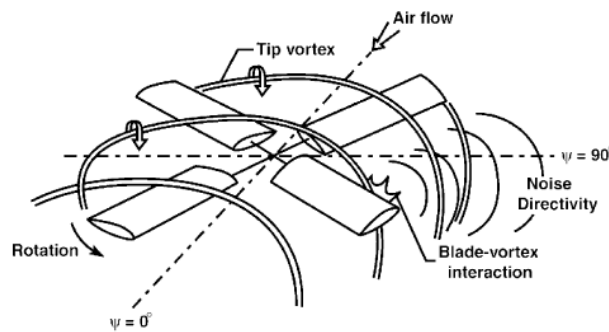


Figure 2.3: Blade wake interaction noise (reproduced from Villafana [6])

In addition to interacting with subsequent rotor blades, the vortices are convected downstream within the rotor wake and may impinge on nearby airframe surfaces, generating further unsteady pressure fluctuations and contributing to rotor-airframe interaction noise.

## 2.2 Computational Aeroacoustics

The following section reviews Linear Acoustics, Lighthill's Acoustic Analogy, Curle's Acoustic Analogy, the Farassat 1A and Garrick Triangle formulations of the Ffowcs Williams–Hawkins equation, and the Brooks, Pope and Marcolini (BPM) semi-empirical model. Although Linear Acoustics and Lighthill's Acoustic Analogy are not applied explicitly in the present research, they provide the theoretical foundation for the aeroacoustic models considered later in this work.

### 2.2.1 Linear Acoustics

The linear acoustic wave equation is given by Equation 2.1:

$$\frac{1}{c_0^2} \frac{\partial^2 p'}{\partial t^2} - \frac{\partial^2 p'}{\partial x_i^2} = 0 \quad (2.1)$$

For Equation 2.1:

$$\begin{array}{l|l} c_0^2 = \left( \frac{\partial p}{\partial \rho} \right)_s & \text{Isentropic bulk modulus, } c_0 \text{ represents the speed of sound propagation} \\ p' = \rho' c_0^2 & \text{Pressure perturbation about the mean pressure } p_0 \\ \rho' & \text{Density perturbation about the mean density } \rho_0 \end{array}$$

For a one-dimensional plane wave, the solution to the linear wave equation is given by Equation 2.2:

$$p'(x_1, t) = f(t - x_1/c_0) + g(t + x_1/c_0) \quad (2.2)$$

For Equation 2.2:

$$\begin{array}{l|l} f(t - x_1/c_0) & \text{Wave propagation in the positive } x_1 \text{ direction} \\ g(t + x_1/c_0) & \text{Wave propagation in the negative } x_1 \text{ direction} \end{array}$$

For spherical waves which are only a function of  $r$  from the origin, the wave equation is given by Equation 2.3:

$$\frac{1}{c_0^2} \frac{\partial^2 p'}{\partial t^2} - \frac{1}{r} \frac{\partial^2}{\partial r^2} (rp') = 0 \quad (2.3)$$

In which case the solution to the wave equation is given by Equation 2.4 :

$$rp' = f(t - r/c_0) + g(t + r/c_0) \quad (2.4)$$

By considering only the outward propagating waves, the solution to the wave equation can then be given by Equation 2.5:

$$p'(r, t) = \frac{f(t - r/c_0)}{r} \quad (2.5)$$

The features of a sound field can also be determined with respect to a frequency rather than time, as shown by Equation 2.6:

$$p'(r, t) = \frac{A \cos(\omega t - \omega r/c_0 - \phi)}{r} \quad (2.6)$$

For Equation 2.6:

|                |                                                                       |
|----------------|-----------------------------------------------------------------------|
| $\omega$       | Angular frequency in radians per second                               |
| $A$ and $\phi$ | Real constants to determine the wave amplitude and phase respectively |

However, the assumptions used for the equations of motion for linear acoustics do not apply to turbulent flows [3]. Therefore, for the purposes of noise generated by high-speed turbulent flow such as jet exhaust, Lighthill's Acoustic Analogy was proposed (discussed in 2.2.2). For stationary solid boundaries, Curle's analogy extends Lighthill's formulation, while for moving surfaces, such as propellers and rotors, the Ffowcs Williams and Hawkings equation is required.

## 2.2.2 Lighthill's Acoustic Analogy

For turbulent flows, such as jet exhausts, the assumptions of linear acoustics are no longer sufficient [3]. In such cases, the flow itself acts as a source of sound. Lighthill's acoustic analogy recasts the compressible flow equations into the form of a wave equation with an equivalent source term.

Lighthill's wave equation is given by Equation 2.7:

$$\frac{\partial^2 \rho'}{\partial t^2} - c_\infty^2 \frac{\partial^2 \rho'}{\partial x_i^2} = \frac{\partial^2 T_{ij}}{\partial x_i \partial x_j} \quad (2.7)$$

where the Lighthill stress tensor is defined by Equation 2.8:

$$T_{ij} = \rho v_i v_j - [(p - p_\infty) - c_\infty^2 (\rho - \rho_\infty)] \delta_{ij} - \sigma_{ij} \quad (2.8)$$

For Equations 2.7 and 2.8:

|            |                                                      |               |                                               |
|------------|------------------------------------------------------|---------------|-----------------------------------------------|
| $\rho'$    | Density perturbation relative to the ambient density | $p_\infty$    | Ambient pressure                              |
| $c_\infty$ | Ambient speed of sound                               | $\rho_\infty$ | Ambient density                               |
| $T_{ij}$   | Lighthill stress tensor                              | $\delta_{ij}$ | Kronecker delta                               |
| $\rho$     | Fluid density                                        | $\sigma_{ij}$ | Viscous stress tensor                         |
| $v_i$      | Flow velocity component                              | $x_i$         | Cartesian coordinate in the $i$ -th direction |
| $p$        | Local pressure                                       |               |                                               |

The right-hand side of Equation 2.7 represents the aerodynamic sound sources within the flow field. Lighthill's analogy is therefore well suited to free turbulent flows, but it does not explicitly account for the effects of solid boundaries. For stationary rigid surfaces, this limitation is addressed by Curle's acoustic analogy. For moving surfaces, such as rotating blades, the formulation is further extended by the Ffowcs

Williams and Hawkings equation.

### 2.2.3 Curle's Acoustic Analogy

The first analogy used in this work was the Curle analogy [27]. Curle extended Lighthill's acoustic analogy to account for stationary solid boundaries, allowing aerodynamic noise generated by unsteady surface forces to be represented in addition to the quadrupole sources of the turbulent flow.

For low-Mach-number flow around rigid bodies, Curle's acoustic analogy relates the far-field acoustic pressure fluctuation to the unsteady aerodynamic force acting on the body. In the OpenFOAM libAcoustics, the Curle analogy was implemented according to Equation 2.9 by Epikhin et al. [28]:

$$p'(\tilde{x}, t) = \frac{1}{4\pi c_0} \frac{x_i}{r^2} \left[ \frac{\partial F_i}{\partial t} + \frac{F_i}{r/c_0} \right] \quad (2.9)$$

where the force term  $F_i$  is given by Curle [27]

$$F_i(t) = \int_S P_i(y, t) dS(y) \quad (2.10)$$

For Equations 2.9 and 2.10:

|                    |                                          |             |                                                                        |
|--------------------|------------------------------------------|-------------|------------------------------------------------------------------------|
| $p'(\tilde{x}, t)$ | Acoustic pressure fluctuation            | $F_i$       | $i$ -th component of the unsteady aerodynamic force acting on the body |
| $c_0$              | Speed of sound                           | $P_i(y, t)$ | Surface force per unit area component                                  |
| $x_i$              | Observer-position component              | $S$         | Body surface                                                           |
| $r$                | Distance from the source to the observer | $dS(y)$     | Differential surface element                                           |
|                    |                                          | $t$         | Time                                                                   |

The full derivation of Curle's analogy can be found in Curle [27].

### 2.2.4 FW-H Acoustic Analogy

The second analogy used was the Ffowcs Williams-Hawkings (FW-H) acoustic analogy [29]. For a permeable FW-H surface, the FW-H equation is written in differential form as

$$\square^2(\rho - \rho_0) = \frac{\partial}{\partial t} [Q_n \delta(f)] - \frac{\partial}{\partial x_i} [L_i \delta(f)] + \frac{\partial^2}{\partial x_i \partial x_j} [T_{ij} H(f)] \quad (2.11)$$

where the variables  $Q_n$  and  $L_i$  are defined according to Equations 2.12 and 2.13 respectively:

$$Q_n = Q_i \hat{n}_i = [\rho_0 v_i + \rho(u_i - v_i)] \hat{n}_i \quad (2.12)$$

$$L_i = L_{ij} \hat{n}_j = [P_{ij} + \rho u_i(u_j - v_j)] \hat{n}_j \quad (2.13)$$



The wave operator is given by Equation 2.14

$$\square^2 = \frac{1}{c_0^2} \frac{\partial^2}{\partial t^2} - \nabla^2 \quad (2.14)$$

The first term on the right-hand side of Equation 2.11 corresponds to the thickness source term, whilst the second term corresponds to the loading source term. The third term is the quadrupole source term. In libAcoustics, the implementation ignores the quadrupole source term. The acoustic pressure is then decomposed into thickness and loading contributions according to Equation 2.15:

$$p'(\mathbf{x}, t) = p'_T(\mathbf{x}, t) + p'_L(\mathbf{x}, t) \quad (2.15)$$

For Equations 2.11–2.15:

|             |                                                  |             |                                         |
|-------------|--------------------------------------------------|-------------|-----------------------------------------|
| $\rho$      | Fluid density                                    | $Q_i$       | Thickness source vector                 |
| $\rho_0$    | Ambient density                                  | $L_{ij}$    | Loading tensor                          |
| $f$         | FW-H surface function                            | $P_{ij}$    | Compressible stress tensor              |
| $\delta(f)$ | Dirac delta function                             | $\hat{n}_i$ | Outward unit normal to the FW-H surface |
| $H(f)$      | Heaviside function                               | $u_i$       | Fluid velocity component                |
| $Q_n$       | Thickness source term normal to the FW-H surface | $v_i$       | FW-H surface velocity component         |
| $L_i$       | Loading source term                              | $c_0$       | Speed of sound                          |
| $T_{ij}$    | Lighthill stress tensor                          | $p'$        | Acoustic pressure fluctuation           |
|             |                                                  | $p'_T$      | Thickness noise contribution            |
|             |                                                  | $p'_L$      | Loading noise contribution              |

#### 2.2.4.1 Farassat Formulation 1A

For moving-source configurations, the thickness and loading contributions according to the Farassat 1A formulation [30] of the FW-H equation are given by:

$$4\pi p'_T(\mathbf{x}, t) = \int_{f=0} \left[ \frac{\dot{Q}_n + Q_{\dot{n}}}{r(1 - M_r)^2} \right]_{\text{ret}} dS + \int_{f=0} \left[ \frac{Q_n (r\dot{M}_r + c_0(M_r - M^2))}{r^2(1 - M_r)^3} \right]_{\text{ret}} dS \quad (2.16)$$

$$4\pi p'_L(\mathbf{x}, t) = \frac{1}{c_0} \int_{f=0} \left[ \frac{\dot{L}_r}{r(1 - M_r)^2} \right]_{\text{ret}} dS + \int_{f=0} \left[ \frac{L_r - L_M}{r^2(1 - M_r)^2} \right]_{\text{ret}} dS + \frac{1}{c_0} \int_{f=0} \left[ \frac{L_r (r\dot{M}_r + c_0(M_r - M^2))}{r^2(1 - M_r)^3} \right]_{\text{ret}} dS \quad (2.17)$$

The subscript ‘ret’ denotes evaluation at the retarded time,  $\tau_{\text{ret}}$ , obtained from:

$$g = \tau_{\text{ret}} - t + \frac{r}{c_0} = 0 \quad (2.18)$$

The other terms in Equations 2.16 and 2.17 are given by:

$$M_r = M_i \hat{r}_i, \quad \dot{M}_r = \frac{\partial M_i}{\partial \tau} \hat{r}_i \quad (2.19)$$

$$Q_n = Q_i \hat{n}_i, \quad \dot{Q}_n = \frac{\partial Q_i}{\partial \tau} \hat{n}_i, \quad Q_{\dot{n}} = Q_i \frac{\partial \hat{n}_i}{\partial \tau} \quad (2.20)$$

$$L_r = L_i \hat{r}_i, \quad \dot{L}_r = \frac{\partial L_i}{\partial \tau} \hat{r}_i, \quad L_M = L_i M_i \quad (2.21)$$

For Equations 2.16–2.21:

|             |                                                       |                        |                                                          |
|-------------|-------------------------------------------------------|------------------------|----------------------------------------------------------|
|             |                                                       | $\dot{Q}_n$            | Source-time derivative of $Q_n$                          |
|             |                                                       | $Q_{\dot{n}}$          | Contribution due to time variation of the surface normal |
| $r$         | Distance between source and observer at emission time | $L_r$                  | Loading term projected onto the radiation direction      |
| $dS$        | Differential surface element                          | $\dot{L}_r$            | Source-time derivative of $L_r$                          |
| $M_i$       | Source Mach number component, $M_i = v_i/c_0$         | $L_M$                  | Loading term projected onto the Mach vector              |
| $M_r$       | Radiation-direction Mach number component             | $\hat{r}_i$            | Unit radiation vector from source to observer            |
| $\dot{M}_r$ | Source-time derivative of $M_r$                       | $\tau$                 | Source time                                              |
| $M^2$       | Magnitude squared of the source Mach number vector    | $\tau_{\text{ret}}$    | Retarded emission time                                   |
|             |                                                       | $[\cdot]_{\text{ret}}$ | Quantity evaluated at the retarded time                  |
|             |                                                       | $t$                    | Observer time                                            |

The full derivation of the Farassat 1A formulation can be found in [30].

#### 2.2.4.2 Garrick Triangle Formulation

The Garrick Triangle (GT) formulation [31] of the FW-H equation is an adaptation of the 1A formulation, for cases where both the source and observer are stationary in a uniform flow. It is also referred to as the wind tunnel formulation. The full derivation can be found in Brès et al. [31]. For the GT formulation the retarded time is given by

$$\tau_{\text{ret}} = t - \frac{R}{c_0} \quad (2.22)$$

where the effective acoustic distance,  $R$ , is defined as

$$R = \frac{-M_0(x_1 - y_1) + R^*}{\beta^2} \quad (2.23)$$

with

$$R^* = \sqrt{(x_1 - y_1)^2 + \beta^2 [(x_2 - y_2)^2 + (x_3 - y_3)^2]} \quad (2.24)$$

and

$$\beta = \sqrt{1 - M_0^2} \quad (2.25)$$

The unit radiation vector components and projected Mach number are then given by

$$\hat{R}_1 = \frac{-M_0 R^* + (x_1 - y_1)}{\beta^2 R}, \quad \hat{R}_2 = \frac{x_2 - y_2}{R}, \quad \hat{R}_3 = \frac{x_3 - y_3}{R}, \quad M_R = M_i \hat{R}_i \quad (2.26)$$

The source terms are modified as

$$Q_n = [-\rho_0 U_{0i} + \rho(u_i + U_{0i})] \hat{n}_i \quad (2.27)$$

$$L_i = [P_{ij} + \rho u_i(u_j + U_{0j})] \hat{n}_j \quad (2.28)$$

The resulting GT thickness and loading integrals are

$$4\pi p'_T(\mathbf{x}, t) = \int_{f=0} \left[ \frac{\dot{Q}_n}{R(1 - M_R)^2} \right]_{\text{ret}} dS + \int_{f=0} \left[ \frac{Q_n c_0 (M_R - M^2)}{R^2 (1 - M_R)^3} \right]_{\text{ret}} dS \quad (2.29)$$

$$4\pi p'_L(\mathbf{x}, t) = \frac{1}{c_0} \int_{f=0} \left[ \frac{\dot{L}_R}{R(1 - M_R)^2} \right]_{\text{ret}} dS + \int_{f=0} \left[ \frac{L_R - L_M}{R^2 (1 - M_R)^2} \right]_{\text{ret}} dS \\ + \int_{f=0} \left[ \frac{L_R (M_R - M^2)}{R^2 (1 - M_R)^3} \right]_{\text{ret}} dS \quad (2.30)$$

For Equations 2.22–2.30:

|             |                                                       |          |                                                                 |
|-------------|-------------------------------------------------------|----------|-----------------------------------------------------------------|
| $R$         | Effective acoustic distance in uniform flow           | $M_R$    | Radiation-direction Mach number component in the GT formulation |
| $R^*$       | Auxiliary distance term in the GT formulation         | $U_{0i}$ | Mean-flow velocity component                                    |
| $\beta$     | Compressibility factor,<br>$\beta = \sqrt{1 - M_0^2}$ | $L_R$    | Loading term projected onto $\hat{R}_i$                         |
| $M_0$       | Freestream Mach number                                | $x_i$    | Observer Cartesian coordinate                                   |
| $\hat{R}_i$ | Unit radiation vector component in the GT formulation | $y_i$    | Source Cartesian coordinate                                     |

### 2.2.5 The Brooks, Pope and Marcolini (BPM) Semi-Empirical Model

The Brooks, Pope and Marcolini (BPM) model is a semi-empirical trailing-edge noise model based on the scattering of turbulent boundary-layer fluctuations by a sharp trailing edge, developed by Brooks et al. [5]. The code used in this research was validated by Lee et al. [32]. A more in-depth discussion of the model can be found in [5] and [32]. The basis of the model is a scaling law for the mean-square acoustic pressure, given by

$$\langle p^2 \rangle \propto \rho^2 v'^2 \frac{U_c^3}{c_0} \frac{L\Lambda}{r^2} \overline{D} \quad (2.31)$$

For Equation 2.31:

|                       |                                            |                |                                          |
|-----------------------|--------------------------------------------|----------------|------------------------------------------|
| $\langle p^2 \rangle$ | Mean-square acoustic pressure              | $L$            | Spanwise length of the radiating section |
| $\rho$                | Fluid density                              | $\Lambda$      | Characteristic turbulence length scale   |
| $v'^2$                | Mean-square turbulent velocity fluctuation | $r$            | Distance from source to observer         |
| $U_c$                 | Turbulence convection velocity             | $\overline{D}$ | Directivity factor                       |
| $c_0$                 | Speed of sound                             |                |                                          |

From this scaling, the BPM model introduces a scaled one-third octave sound pressure level, defined as

$$Scaled-SPL_{1/3} = SPL_{1/3} - 10 \log \left( \frac{M^5 \delta^* L \overline{D}}{r^2} \right) \quad (2.32)$$

where  $M = U/c_0$  is the freestream Mach number. The scaled spectrum is then assumed to take the form

$$Scaled-SPL_{1/3} = F(St) + K \quad (2.33)$$

where  $F$  is a universal spectral shape function of the Strouhal number and  $K$  is an empirical constant.

For Equations 2.32 and 2.33:

|                    |                                              |         |                                   |
|--------------------|----------------------------------------------|---------|-----------------------------------|
| $Scaled-SPL_{1/3}$ | Scaled one-third octave sound pressure level |         |                                   |
| $SPL_{1/3}$        | One-third octave sound pressure level        | $F(St)$ | Universal spectral shape function |
| $M$                | Freestream Mach number                       | $K$     | Empirical constant                |
| $\delta^*$         | Boundary-layer displacement thickness        | $St$    | Strouhal number                   |

The final BPM formulation separates the trailing-edge noise into suction-side, pressure-side, and angle-of-attack-dependent contributions. These are given by

$$SPL_s = 10 \log \left( \frac{\delta_s^* M^5 L D}{r^2} \right) + A \left( \frac{St_s}{St_1} \right) + (K_1 - 3) + \Delta K_1 \quad (2.34)$$

$$SPL_p = 10 \log \left( \frac{\delta_p^* M^5 L D}{r^2} \right) + A \left( \frac{St_p}{St_1} \right) + (K_1 - 3) \quad (2.35)$$

$$SPL_\alpha = 10 \log \left( \frac{\delta_s^* M^5 L D}{r^2} \right) + B \left( \frac{St_p}{St_2} \right) + K_2 \quad (2.36)$$

The total trailing-edge noise level is then obtained by logarithmically summing the individual contributions:

$$SPL_{TOT} = 10 \log_{10} \left( 10^{SPL_s/10} + 10^{SPL_p/10} + 10^{SPL_\alpha/10} \right) \quad (2.37)$$

For Equations 2.34–2.37:

|              |                                                           |              |                                                                        |
|--------------|-----------------------------------------------------------|--------------|------------------------------------------------------------------------|
| $SPL_s$      | Suction-side trailing-edge noise contribution             | $r$          | Distance from source to observer                                       |
|              |                                                           | $St_s$       | Suction-side Strouhal number                                           |
| $SPL_p$      | Pressure-side trailing-edge noise contribution            | $St_p$       | Pressure-side Strouhal number                                          |
|              |                                                           | $St_1$       | Reference Strouhal number for suction- and pressure-side noise         |
| $SPL_\alpha$ | Angle-of-attack-dependent contribution                    | $St_2$       | Reference Strouhal number for angle-of-attack-dependent noise          |
| $SPL_{TOT}$  | Total trailing-edge noise level                           |              |                                                                        |
| $\delta_s^*$ | Suction-side displacement thickness at the trailing edge  | $A(\cdot)$   | Empirical spectral shape function for suction- and pressure-side noise |
| $\delta_p^*$ | Pressure-side displacement thickness at the trailing edge | $B(\cdot)$   | Empirical spectral shape function for angle-of-attack-dependent noise  |
| $M$          | Freestream Mach number                                    | $K_1$        | Empirical amplitude correction                                         |
| $L$          | Spanwise length of the airfoil section                    | $\Delta K_1$ | Additional suction-side correction                                     |
| $D$          | Directivity factor                                        | $K_2$        | Empirical angle-of-attack correction                                   |

## 2.3 UAMV Noise

The noise characteristics of UAMVs have been investigated for both isolated propellers and complete vehicles, with most studies focussing on small-scale drone propellers and small multirotor drone platforms. With respect to isolated propellers under various flight conditions, Yang et al. [33] carried out an experimental investigation of an isolated multicopter rotor in a low-speed anechoic wind tunnel under forward-flight conditions, with rotational speed varied from 4000 to 7900 rpm, freestream velocity from 10 to 20 m/s, and shaft angle from  $0^\circ$  to  $20^\circ$ . The authors found that, at fixed rotational speed and shaft angle, increasing freestream velocity increased the time-averaged thrust, power and moments, whereas at fixed freestream velocity and rotational speed, increasing shaft angle reduced them. They also found that both tonal and broadband noise increased with increasing freestream velocity and decreasing shaft angle. Lößle et al. [34] carried out experimental measurements on a clockwise rotating  $16 \times 6$  multicopter rotor in DLR's 1 m wind tunnel at 4000 rpm, with advance ratio varied from 0.05 to 0.26 and tilt angle from  $-10^\circ$  to  $10^\circ$ . Acoustic measurements were performed with a microphone array and the tip vortices were visualised using background-oriented schlieren. The authors found that the first harmonic dominated the OASPL for all cases, and that the noise was higher upstream than downstream. They also showed that higher-harmonic and broadband noise increased with advance ratio and tilt angle, and that tip-vortex interaction was visible from an advance ratio of about 0.12 onward for the  $0^\circ$  tilt angle. For a propeller operating at a tip Reynolds number of  $4.3 \times 10^4$  to  $4.38 \times 10^4$ , Grande et al. [35] carried out an experimental study in an anechoic wind tunnel over advance ratios from 0 to 0.6. Noise was measured with a microphone array, aerodynamic forces with load and torque cells, and the blade flow was examined using oil-flow visualisation and PIV. The authors found that the blade flow featured a laminar separation bubble on the suction side for  $J = 0, 0.24$  and  $0.4$ , and that this bubble moved toward the leading edge and reduced in size as advance ratio decreased. At  $J = 0.6$  they found laminar separation without reattachment, which produced vortex shedding and a high-frequency hump in the far-field noise spectra.

With respect to propeller blade design optimisation, for an electric mini unmanned air vehicle in loiter at sea level, with a flight speed of 10 m/s and a required thrust of 3.5 N, Gur and Rosen [36] used a multidisciplinary design optimisation framework to design a quiet propeller under power, structural and side constraints. The design variables included blade geometry, cone angle, propeller radius, number of blades and rotational speed. The authors found that, for a given blade number, reducing rotational speed was the main way of reducing propeller noise, although this required increases in blade chord and pitch to maintain the required thrust. They also showed that adding structural constraints increased the optimum noise level. Weitsman and Greenwood [37] developed a computational framework coupling OpenVSP with PSU-WOPWOP to evaluate eVTOL rotor acoustic design trades in hover. The study examined the effects of disk loading, taper and twist. The authors found that reducing disk loading was beneficial for both hover performance and acoustics, whereas most other design changes involved a trade-off between noise and aerodynamic efficiency. They also reported that a rotor combining acoustically favourable taper and twist changes achieved about 7 dB noise reduction relative to the baseline, with a 15% reduction in power loading.

Torija et al. [38] investigated contra-rotating UAV propellers experimentally through far-field acoustic measurements and psychoacoustic analysis, varying the axial spacing  $z/D$  from 0.05 to 1 at a thrust setting

of 10 N. The authors found that the measured spectra were dominated by tonal components between about 0.1 and 2 kHz, and that the potential-field interaction tones weakened rapidly as the spacing increased from  $z/D = 0.05$  to 0.2. They also found that psychoacoustic annoyance was lowest for  $z/D = 0.2$  to 0.4, whereas at larger spacings broadband interaction noise increased.

With respect to UAMV noise prediction and experimental validation, Han et al. [39] developed a noise-prediction approach for multi-rotor UAVs based on an RPM fluctuation correction method to account for frequency modulation caused by small rotor-speed variations in flight. The method was assessed using experiments on a single isolated rotor and on a full vehicle. The authors found that the correction improved the prediction of the first blade-passing-frequency component and the associated spectral spreading compared with a fixed-RPM approach, although they also noted that it could not capture loading increases caused by strong blade-wake interaction. Kostek et al. [40] carried out an experimental and computational cross-validation study on a single fixed-pitch UAV rotor operating under varying flight parameters. The aerodynamic tools ranged from blade element theory to Navier–Stokes solvers, and the resulting loadings were used for noise prediction at the blade-passing frequency on a plane below the rotor. The authors found satisfactory agreement between computations and experiments for both forces and noise levels, and also reported that scattering by the ground and rotor platform was observable even at low frequencies.

With respect to full vehicles, Intaratep et al. [41] carried out an experimental study on a DJI Phantom II quadcopter at static thrust conditions using a single microphone and a load cell, and also compared one-, two- and four-rotor operation. The authors found that tones at the shaft rate, blade-passing frequency and their harmonics dominated the spectrum up to about 6000 Hz, and that a broadband hump in the mid-frequency range rose by nearly 10 dB above the low-frequency broadband level. They also reported that moving from two non-adjacent rotors to four rotors produced a marked increase in broadband noise. Zawodny and Pettingill [42] carried out acoustic wind-tunnel measurements on a representative quadcopter in hover and forward flight, with forward-flight measurements performed at  $M_\infty = 0.045$  and a vehicle pitch angle of  $\alpha_v = -10^\circ$ . The authors found that hover noise showed strong harmonics of the blade-passing frequency, which they attributed mainly to rotor-airframe interaction. They also found that forward flight produced higher acoustic levels than hover, with the aft rotors operating about 20% faster than the forward rotors in trimmed flight, and with the spectrum becoming mainly broadband above about 1 kHz.

Several studies have focussed on rotor-rotor and rotor-airframe interactions. Wang et al. [43] investigated the aerodynamic and aeroacoustic characteristics of a rotor-airframe system based on a DJI S1000 rotor and a single cylindrical support airframe, considering both top-mounted and bottom-mounted rotor orientations with a constant rotor-airframe spacing of 27 mm. The study used acoustic measurements, PIV and pressure-transducer measurements, and compared them with CFD results that used the IDDES turbulence model. The authors found that the top-mounted case produced an asymmetric wake because half of the wake was blocked by the airframe, whereas the bottom-mounted case showed a nearly symmetric wake. They also reported pressure spikes in the top-mounted case when the blade was collinear with the airframe, while the bottom-mounted case produced stronger blade-passing-frequency harmonics at the in-plane receiver. Heydari et al. [44] carried out a computational study on a quadrotor biplane unmanned aerial system using the  $k - \omega$  SST model for the flow and the Ffowcs Williams–Hawkings equations for the acoustics. The study varied vehicle pitch angle, propeller RPM and flight speed, and evaluated both

far-field noise and pressure fluctuations on the structure. The authors found that the UAS noise increased with pitch angle and propeller RPM, but showed an irregular trend with vehicle speed. They also reported that propeller-propeller interaction dominated at small pitch angle, whereas propeller-structure interaction became more important at large pitch angle. Park and Jee [45] numerically investigated the interactional aerodynamics of a quadcopter in hover using an overset-mesh OpenFOAM approach with incompressible URANS and the SA-noft2-R Spalart-Allmaras turbulence model. The authors found that adjacent rotors generated  $\Omega$ -shaped vortical structures from rotor-rotor interaction, and that the fuselage deflected the wakes of the four rotors toward the vehicle centre. They also showed that the overall quadcopter thrust exhibited a major 2/rev fluctuation linked mainly to rotor-rotor and rotor-fuselage interaction, while the 8/rev component was the second-largest fluctuation and was associated with the multiple rotors and the presence of the fuselage.

Aziz and Shi [46] used the virtual blade method for the aerodynamics and an FW-H-based numerical code for the acoustics to compare conventional, coaxial and hybrid octocopter configurations in hover and steady forward flight, and also examined the effect of vertical spacing in the coaxial case. For normal-sized rotors, the authors found that in hover the coaxial configuration was the loudest, with a peak SPL of 93 dB, and that reducing the vertical spacing increased the noise further, with the 0.125R case about 10 dB louder than the 1.5R case. In steady forward flight, they reported that the hybrid configuration gave the lowest peak SPL, 85 dB, about 7 dB below the conventional configuration.

For a ducted contra-rotating-rotor UAV in hover, Zhao et al. [47] developed a hybrid computational aeroacoustic method that projected the aeroacoustic sources onto the in-duct modal space and linked them to the far field through duct modes. The authors found that rotor loading noise was the main contributor to the far field, with rotor-rotor interaction dominating at even harmonic orders of the blade-passing frequency, whereas at odd harmonic orders rotor-strut interaction dominated through the lower-order duct modes. They also reported generally good agreement between the predicted and measured far-field directivity.

With respect to the propagation and perception of UAMV noise, Kloet et al. [48] measured the acoustic signature of a 2.1 kg small multi-rotor unmanned aircraft system in outdoor tests by varying observer distance, altitude and orientation, and also used an acoustic array to identify the principal sound sources. The authors found that the sound level decayed approximately in line with a 6 dB(A) reduction for a doubling of distance, and that the sound profile in the propeller plane did not vary significantly around the vehicle. They also reported that, for observers underneath the vehicle, the SPL varied by as much as 10 dB(A), with the greatest level at about 45° from horizontal. Christian and Cabell [49] carried out an initial psychoacoustic investigation of small unmanned aerial system noise using field recordings and human-subject testing. The study compared annoyance responses for recorded sUAS sounds and road-vehicle sounds using a range of contemporary noise metrics. The authors found that the results suggested a systematic difference between annoyance responses to sUAS noise and road-vehicle noise when judged using those metrics. They also showed, within the limits of the test, that changing height alone did not remove this difference in annoyance response. Barbarino et al. [50] carried out a computational aeroacoustic study of a VTOL aircraft operating near an urban area using a free-wake aerodynamic model, FW-H, and a BEM-FMM solver for acoustic scattering. The authors found that the FW-H and BEM



approaches gave very similar predictions of the incident acoustic field, whereas low-fidelity ray tracing provided only qualitative shielding estimates when near-field effects and complex building geometries were important. They also reported sound pressure level reductions of around 5 dB in building shadow zones.

From these studies, it can be seen that UAMV noise is influenced by operating conditions, rotor design and rotor-rotor and rotor-airframe interactions, with the vehicle's acoustic signature containing both tonal and broadband contributions. With noise being one of the major barriers to commercial viability, these contributions need to be assessed and mitigated.

## 2.4 Boundary Layer Tripping

Boundary layer tripping involves deliberately disturbing the laminar boundary layer on a surface to promote its transition to turbulence. This approach is frequently applied in experimental and Computational Fluid Dynamics (CFD) studies to replicate realistic flow conditions or manage aerodynamic behaviour. Tripping methods include sandpaper strips, wires, tapes or other vortex generating structures.

With respect to trip type, Preston [51] tested various configurations, and concluded that a trip strip achieved transition at a lower height than wire-type trips, whilst the suppression of turbulence separation is better achieved using slotted strips. It was also found that the minimum height of the trip should exceed the momentum thickness based Reynolds number of  $Re_\theta = 320$ . For zigzag trips, Zhang et al. [52] found that a trip-height ( $k$ ) based Reynolds number greater than  $Re_k = 200$  induces transition to turbulence, caused by the shedding process of the shear layer of the trip. The critical  $Re_k = 200$  had also been used by Elsinga and Westerweel [53], who also noted that zig-zag trips may be more efficient than wires since the spanwise shed vortices are naturally undulated, enhancing 3-D flow. For an undisturbed boundary layer height, Santos et al. [54] tested ratios of  $k/\delta_k$  between 0.17 and 2.14, and found that a ratio of  $0.5 < k/\delta_k < 1.05$  should be used to ensure a turbulent trailing edge whilst minimising low frequency far-field noise. For equal roughness heights, Ye et al. [55] found that distributed spherical roughness elements were less effective than zigzag trips at low velocities, whilst at higher velocities, these elements were able to move transition even further upstream than the zigzag trip.

One of the first attempts to reconcile numerical data with experimental data was made by Jansen [56] in which LES was conducted on a forward-facing serrated-type trip on a NACA 4412 airfoil. The configuration was intended to match the experiments done by Wadcock [57]. Although computationally expensive at the time, the boundary layer results showed an improvement over the work done by Jansen [58].

For  $Re = 2.32 \times 10^6$  and an angle of attack of  $\alpha = 2.5^\circ$ , Garrec et al. [59] and Gloerfelt and Garrec [60] conducted compressible LES and Direct Noise Computations (DNC) on a truncated NACA 0012 airfoil with a step-type trip. The studies investigated the combined effect of a turbulent boundary layer and vortex shedding from the blunt trailing edge. The results showed reasonable agreement with the ONERA experiments.

Zhang et al. [61] used LES and zonal RANS-LES to investigate the flow over an airfoil-flap configuration at  $Re = 1 \times 10^6$  and  $M = 0.12$ , with the airfoil at  $\alpha = 0^\circ$  and the flap at  $\alpha = 34^\circ$ . For the zonal approach,

RANS modelling was performed in regions with attached boundary layers whilst LES was used in highly unsteady regions. For the LES, temporary numerical perturbations were used to trip the flow and the trips were removed after a turbulent boundary layer was generated. It was found that after removing the trips, the suction side remained turbulent whilst the pressure side relaminarised due to a coarse mesh resolution. For the zonal RANS-LES approach, the Spille and Kaltenbach [62] forcing method was used to generate turbulent inflow in the LES region. The zonal RANS-LES approach showed good agreement with the LES data and experimental data from the SWING+ project, whilst reducing the computational effort by nearly 50%.

Wolf and Lele [63] used compressible LES coupled with the Ffowcs Williams and Hawkings acoustic analogy [29] to investigate a NACA 0012 airfoil with a rounded trailing edge at  $Re = 408,000$ ,  $\alpha = 5^\circ$  and  $M = 0.115$  and  $0.4$ . Tripping was implemented using localised suction and blowing on the suction side of the airfoil only. The SPL spectra for the configuration with  $M = 0.115$  showed good agreement with the experimental work done by Brooks et al. [5], whilst it was reported that nonlinear quadrupole noise sources play an important role in the sound radiated by the configuration with  $M = 0.4$ . Also for a NACA 0012 with a rounded trailing edge, at  $Re = 5 \times 10^5$ ,  $\alpha = 0^\circ$  and  $M = 0.2$ , Bodling and Sharma [64] used implicit LES and the Ffowcs Williams and Hawkings acoustic analogy [29] to compare a baseline airfoil to one with trailing edge finlet fences. Two trips were implemented, one being a geometry-resolved square cylinder over the span of the airfoil and the other being the application of tangential reverse blowing. The authors noted that especially for the resolved trip, it is important to place the permeable FW-H integration surface outside any high frequency waves that emanate from the trip to obtain reliable far-field noise predictions.

The NACA 6512-63 airfoil at  $\alpha = 0^\circ$ ,  $Re = 1.9 \times 10^5$  and  $M \approx 0.063$  subjected to boundary layer tripping has been extensively studied by Winkler et al. [65], Winkler et al. [66] and Winkler et al. [67]. In Winkler et al. [65], a geometry-resolved forward-facing serrated trip and a simplified step-type trip were compared to an untripped configuration. Far-field acoustics were evaluated using Ffowcs Williams and Hall's formulation [68], Amiet's trailing-edge model [69], and Curle's analogy [27]. The step-type trip produced significantly higher wake RMS velocity fluctuations than the serrated trip. Acoustic predictions using Ffowcs Williams and Hall and Amiet showed that the serrated-trip LES agreed best with the medium-to-heavy tripped experimental data, whereas the step-trip LES agreed better with the lightly tripped experimental data. Curle's formulation yielded comparable results to Ffowcs Williams and Hall and Amiet only at low to medium frequencies, but overpredicted the SPL at higher frequencies. Kamliya Jawahar et al. [70] applied a step-type trip to a NACA 6512-63 airfoil at  $\alpha = 0^\circ$ ,  $Re = 1.9 \times 10^5$  and  $M \approx 0.062$ . LES turbulence modelling was coupled with Curle's acoustic analogy [27] for far-field noise prediction. The mean surface pressure distributions and local pressure peaks showed good agreement with experimental data from Winkler et al. [65], whilst aeroacoustic predictions exhibited overall agreement, with slight overprediction at higher frequencies. In Winkler et al. [66], only the serrated-trip configuration was considered. The influence of spanwise grid resolution and spanwise computational extent was investigated, and acoustic predictions were performed using Ffowcs Williams and Hall [68], Howe's vortex sound formulation [71], Amiet's model [69], and a finite-element solution of Lighthill's analogy [72]. Increasing the spanwise resolution allowed for small-scale turbulence to be resolved. Doubling the spanwise domain extent improved the capturing of the largest turbulent structures. Among the tested models, Ffowcs

Williams and Hall's [68] formulation provided the most robust acoustic predictions. In Winkler et al. [67], the serrated and step-type trips were re-examined using Ffowcs Williams and Hall [68] and Amiet's [69] models. Under sufficiently tripped conditions, no noticeable differences were observed in the far-field broadband SPL between the two trip geometries. Both acoustic formulations were reported to be in good agreement with experimental data and compressible DNS results.

It can therefore be seen from both the experimental and computational studies that trip geometry and height are key parameters in forcing transition to turbulence and maintaining a turbulent boundary layer. These parameters affect the resulting flow field and acoustics.

## 2.5 Noise Mitigation Strategies

The following section outlines strategies that have been investigated to mitigate noise in rotors and airfoils.

### 2.5.1 Serrations

#### 2.5.1.1 Leading-Edge Serrations

Noise generation in airfoils is typically caused by an interaction between the airfoil and turbulent flow, such as rotor wake turbulence or atmospheric turbulence. When operating in such flow, broadband noise is generated at the leading-edge of the airfoil by the scattering of vorticity [73]. The comb-like structures (serrations) on the leading-edge of owl wings are believed to play a major role in the silent flight of these birds [74]. As such, there have been many investigations conducted on the application of leading-edge serrations to rotors for the purpose of noise reduction. Some of these investigations are discussed in this section.

One of the first investigations to be carried out on the application of leading-edge serrations to rotors was conducted by Soderman [75]. Two rotors were tested - a small-scale, 2-bladed rotor with a diameter of 1.52 m and a large-scale, 2-bladed rotor with a diameter of 2.59 m. The resulting tip Reynolds numbers for the small blade ranged from  $Re = 183,000$  to  $Re = 550,000$ , and  $Re = 994,000$  to  $Re = 3,180,000$  for the large rotor. The rotors were also tested at various blade angles. The serrations were attached to the pressure side of the blades, near the leading-edge. For the small rotor, the Overall Sound Pressure Level (OASPL) noise reductions ranged between 4–8 Decibels (dB), and between 3–17 dB in the high octave bands. For the large-scale rotor, the maximum OASPL noise reduction achieved was 4 dB. Although the author concluded that the serrations were effective in reducing high frequency noise but not low frequency rotational noise, this may not be an issue since the computational investigation done by Li and Lee [4], found that it is the high frequency noise produced by UAMVs that is not masked by background noise in a community.

In another investigation regarding leading-edge serrations applied to open rotors, Hu et al. [76] conducted an experimental investigation on the aerodynamic and aeroacoustic properties of seven multi-copter rotors with leading-edge serrations under forward flight conditions, and compared the results to the baseline rotor. The rotor had a profile of a NACA 2416 airfoil for  $r/R < 0.2$  and an A18 airfoil for  $r/R \geq 0.2$ .

The rotors were tested at 8 rotational speeds between 3,000 and 7,200 Revolutions Per Minute (RPM), and the maximum tested wind tunnel flow speed was 100 m/s. The resulting Reynolds number at a position 0.75R based on rotor chord length and tangential velocity of the blade was therefore between  $Re = 25,000$

and  $Re = 60,000$ .

The following conclusions were drawn:

- All serrated rotors produced less thrust than the baseline whilst increasing torque
- Torque and power consumption increase with an increasing amplitude and decrease with an increasing wavelength.
- Serrations significantly reduce tonal noise and mid-high frequency broadband noise at 1 Blade Pass Frequency (BPF).
- Noise reduction is more sensitive to amplitude rather than wavelength, although larger wavelengths result in larger broadband noise reduction
- Serration amplitude-to-wavelength is potentially a critical parameter to determine aerodynamic and aeroacoustic performance of rotors with leading-edge serrations.

Although the investigation was conducted on relatively small rotors ( $R = 0.103$  m) and at low Reynolds numbers (between  $Re = 25,000$  and  $Re = 60,000$ ), which would not be the case for a full-scale UAMV, valuable conclusions were drawn with respect to the effect that amplitude and wavelength have on both the aerodynamic and aeroacoustic properties of rotors.

The Strouhal number is a dimensionless value used for analysing oscillating transient flow motion. It is defined as the ratio of inertial forces caused by the flow's local acceleration to the inertial forces caused by convective acceleration [77]. In the case of leading-edge serrations, the Strouhal number can be given by Equation 2.38:

$$St_h = \frac{fh}{U} \quad (2.38)$$

For Equation 2.38:

$St_h$  | Strouhal number     $f$  | Frequency of oscillations     $h$  | Serration Amplitude     $U$  | Flow speed

In the case described above, Hu et al. [76] found that for the same rotation speed, the noise reduction for all the tested serrated rotors exhibited a linear logarithmic relation with the Strouhal number, given by  $\Delta SPL \sim C \log_{10} St_h$ , until reaching the maximum.

Wei et al. [78] conducted both computational and experimental investigations for rotors with five different owl-inspired serration shapes, and compared the results to the baseline rotor (P17  $\times$  5.8 carbon fibre propeller). The rotors were tested at RPMs ranging between 2,000 - 3,000 RPM, resulting in the tip Reynolds number ranging between  $Re = 1.7 \times 10^5$  and  $Re = 2.7 \times 10^5$ . For the computational case, a 3 million element cylindrical domain was used with a rotational component, making use of a steady Reynolds-averaged Navier-Stokes (RANS) simulation and using the  $k - \omega$  Shear Stress Transport (SST) turbulence model to close the Navier-Stokes equation. Two separate experiments were carried out, one being a single-rotor testing in a hemi-anechoic chamber and the second involved attaching the rotors to a drone, and measuring the noise radiation for the drone hovering in an outdoor environment at 5 m and 8 m altitude. Wei et al. [78] found that the leading-edge serrations act as vortex generators which bring about

changes on the suction side of the propeller with respect to the laminar-turbulent transition and turbulence distribution on the surface. Symmetrical serrations such as sawtooth serrations were more effective at generating vortices. Sawtooth serrations also performed the best compared to other serrations, resulting in a maximum of 4.18 dB noise reduction, and was able to suppress noise over frequencies ranging from 0.2 Kilohertz (kHz) to 12.5 kHz. Although the authors claim that serrations can reduce noise whilst enhancing thrust force, it is important to note that according to Hu et al. [76], at the same RPM, serrated rotors were found to consume more power compared to the baseline rotor.

Chong et al. [79] conducted an experimental aeroacoustic investigation on a NACA 65(12)-10 airfoil subjected to 12 different leading-edge serrations, with varying wavelengths ( $\lambda$ ) and amplitudes ( $A$ ). The serrations were cut directly into the body of the airfoil rather than being added onto the airfoil, thus maintaining the airfoil chord length. The investigation was conducted for chord-based Reynolds numbers ranging between  $Re = 2 \times 10^5$  and  $Re = 6 \times 10^5$ , corresponding to jet speeds ranging from 20 m/s to 60 m/s respectively. The authors determined that the most effective serration configuration was the one with the largest amplitude and smallest wavelength. The effectiveness of the serrations in suppressing four different types of noise sources was investigated and the following was found:

- Laminar instability tonal noise - was reduced due to the streamwise vortices at the leading-edge suppressing the separation bubble at the trailing-edge.
- Leading-edge separation bubble noise - was reduced based on the geometry of the serrations itself
- Turbulence leading-edge interaction noise - was reduced by the streamwise vortices generated by the serrations. However, in this case, superfluous noise was generated at high frequencies.
- Trailing-edge self-noise - was reduced by the leading-edge serrations disrupting the hydrodynamic growth of the turbulent boundary layer at the trailing edge.

With respect to the aerodynamic influence of leading-edge sawtooth serrations, Ito [80] conducted an experimental investigation by attaching three different types of jigsaw blades to the leading-edge of a NACA63-414 airfoil. The chord-based Reynolds numbers tested were  $Re = 2.1 \times 10^4$  and  $Re = 2.1 \times 10^5$ , corresponding to a freestream velocity of 2 m/s and 20 m/s respectively. It was found that serrations appeared to be more effective at the lower tested Reynolds number, and that lift characteristics improve with the use of finer serrations. Similar to the work done by Wei et al. [78], the serrations were found to control flow separation on the suction side of the airfoil at the angle of attack in which separation occurred in the baseline airfoil.

A general trend that can be noticed with respect to sawtooth serrations is that high amplitude, low wavelength serrations have a positive impact on both the noise reduction and aerodynamic performance of the airfoil, although for the purpose of broadband noise reduction, larger wavelengths are more beneficial.

Sinusoidal leading-edge serrations for the purpose of airfoil noise reduction have also been studied in much detail. Narayanan et al. [81] conducted an experimental investigation by applying sinusoidal serrations of wavelengths  $\lambda$  and amplitudes  $2h$  to a flat plate. The chord length of the serrated flat plate was given by Equation 2.39.

$$c(r) = c_0 + h \sin(2\pi r/\lambda) \quad (2.39)$$

For Equation 2.39:

$$c_0 \mid \text{Mean chord} \quad h \mid \text{Amplitude} \quad \lambda \mid \text{Wavelength} \quad r \mid \text{Spanwise distance}$$

Twenty combinations of wavelengths and amplitudes were tested at flow velocities of 20, 40, 60 and 80 m/s. It was found that noise reductions were insignificant at low frequencies but in the mid-frequency range (500 Hertz (Hz) to 8 kHz), noise reductions were significant. The maximum noise reduction for the serrated flat plates was approximately 9 dB, significantly higher than the maximum noise reduction for the same serrations applied to a NACA-65 airfoil, which was found to be approximately 7 dB. Similar to other investigations, noise reduction was found to be more sensitive to serration amplitude rather than wavelength. In addition, the authors also conducted a computational investigation for two of the leading-edge serration profiles using a  $\approx 3.1$  million point grid, using compressible Euler equations and the Ffowcs Williams and Hawkings (FW-H) analogy to solve the far-field pressure. There was good agreement between the computational and experimental results up to a frequency of 1 kHz.

Biedermann et al. [82] applied five different sinusoidal serrations to a NACA65(12)-10 airfoil. Similar to the work done by Chong et al. [79], the serrations were cut into the airfoil to maintain the chord length. Tests were carried out for  $Re = 350,000$  and a beamforming approach was used so that the aeroacoustic results were not influenced by external noise sources. With respect to aerodynamic characteristics, it was found that all serrations showed an increased drag coefficient compared to the baseline, whereas lift was positively affected by larger amplitude and wavelength serrations. However, high amplitudes and small wavelengths were found to be more beneficial for noise reduction purposes. The maximum noise reduction for the 2 kHz octave band was found to be 13.6 dB. It was also found that  $\Delta SPL = 15 \log_{10}(St_h) + b$  where  $b$  was found to be 10 for the smallest serration wavelength. The corresponding serration wavelength to transverse turbulent length scale ratio ( $\lambda/\Lambda_t$ ) was 2.6. As mentioned earlier, Hu et al. [76] found a similar linear logarithmic relationship between the noise reduction and the Strouhal number.

### 2.5.1.2 Trailing-Edge Serrations

Trailing edge serrations have been applied to and investigated with respect to flat plates, airfoils and propeller blades.

For NACA 0012 airfoils, Chong and Joseph [83] performed an experimental study on a NACA 0012 airfoil with trailing-edge serrations at  $Re = 1 \times 10^5$  to  $6 \times 10^5$ . The work focused on instability tonal noise, with particular attention to an effective angle of attack of about  $\alpha_{eff} = 4.2^\circ$ . The authors found that the serrations could suppress or strongly reduce the discrete tones, and they stated that the laminar separation bubble had to lie within the serration region for the control to be effective. Chong et al. [84] used experiments to investigate a NACA 0012 airfoil with nonflat plate sawtooth trailing-edge serrations at  $Re = 2 \times 10^5$  to  $6 \times 10^5$  and an effective angle of attack of about  $\alpha_{eff} = 4.2^\circ$ . The study considered both instability tonal noise and turbulent broadband self-noise. The authors reported that cutting the serrations directly into the airfoil introduced narrowband vortex-shedding noise due to the bluntness at the serration roots, but they also showed that some of the geometries still produced significant tonal-noise reduction

and moderate broadband-noise reduction across a fairly wide velocity range. Vathylakis et al. [85] carried out an experimental study on a NACA 0012 airfoil with poro-serrated trailing edges at  $Re = 2 \times 10^5$  to  $6 \times 10^5$  and  $\alpha = 0^\circ$ . Porous material was placed in the gaps between adjacent sawtooth members in order to suppress the bluntness-induced vortex shedding associated with the nonflat plate serration concept. The authors found that the poro-serrated configurations could suppress the vortex-shedding tones while still preserving broadband trailing-edge noise reduction, and they reported reductions of up to 7 dB without significant loss of efficiency.

For NACA 633-018 airfoils, Luesutthiviboon et al. [86] presented an experimental benchmark study for a NACA 633-018 airfoil with straight and serrated trailing edges over  $Re_c = 2 \times 10^5$  to  $3 \times 10^6$ , with varying angle of attack, serration shape and serration flap angle. Two airfoil models were tested in two different wind tunnels, and both natural and forced transition were considered for the straight trailing edge. The authors found that forcing transition increased the straight trailing-edge noise by up to 5 dB, and for the serrated cases they reported that the noise reduction depended on airfoil lift and Reynolds number, with the iron-shaped serrations giving greater attenuation than the conventional sawtooth serrations. Lima Pereira et al. [87] carried out an experimental parametric study on a NACA 633-018 airfoil with serrated trailing edges at  $Re_c = 2.0 \times 10^5$ ,  $2.7 \times 10^5$  and  $4.0 \times 10^5$ , using several angles of attack. The study varied serration height, wavelength and aspect ratio about a reference sawtooth geometry, and also examined concave-shaped and combed-sawtooth serrations. The authors reported that increasing serration height improved the noise reduction only up to a limit, whilst the serration wavelength mainly affected the frequency range over which noise reduction was obtained, with wider serrations reducing more noise than narrower ones under the symmetric conditions tested. They also found that aerodynamic loading degraded the high-frequency performance, and that the concave-shaped and combed configurations could outperform the reference sawtooth geometry under the conditions tested.

Numerically, Avallone et al. [88] used compressible lattice-Boltzmann simulations together with the Ffowcs Williams and Hawkings acoustic analogy to investigate a NACA 0018 airfoil with straight, sawtooth and combed-sawtooth trailing edges at  $Re = 280,000$ ,  $M = 0.06$  and  $\alpha = 0^\circ$ . Boundary-layer transition was forced on both sides using a zig-zag strip. The authors reported that the sawtooth serrations gave about 6 dB noise reduction relative to the straight trailing edge, while the combed-sawtooth serrations gave an additional 3 dB reduction. They also found that the reduction depended on frequency, with no noise reduction at higher frequencies, and attributed the effect mainly to destructive interference of the scattered pressure waves. Ahn et al. [89] used LES and spectral proper orthogonal decomposition to investigate a NACA6512-63 airfoil with sawtooth serrations at  $Re = 1.35 \times 10^6$  and  $M = 0.147$ . Three serration geometries were considered with  $\lambda/h = 0.15$ , 0.3 and 0.6. The authors found that the serration angle altered the 3-D flow near the trailing edge, and they reported that the  $\lambda/h = 0.3$  case gave the most effective noise reduction, whilst the  $\lambda/h = 0.6$  case showed the weakest reduction effect.

With respect to rotating blades, for a full-scale 94 m diameter wind turbine, Oerlemans et al. [90] carried out field measurements on one baseline blade, one blade fitted with trailing-edge serrations, and one blade with an acoustically optimised airfoil shape. The authors found that both modified blades reduced the low-frequency trailing-edge noise, with the serrated blade giving the larger reduction, and they reported an average overall noise reduction of 3.2 dB for the serrated blade. They also noted that the modified

blades showed high-frequency tip noise, which was most important at low wind speeds.

Ning and Hu [91] carried out an experimental study in hover on two baseline propellers and four small propellers with sawtooth trailing-edge serrations, over rotational speeds up to 7000 rpm. Aerodynamic forces, sound pressure levels and flow structures were measured using a load cell, acoustic measurements and PIV. The authors found that the sawtooth serrations increased the trailing-edge vorticity and reduced the broadband noise by up to 5 dB. They also reported that the increase in trailing-edge vorticity was significant near the tip and small near the root. Lee et al. [92] carried out an experimental study in an anechoic chamber on a propeller with quarter flat-tip, half flat-tip and rectangular trailing-edge serrations, over rotational speeds from 1500 to 3000 rpm. Noise was measured at four microphone positions, and the thrust force was also recorded. The authors found that the half flat-tip serrated propeller gave the best overall acoustic performance, with overall noise reductions of up to 5.8 dBA, whilst the rectangular serrated propeller was only effective at 2000 rpm and the quarter flat-tip serration performed poorly. They also reported that the half flat-tip serrated propeller had a 16.8% lower thrust penalty than the rectangular serrated propeller. Yuliang et al. [93] used experiments to investigate a baseline  $P17 \times 5.8$  propeller together with flaplet, concave and sawtooth trailing-edge serrations, and also examined sawtooth serrations with different distribution lengths, over rotational speeds from 2000 to 3000 rpm. The study considered both acoustic and thrust performance. The authors found that all of the serrated propellers reduced noise relative to the baseline propeller, and that the sawtooth serration gave the best overall noise-reduction performance among the three shapes, particularly at the higher Reynolds number tested. They also reported that the serration distribution length had to be chosen appropriately, with the 3/4-DL propeller giving the best overall performance, namely an average noise reduction of 2.7 and 3.0 dB at the two tested positions together with a 6.7% improvement in aerodynamic performance. Candeloro et al. [94] carried out an experimental study on serrated trailing-edge drone propellers based on an APC  $9 \times 4e$  propeller. Acoustic measurements in hover were performed at  $\Omega = 4000$  rpm, and further aerodynamic and aeroacoustic measurements were performed in an anechoic wind tunnel at  $\Omega = 5000$  rpm and different advance ratios. A total of 23 propellers were designed and tested in order to relate serration geometry to noise mitigation. The authors found that, with an appropriate choice of serration geometry, both tonal and broadband noise could be reduced, by about 3 dB and 4 dB, respectively. They also reported that this came with a drawback, namely a nearly 20% loss in thrust coefficient during advanced flight, although there was also about a 20% reduction in energy consumption.

Numerical studies have also been conducted on serrated trailing-edge propellers in conjunction with experiments. Gu et al. [95] combined CFD, LES and the Lighthill acoustic analogy with experiments to investigate a DJI 9450S propeller with various owl-inspired serrated trailing-edge configurations, including different serration ratios, shapes, coverage degrees and materials. The authors found that serrated trailing edges reduced the propeller noise, with the best performance obtained for a half-covered round serration with serration ratio 0.9 and round radius 0.3 mm, which gave a maximum reduction of 4.4 dBA at  $T = 350$  g. They also reported that the serrations disrupted the large-scale tip vortices, accelerated vortex shedding and dissipation, and promoted stronger wake mixing, whilst sharp serrations could introduce new spectral peaks that weakened the overall reduction compared with the round serrations. Yuliang et al. [96] used numerical simulations and experiments to investigate a commercial T-motor P1758 propeller with leading-edge serrations, trailing-edge serrations, and a coupled owl-inspired leading-edge/trailing-edge



configuration. For the simulations, the chord-based Reynolds number at 3000 rpm was reported as  $Re = 1.5 \times 10^5$ . The authors found that the coupled propeller outperformed the single-structure serrated propellers, achieving a maximum noise reduction of 3.1 dB together with a 7.9% improvement in thrust efficiency. They also reported that the coupled configuration suppressed both the tonal noise, particularly at 1 BPF, and the broadband noise above 2 kHz, whilst in outdoor hovering tests on a quad-rotor UAV it reduced the overall noise by about 2.6 to 3.8 dB. Mane et al. [97] used CFD with the SST  $k - \omega$  model and the Ffowcs Williams and Hawkings acoustic analogy to compare single-, two- and three-blade UAV propellers, each with and without bio-mimetic serrations, in a low-Mach-number regime. The authors found that increasing blade count reduced the dominant tonal noise by redistributing the aerodynamic loading, whilst the bio-mimetic serrations redistributed acoustic energy and reduced tonal sharpness. Among the cases considered, the three-blade serrated propeller gave the lowest sound pressure levels at 2500 and 5000 Hz, and the numerical results were reported to be in good agreement with published experimental data.

It can therefore be seen that both leading-edge and trailing-edge serrations have potential to reduce tonal and broadband noise, although this generally results in aerodynamic performance penalties. For an EUAMV, with the current battery technology still having a significantly lower energy density compared to conventional fuels, sacrificing aerodynamic performance for noise reduction may not be desirable for manufacturers.

### 2.5.2 Winglets

A winglet can be defined as “a vertical or inclined endplate attached to a wing tip to block some of the leakage of air from the high-pressure to the low-pressure side, thereby reducing the strength of wing-tip vortices and induced drag” [98]. For a subsonic aircraft under optimum cruise conditions, about half of the total drag can be attributed to the trailing vortex system. The application of winglets to the wings of aircraft, more specifically blended winglets, have been found to be effective in reducing induced drag, and thus an engine noise reduction from reduced take-off thrust [99].

Whitcomb [100] conducted experimental investigations on a winglet applied to an aircraft wing at Mach 0.78. It was found that the winglets reduced induced drag by approximately 20%, and increased the wing lift-to-drag ratio by approximately 9%, which was more than double the improvement in the lift-to-drag ratio compared to extending the wing tips. Kroo [101] found that for a winglet of height  $h$ , a 45% reduction in tip loss effects could be achieved compared to extending the wing spanwise by the same height  $h$ .

However, when it comes to rotational applications, aerodynamic and aeroacoustic investigations regarding winglets have been confined to wind turbines and axial/diagonal fans. Nevertheless, the findings could be applicable to open rotors, and are therefore discussed in this subsection.

With respect to the impact of winglets on aerodynamic performance, Imamura et al. [102] used a vortex lattice method to determine the aerodynamic effects of applying winglets to wind turbine rotors. The airfoil used for the rotors was a NACA 0012 airfoil which is not common for wind turbine rotors. The winglets were tested at various installation angles ( $\phi$ ), where  $\phi = 0^\circ$  represents a winglet perpendicular to the rotor and  $\phi = 90^\circ$  represents a wing extension. It was found that for smaller installation angles, a greater enhancement in the power coefficient was achieved. The winglets also increased the flatwise

bending moment at the root of the blade, which increased with increasing  $\phi$ . The flatwise bending moment at the tip of the blade decreases with decreasing  $\phi$ . It was concluded that the increase in efficiency was caused by a reduction in the axial induced velocity, and an increase in circulation near the blade tip.

Johansen and Sørensen [103] used CFD to test the aerodynamic effects of five winglets with varying twist distribution and camber, one of which was pointed towards the suction side of the blade. An incompressible, steady-state solver was used with the  $k - \omega$  SST turbulence model. The heights of the tested winglets were approximately 1.5% the rotor radius. The tested winglets increased the force production on the outer 14% of the blade. Furthermore, the winglets produced additional power ranging from 0.6% to 1.4% and increased thrust by between 1.0% and 1.6%. The winglet that was pointed towards the suction side produced even more power. Although the additional thrust produced may pose a loading issue for wind turbines, it would be a positive outcome if applied to UAMVs. As mentioned by the authors, the best performing winglet was the one pointed towards the suction side, which would cause clearance issues in a wind turbine.

Johansen and Sørensen [104] followed on from the work done by Johansen and Sørensen [103], and tested the aerodynamic effects of curvature radius, sweep, twist and winglet height. For this investigation, all winglets were pointed towards the suction side. This time, the increase in power production ranged between 1.0% and 2.8% and the increase in thrust ranged between 1.2% and 3.6%. The following conclusions were drawn:

- Decreasing curvature radius results in increasing mechanical power and thrust
- A 30° backwards sweep of the winglet does not increase mechanical power
- There is only a small dependence on winglet twist
- Increasing winglet height increases mechanical power and thrust

The best winglet was found to be the largest winglet, however, the authors suggested that the high aspect ratio may not be structurally feasible considering the high loading on the winglet.

Elfarra et al. [105] used RANS simulations with the  $k - \epsilon$  Launder-Sharma turbulence model to optimise a winglet design for a 2-bladed wind turbine rotor. The rotor blade used was the National Renewable Laboratory (NREL) Phase VI wind turbine blade, and three different wind speeds were tested. For the optimised winglet, a power increase of 9% was achieved compared to the baseline rotor, considerably larger than that obtained by Johansen and Sørensen [104]. The power increase was attributed to a combination of an increase in the normal and tangential forces at the tip region, and the flow being more attached at the tip.

With respect to the aeroacoustics of winglets applied to wind turbines, Ebrahimi and Mardani [106] conducted CFD analysis on the NREL Phase VI wind turbine with winglets applied. The  $k - \omega$  SST turbulence model was used and the FW-H analogy was implemented for noise prediction. It was found that the winglet had no perceptible effect on noise production at low frequencies but was effective in reducing noise emissions at high frequencies. With the winglet applied, a 10 dB noise reduction was achieved at 200 Hz. However, Arakawa et al. [107] advised that a RANS simulation cannot supply the pressure fluctuations needed to predict aerodynamic broadband noise adequately, therefore, although the results of the work done by Ebrahimi and Mardani [106] looks promising for the application of winglets for noise

suppression, more accurate computational methods need to be applied to further investigate this.

When applied to cooling fans, Nashimoto et al. [108] found that winglets reduced the aerodynamic noise by 15%. A maximum reduction of 1 dB was achieved for the frequency range under 6 kHz, although the noise produced tended to increase in the frequency range above 6 kHz. The noise reduction was attributed to a decrease in the tip vortex speed with winglets applied, as well as a reduction in the size of the vortices produced with winglets applied.

Zhang et al. [109] investigated the effects of applying winglets to a diagonal fan using CFD. A steady-state RANS simulation with the  $k - \epsilon$  turbulence model was used to initialise a Large Eddy Simulation (LES) based on the Smagorinsky-Lilly model. This work was followed up by Zhang et al. [110], where the same approach was used to determine the effect of winglets pointing towards both the pressure and suction surface, the suction surface only and the pressure surface only. This case considered a shrouded fan, therefore the mechanism of noise reduction may not be strictly applicable to open rotor configurations. Nevertheless, the following conclusions were drawn:

- At the blade tip, the blended blade tip and winglet weakened the intensity of the main, secondary and leakage flows
- The blended blade tip and winglet reduced the tip leakage vortex intensity, which in turn reduced the SPL of the near and far-field noise
- Winglets pointed to the suction side had the greatest impact on the two points mentioned above
- Winglets pointed to the suction side were found to effectively reduce middle and low frequency band noise, which in turn brought about a significant reduction in far-field noise

Compared to other noise mitigation strategies, when applied to wind turbines and axial/diagonal fans, winglets have been found to bring about significant noise reductions whilst enhancing aerodynamic performance at the same time. However, there is currently a gap in knowledge that exists with respect to the application of winglets for open rotor configurations, and its effects still need to be determined for such configurations.

### 2.5.3 Porous Coatings

There has been increasing interest in using passive flow control techniques such as porous coatings on bluff bodies, largely because of the need to reduce the noise they generate, such as that from landing gear struts, wind turbine structures and heat exchangers.

For randomised or foam coatings, Sueki et al. [111] experimentally tested urethane and metal coatings at  $Re = 8.3 \times 10^4$ , and found that for porous coated cylinders, there was a wide region of zero velocity in the wake. The noise reduction mechanism was attributed to the deprivation of momentum in the wake and the suppression of the unsteady motion of vortices. It was also found that the noise reduction is independent of the hardness of the porous material [111], [112]. For  $0.8 \times 10^5 < Re < 1.6 \times 10^5$ , Arcondoulis and Liu [113] investigated polyurethane coated cylinders with varying porosity ( $\phi = 0.8, 0.9$  and  $0.95$ ) and showed that the coated cylinders shift the vortex shedding Strouhal number. By equating the coated cylinder Strouhal number to that of the bare cylinder, an effective diameter increase of approximately 8% compared to the bare cylinder was inferred. Significant reduction in the vortex shedding tonal noise was also observed in the tested Reynolds number range. For  $1.6 \times 10^4 \leq Re \leq 1 \times 10^5$  [114], and later up to  $Re \approx 2.4 \times 10^5$  [115], cylinders with porous coatings spanning a wide range of air flow resistivities were investigated, and it was found that materials with low air flow resistivity yielded a more pronounced noise reduction, primarily through a narrowing of the aeolian tone spectral peak. Similar trends were confirmed by Geyer [116]. A reduction in broadband noise was also observed, particularly at higher Reynolds numbers.

Structured porous coatings offer better control of pore geometry and arrangement, and possibly easier numerical modelling compared to randomised porous coatings. Arcondoulis et al. [117] designed and tested a 3-D printed coating that showed similar acoustic behaviour to metal foam and polyurethane coatings for the same Pores Per Inch (PPI). For  $0.8 \times 10^5 < Re < 2 \times 10^5$ , the SPCC was able to significantly reduce vortex shedding tone and slightly reduce the average vortex shedding Strouhal number (from  $\approx 0.18$  to  $\approx 0.171$ ) compared to the bare cylinder over the tested  $Re$  range [117], [118]. Using the same base SPCC, for  $Re = 10^5, 1.53 \times 10^5$  and  $1.66 \times 10^5$ , Arcondoulis et al. [119] tested circumferentially varying porosity SPCCs and found two distinct tones at  $St = 0.2$  and  $St = 0.4$ , where the magnitude of these tones was determined by the overall porosity rather than the orientation. It was also found that a lower porosity on the windward side increases high-frequency broadband noise contribution. Similar findings were obtained by Arcondoulis et al. [120] for SPCCs that had varying porosity both circumferentially and in the spanwise direction. Water tunnel tests were also conducted on transparent SPCCs with square-shaped pores ( $PPI = 4.6$  and  $\phi = 0.81$ ) and elliptical-shaped pores ( $PPI = 5.1$  and  $\phi = 0.87$ ), for  $Re = 0.9 \times 10^4$  and  $Re = 7.3 \times 10^4$  ([121]). It was found that the SPCC with square-shaped pores exhibited greater recirculation through the leeward side of the SPCC, and the SPCC with elliptical-shaped pores exhibited a smaller reversed flow region in the wake.

For  $6.1 \times 10^4 < Re < 1.5 \times 10^5$ , Bathla and Kennedy [122] tested various lattice-type cylinder coatings, and found that for a constant porosity, the different coating types yielded similar turbulence intensity and mean velocity in the wake. It was also found that a higher porosity ( $\phi \geq 0.9$ ) gave notable reduction in wake turbulence intensity and mean velocity, and the optimal ratio of coating thickness, ( $t$ ), to bare cylinder diameter, ( $d$ ), was found to lie within  $0.45 \leq t/d \leq 0.65$ .

Numerically, for  $Re = 8.2 \times 10^4$ , Wen et al. [123] used 2-D structured resolved simulations with the  $k - \omega$  RANS model and the FW-H analogy to investigate the effect of porosity ( $\phi = 0.72$  to  $\phi = 0.98$ ) on porous coated cylinders. The porous region was modelled as an array of smaller cylinders around the larger bare cylinder. It was found that increased porosity yielded a greater reduction in OASPL, and all the tested coatings achieved a decreased OASPL compared to the bare cylinder. The use of RANS modelling for acoustics is known to have limitations, although a comparison between cases may capture relative trends in OASPL rather than actual acoustic levels. For  $Re = 4.7 \times 10^4$ , Liu et al. [124] used 2-D Large Eddy Simulation (LES), and the FW-H analogy to test coatings with various porosity, pore density and porous layer thickness. The porous region was modelled using the Ergun equation [125]. It was found that increasing these parameters reduced the OASPL, with porosity having the strongest influence. For simulations conducted at higher flow speeds, the porous coatings achieved greater noise attenuation. For  $Re = 1.6 \times 10^4$ ,  $7.26 \times 10^4$  and  $1.06 \times 10^5$ , Geyer et al. [126] used 2-D Detached Eddy Simulation (DES), with Darcy–Forchheimer-type porous modelling and FW-H acoustic methods to investigate materials with varying air resistivities. The simulations predicted that fluid enters the porous region and causes vortex shedding from both the outer diameter and the inner cylinder. The inertial resistance term was mentioned, but its formulation and coefficient values were not explicitly reported. Although the use of 2-D LES and DES is questionable, these studies provide valuable insight into the noise reduction mechanisms and internal flow features of porous coated cylinders.

The 3-D LES conducted by Yuan et al. [127] using Darcy–Forchheimer-type modelling and the FW-H analogy showed similar trends to Liu et al. [124] in that increasing porosity reduced OASPL, although the effect was comparatively weak for the tested porosity range  $\phi = 0.85$  to  $\phi = 0.97$  (within 1 dB). In contrast to Liu et al. [124], it was found that increasing pores per inch increased OASPL. An optimal thickness ratio of  $t/d = 0.4$  was identified, where the OASPL was minimised, with both thicker and thinner coatings resulting in a higher OASPL. For  $Re = 4.7 \times 10^4$ , Zhang et al. [128] used 3-D LES and the FW-H analogy to investigate circular cylinders with non-uniform porous coatings, with the porous region being modelled by the Ergun equation. Uniform high porosity coatings ( $\phi = 0.95$ ) significantly reduced drag, lift fluctuations and the OASPL. The mechanism was found to be the suppression of vortex shedding and an increase in back pressure. Lower porosity coatings were found to be less effective. For  $Re = 1.27 \times 10^5$ , Sharma et al. [129] used 3-D LES and the FW-H acoustic analogy to test coatings with two different air resistivities. The porous zone was modelled using the Darcy–Forchheimer equation. It was found that increasing the coating thickness reduced the radiated noise for the low air resistivity material whilst the higher resistivity material showed limited noise reduction due to limited entrainment of fluid through the porous region.

For both the experimental and numerical studies, porosity appears to be the key parameter controlling the reduction in OASPL, and for CFD, the FW-H acoustic analogy appears to be the most commonly used approach to predict far-field acoustics.

## 2.6 Summary, Aims and Objectives

The literature relating to UAMVs indicates that rotor-rotor and rotor-airframe interaction noise are important contributors to the vehicle's acoustic signature, producing both tonal and broadband noise. The propeller and full-vehicle studies generally investigate the noise associated with their respective configurations,

and with the exception of leading-edge serrations, much less attention has been paid to interaction noise mitigation using passive noise reduction techniques. Trailing-edge serrations, although studied extensively, primarily targets airfoil-self noise mechanisms.

Porous coatings have demonstrated considerable potential for cylinder noise reduction, with mechanisms attributed to deprivation of momentum in the wake and the altering of vortex-shedding behaviour. However, these studies predominantly consider uniform inflow conditions. There also remains a gap with respect to the utilisation of porous coatings applied to an airfoil leading edge to mitigate vortex-airfoil interaction noise. These two gaps, namely, vortex interaction with a porous coated cylinder and an airfoil leading edge, form the motivation for this research. Based on these gaps, the aim and objectives of this research are outlined below:

*The aim of this research is to use computational fluid dynamics and computational aeroacoustics to investigate aerodynamic noise mechanisms and assess the noise-reduction potential of porous coatings in idealised configurations representative of rotor-airframe installation effects in Urban Air Mobility Vehicles.*

The objectives of this research are to:

- Validate a computational methodology for predicting low Mach-number flow and acoustics using pressure-based methods.
- Critically assess the influence of turbulence modelling and acoustic analogies for airfoil self-noise predictions.
- Investigate the aeroacoustic behaviour of structured porous coatings using both fully resolved and reduced fidelity modelling approaches.
- Determine the effect of porous coatings on the flow and noise generated by modelled tip-vortex impingement on airframe mounting pylon- and wing-representative geometries.

# 3 NUMERICAL METHODOLOGY

## 3.1 Software Used

The following is a summary of the software used for various parts of the research:

- **Meshing:** Pointwise (versions 18.4 and 18.6), Fidelity Pointwise (2022, 2023)
- **Simulations:** OpenFOAM (versions v1906, v2112 and v2212)
- **Acoustics:** OpenFOAM libAcoustics
- **Unsteady data collection:** upSuite I/O library
- **Post-processing:** ParaView (versions 5.10.1 and 5.13.0)
- **Data analysis and figures:** MATLAB R2023b

## 3.2 Solvers

The rotors of UAMVs are expected to operate with a tip Mach number of  $M \approx 0.3$ , and a forward flight speed of approximately 150 mph (67 m/s), corresponding to  $M \approx 0.2$ . UAMVs therefore operate within the incompressible-to-weakly-compressible regime. Pressure-based solvers such as rhoSimpleFoam and rhoPimpleFoam have traditionally been used for incompressible and weakly compressible flows. Although the cases considered in this work have been set up to match experiments, up to  $M \approx 0.15$  for steady-state cases and  $M \approx 0.09$  for transient cases, the framework developed here is expected to be applicable to full-scale UAMVs, with  $Re$  increasing by a factor of 10. This is particularly pertinent to the later consideration of hybrid RANS-LES modelling. The following solvers were used in OpenFOAM for steady-state and transient simulations respectively:

### 3.2.1 rhoSimpleFoam

All steady-state simulations were conducted using rhoSimpleFoam. rhoSimpleFoam is OpenFOAM's compressible, steady-state solver with turbulence and heat transfer capabilities. It makes use of the Semi-Implicit Method for Pressure Linked Equations (SIMPLE) algorithm to solve the coupled pressure-momentum system.

### 3.2.2 rhoPimpleFoam

All transient simulations were conducted using rhoPimpleFoam. rhoPimpleFoam is OpenFOAM's compressible, transient solver with turbulence and heat transfer capabilities. rhoPimpleFoam makes use of a combination of the Pressure-Implicit with Splitting of Operators (PISO) and SIMPLE algorithms to solve the coupled pressure-momentum system.

### 3.2.3 Governing Equations

The governing equations for compressible flow are given below. The conservation of mass is given by:

$$\frac{\partial \rho}{\partial t} + \nabla \cdot (\rho \mathbf{U}) = 0 \quad (3.1)$$

The conservation of momentum is given by:

$$\frac{\partial (\rho \mathbf{U})}{\partial t} + \nabla \cdot (\rho \mathbf{U} \mathbf{U}) = -\nabla p + \nabla \cdot \boldsymbol{\tau} + \mathbf{F} \quad (3.2)$$

The conservation of energy is given by:

$$\frac{\partial (\rho E)}{\partial t} + \nabla \cdot (\rho \mathbf{U} E) = -\nabla \cdot \mathbf{q} + \frac{\partial p}{\partial t} + Q \quad (3.3)$$

Where

$$E = h + \frac{1}{2} \mathbf{U} \cdot \mathbf{U} = e + \frac{1}{2} \mathbf{U} \cdot \mathbf{U} + \frac{p}{\rho} \quad (3.4)$$

For rhoSimpleFoam and rhoPimpleFoam, density is calculated using the equation of state if a 'perfectGas' is specified:

$$p = \rho RT \quad (3.5)$$

### 3.2.4 Temporal and Spatial Discretisation

The backward temporal discretisation scheme is an implicit method that uses the current time level and the two preceding time levels to achieve second-order temporal accuracy.

The Linear Upwind Stabilized Transport (LUST) scheme is a second-order spatial discretisation method that applies a fixed weighting of 75% central differencing and 25% linear ( $2^{nd}$ -order) upwind.

## 3.3 Turbulence Modelling

Figure 3.1 shows a hierarchy of turbulence modelling approaches, with increasing computational cost and turbulence resolution from bottom to top. Due to the large computational cost, Direct Numerical Simulation (DNS) is generally limited to low Reynolds number flows. On the other hand, the predictive accuracy of Reynolds Averaged Navier–Stokes (RANS) is limited for complex flows including flow separation, reattachment and transition. Large Eddy Simulation (LES) resolves most scales, relying on turbulence modelling for approximately 10% of total turbulent kinetic energy. However, resolving the near-wall streaks for attached boundary layers using LES comes at a high cost for increasing Reynolds numbers, as shown by Tucker and Tyacke [130]. To alleviate this, RANS can be used to model the near-wall streaks with minor loss of predictive accuracy. Whilst this research considered rig-scale models, the full-scale Reynolds number would be much higher, hence hybrid RANS-LES approaches were also explored.

### 3.3.1 $y^+$ Estimation

Wall resolved solutions require  $y^+ < 1$ . An estimation of the first cell height was obtained as follows:

From the Reynolds number obtained in Equation 5.3, the skin friction velocity,  $C_f$ , is estimated according to Equation 3.6 [131]:

$$C_f = [2 \log_{10}(Re_x) - 0.65]^{-2.3} \quad (3.6)$$



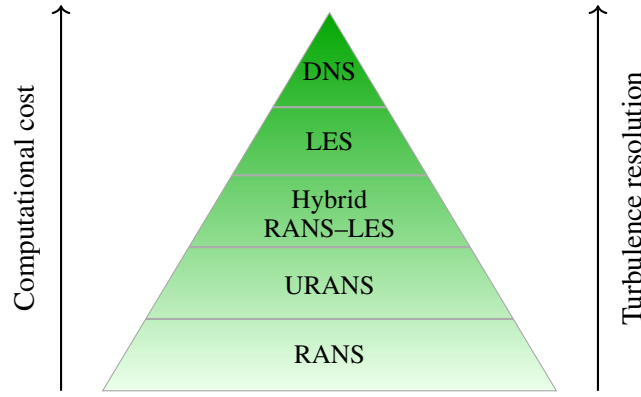


Figure 3.1: Hierarchy of turbulence models.

The wall shear stress,  $\tau_w$ , is then given by Equation 3.7:

$$\tau_w = \frac{1}{2} \rho U^2 C_f \quad (3.7)$$

The friction velocity,  $u^*$ , is then given by Equation 3.8:

$$u^* = \sqrt{\frac{\tau_w}{\rho}} \quad (3.8)$$

Based on a desired  $y^+$  value, the distance to the centroid of the first cell,  $\bar{y}$ , is given by Equation 3.9:

$$\bar{y} = \frac{y^+ \mu}{u^* \rho} \quad (3.9)$$

The height of the first cell,  $y_1$ , is found according to Equation 3.10:

$$y_1 = 2\bar{y} \quad (3.10)$$

The actual local  $y^+$  values are then computed in OpenFOAM according to Equation 3.11:

$$y^+ = \frac{u^* y}{\nu} \quad (3.11)$$

### 3.3.2 The $k - \omega$ SST Turbulence Model

The  $k - \omega$  turbulence model was first developed by Kolmogorov [132], where ‘ $k$ ’ represents the turbulent kinetic energy and ‘ $\omega$ ’ represents the turbulence frequency or specific dissipation rate. In 1988, improvements to the original model were made by Wilcox [133], since the conventional  $k - \omega$  formulations were found to be inaccurate for boundary layers in adverse pressure gradients and the use of wall functions were found to mask the shortcomings of the models. One of the major shortcomings of the model proposed by Wilcox [133] was its extreme sensitivity to the freestream  $\omega$  conditions specified outside the boundary

layer. Since the  $k - \epsilon$  model's near wall damping functions were found to be unreliable for diverse flows, but showed good predictions in the outer region and free shear flows, Menter [134] developed the  $k - \omega$  Baseline (BSL) model that uses the Wilcox [133]  $k - \omega$  model in the inner region of the boundary layer and switches to the  $k - \epsilon$  model in the outer region of the boundary layer and the free shear layer, by the use of a blending function. The definition of the eddy viscosity in the BSL model was then modified to account for the effect of the transport of the principal turbulent shear stress, resulting in the  $k - \omega$  SST model [134]. The model was found to provide major improvements in the prediction of adverse pressure gradient flows.

Since the original formulation of the  $k - \omega$  SST model, several modifications have been made to the model. The model implemented in OpenFOAM (v2112) is based on the 2003 formulation by Menter et al. [135]. The underlying transport equations are given below:

The turbulent kinetic energy transport equation is given by:

$$\frac{\partial(\rho k)}{\partial t} + \frac{\partial(\rho U_i k)}{\partial x_i} = \tilde{P}_k - \beta^* \rho k \omega + \frac{\partial}{\partial x_i} \left[ (\mu + \sigma_k \mu_t) \frac{\partial k}{\partial x_i} \right] \quad (3.12)$$

The specific dissipation rate transport equation is given by:

$$\frac{\partial(\rho \omega)}{\partial t} + \frac{\partial(\rho U_i \omega)}{\partial x_i} = \alpha \rho S^2 - \beta \rho \omega^2 + \frac{\partial}{\partial x_i} \left[ (\mu + \sigma_\omega \mu_t) \frac{\partial \omega}{\partial x_i} \right] + 2(1 - F_1) \rho \sigma_{\omega 2} \frac{1}{\omega} \frac{\partial k}{\partial x_i} \frac{\partial \omega}{\partial x_i} \quad (3.13)$$

### 3.3.3 $k - \omega$ SST IDDES and a Zonal LES Implementation

The IDDES length scale,  $l_{IDDES}$ , for the original  $k - \omega$  SST IDDES formulation is shown in Equation 3.14. Here,  $\tilde{f}_d$  represents the empirical blending function,  $f_e$  represents an elevation function,  $l_{RANS}$  represents the RANS-based length scale and  $l_{LES}$  represents the LES-based length scale. The derivation of these terms can be found in [136].

$$l_{IDDES} = \tilde{f}_d \cdot (1 + f_e) \cdot l_{RANS} + (1 - \tilde{f}_d) \cdot l_{LES} \quad (3.14)$$

The zonal-LES model used the simplified version of the  $k - \omega$  SST IDDES formulation for the IDDES length scale, which removes the elevation term  $f_e$ , is shown in Equation 3.15.

$$l_{IDDES} = \tilde{f}_d \cdot l_{RANS} + (1 - \tilde{f}_d) \cdot l_{LES} \quad (3.15)$$

The empirical blending function,  $\tilde{f}_d$ , was also modified to reflect limiting values of  $y_{RANS}^+$  and  $y_{LES}^+$ . Below  $y_{RANS}^+$  the RANS-based length scale is used. Above  $y_{LES}^+$ , the LES-based length-scale is used. Between these 2 values, linear blending of these length scales is imposed, as shown in Equation 3.16.

$$\tilde{f}_d = \begin{cases} 1, & \text{if } y^+ \leq y_{\text{RANS}}^+, \\ \frac{y_{\text{LES}}^+ - y^+}{y_{\text{LES}}^+ - y_{\text{RANS}}^+}, & \text{if } y_{\text{RANS}}^+ < y^+ < y_{\text{LES}}^+, \\ 0, & \text{if } y^+ \geq y_{\text{LES}}^+. \end{cases} \quad (3.16)$$

The RANS-based length scale,  $l_{\text{RANS}}$ , was left unchanged from the original model, and is shown in Equation 3.17. The constant  $C_\mu$  has a value of 0.09.

$$l_{\text{RANS}} = \sqrt{\frac{k}{C_\mu \omega}} \quad (3.17)$$

Similarly, the LES-based length scale,  $l_{\text{LES}}$ , was also left unchanged from the original model and is given in Equation 3.18. The filter-width,  $\Delta$ , is given in Equation 3.19, where  $C_w$  is a constant with a value of 0.15,  $d_w$  is the distance to the nearest wall and  $h_{\text{max}}$  is the maximum edge length of the cell. The derivation of  $C_{\text{DES}}$  can be found in [136].

$$l_{\text{LES}} = C_{\text{DES}} \Delta \quad (3.18)$$

$$\Delta = \min\{C_w \max[d_w, h_{\text{max}}], h_{\text{max}}\} \quad (3.19)$$

### 3.3.4 The WALE LES Model

The Wall-Adapting Local Eddy-viscosity (WALE) model developed by Franck and Ducros [137] is a subgrid-scale model used in LES. The subgrid eddy viscosity is given by

$$\nu_t = (C_w \Delta)^2 \frac{(S_{ij}^d S_{ij}^d)^{3/2}}{(\bar{S}_{ij} \bar{S}_{ij})^{5/2} + (S_{ij}^d S_{ij}^d)^{5/4}} \quad (3.20)$$

where  $\nu_t$  is the subgrid eddy viscosity,  $C_w$  is the WALE model constant, and  $\Delta$  is the local filter width. The resolved strain-rate tensor,  $\bar{S}_{ij}$ , is defined as

$$\bar{S}_{ij} = \frac{1}{2} \left( \frac{\partial \bar{u}_i}{\partial x_j} + \frac{\partial \bar{u}_j}{\partial x_i} \right) \quad (3.21)$$

where  $\bar{u}_i$  is the filtered velocity component in the  $x_i$  direction. The tensor  $S_{ij}^d$  is the traceless symmetric part of the square of the resolved velocity-gradient tensor, and is given by

$$S_{ij}^d = \frac{1}{2} (\bar{g}_{ij}^2 + \bar{g}_{ji}^2) - \frac{1}{3} \delta_{ij} \bar{g}_{kk}^2 \quad (3.22)$$

where

$$\bar{g}_{ij} = \frac{\partial \bar{u}_i}{\partial x_j} \quad (3.23)$$

and

$$\bar{g}_{ij}^2 = \bar{g}_{ik} \bar{g}_{kj}, \quad (3.24)$$

with  $\delta_{ij}$  denoting the Kronecker delta.

### 3.4 Unsteady Data Collection

The recording of unsteady data is essential to capture the temporal evolution of the flow, and supports analyses such as flow visualisation and time-resolved probe measurements. However, this can become both memory and file intensive. To the knowledge of the author, the most efficient way to sample unsteady data in OpenFOAM is to record the data on cut-planes in .vtk format. However, this results in one file per cut-plane per timestep (file structure shown in Figure 3.2a). The novel upSuite parallel I/O library developed by Upstream CFD GmbH is an HDF5-based library that allows users to choose how many timesteps are written to a single file via a 'flush cycle' option. The file structure of the upSuite I/O library is shown in Figure 3.2b.

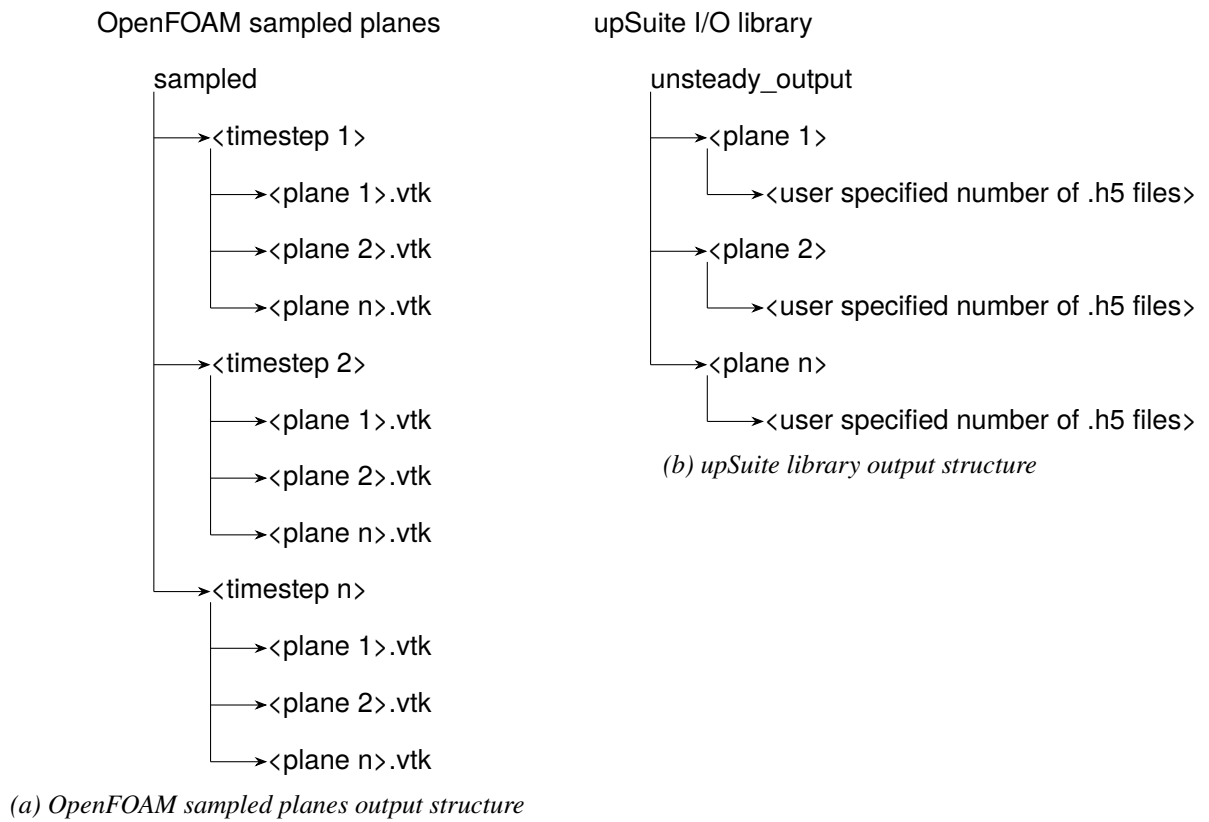


Figure 3.2: Comparison of OpenFOAM sampled plane output and upSuite HDF5-based output structure.

Figure 3.3 shows a comparison between the simulation runtimes for a baseline case with no sampling, OpenFOAM sampling and upSuite sampling. The computational domain consisted of 373,500 elements, and was decomposed into 16, 32 and 64 cores. A midspan plane consisting of 12,599 points was used for the sampling. The simulations were run for 1 s, at a timestep of  $1 \times 10^{-4}$  s, with data being recorded every timestep, resulting in 10,000 timesteps being saved. From Figure 3.3, it can be seen that both OpenFOAM and upSuite sampling add time to the baseline simulations. However, for OpenFOAM sampling, the increase in computational time is more drastic, with a 28.7%, 56.6% and 62.8% increase in computational time for 16, 32 and 64 cores respectively. For upSuite, there is a 3%, 3.1% and 8.5% increase in computational time at 16, 32 and 64 cores respectively. Therefore it can be seen that although the problem size stays the same with respect to the amount of recorded data, sampling from an increased number of cores results in a slowing down of the sampling process, associated with the increased cost of gathering and reconstructing distributed sampled data across a larger number of processors.

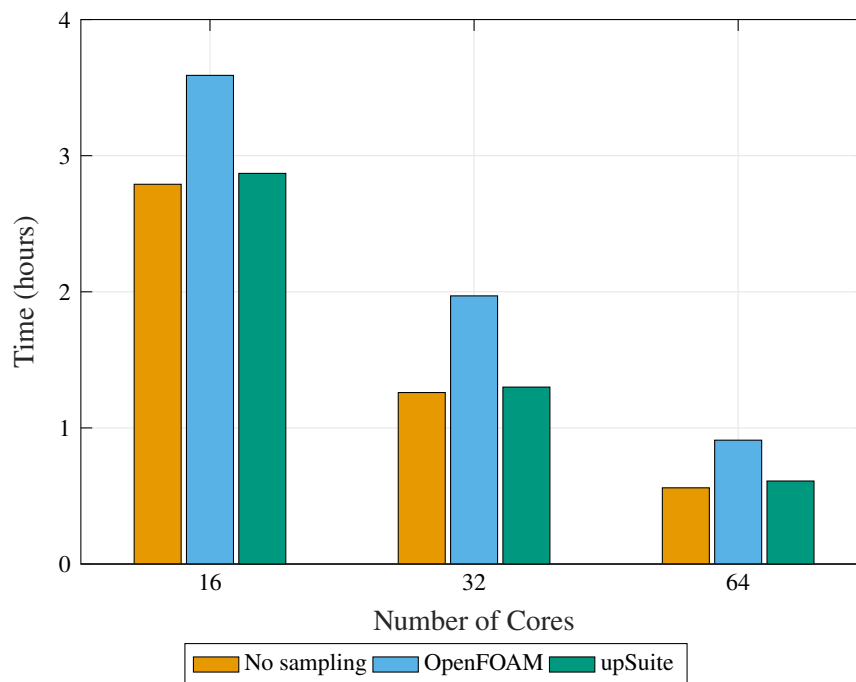


Figure 3.3: Run time comparison comparing OpenFOAM and upSuite sampling with a baseline case.

Table 3.1 shows a comparison between the OpenFOAM built-in sampling and the upSuite I/O library. For the tested case, upSuite was requested to write 1,000 timesteps per file, and therefore produced 1,000 times fewer files and used only 31.1% of the storage used by OpenFOAM sampling. It is, however, important to note that if a large number is chosen for the flush cycle, the timesteps are stored in memory until written. Therefore, depending on the number of points being recorded, a lower flush cycle may need to be chosen.

Table 3.1: Comparison between OpenFOAM and upSuite sampling

| Library  | Points sampled | Timesteps saved | Files outputted | Storage |
|----------|----------------|-----------------|-----------------|---------|
| OpenFOAM | 12,599         | 10,000          | 10,000          | 18 GB   |
| upSuite  | 12,599         | 10,000          | 10              | 5.6 GB  |

# 4

## OPENFOAM SCALING TESTS

Scaling tests are important to ensure the efficient use of computational resources. A brief background on scaling and some studies conducted on the scaling of various OpenFOAM solvers can be found in Appendix A. Scaling tests conducted on the tier 2 HPC, MMM Young, and the tier 1 HPC, Archer2, are discussed in this chapter.

### 4.1 Strong Scaling of OpenFOAM v2112 on MMM Young

A strong scaling test was done on OpenFOAM's rhoPimpleFoam solver on the Tier 2 supercomputer, MMM Young. The mesh consisted of 10,397,750 hexahedral elements, and the simulation was run for 1,000 timesteps in each case. The scaling tests were conducted from 80 to 960 cores (between 2 and 24 nodes), corresponding to an average of between 129,972 and 10,831 cells per core. The results are shown in Figure 4.1a.

The strong scaling case was then run again using a mesh containing twice the number of cells, that is, 20,795,500 cells (also shown in Figure 4.1a). A noticeable increase in run time can be seen at 14 nodes for both the small and large cases. This possibly implies that beyond 13 nodes, internode communication becomes a contributor to the difference observed, in addition to the number of cells per core.

From Figure 4.1a, it can be seen that the graph of CPU time versus number of cores shows asymptotic behaviour at around 13 nodes. This behaviour can be noticed for both the case with 19,996 cells per core and 39,991 cells per core. Elfverson and Lejon [138] found a similar trend, where excellent speedup was achieved up to 35 cores for cases containing both 65,500 and 310,000 cells per core, indicating that the bottleneck was more dependent on the number of cores used rather than the number of cells per core. However, from Figure 4.1a, it can be seen that speedup was achieved all the way to the maximum tested number of cores.

Figure 4.1b shows a representation of the speedup versus the number of cores, using the 80 cores case as a reference. For the smaller case, excellent speedup is shown by rhoPimpleFoam up to 13 nodes, with superlinear speedup being achieved at 12 and 13 nodes. The large case also shows excellent speedup up to 13 nodes, although not superlinear. The speedup plot also shows the bottleneck at 14 nodes for both cases.

Based on Amdahl's Law discussed in Section A.1, the parallel speedup,  $\sigma_n$ , can be given by Equation 4.1:

$$\sigma_n = \frac{T_{ref}}{T_n} \quad (4.1)$$

The parallel efficiency,  $\epsilon_n$ , is then given by Equation 4.2:

$$\epsilon_n = \frac{n_{ref}}{n} \sigma_n \quad (4.2)$$

For Equations 4.1 and 4.2:

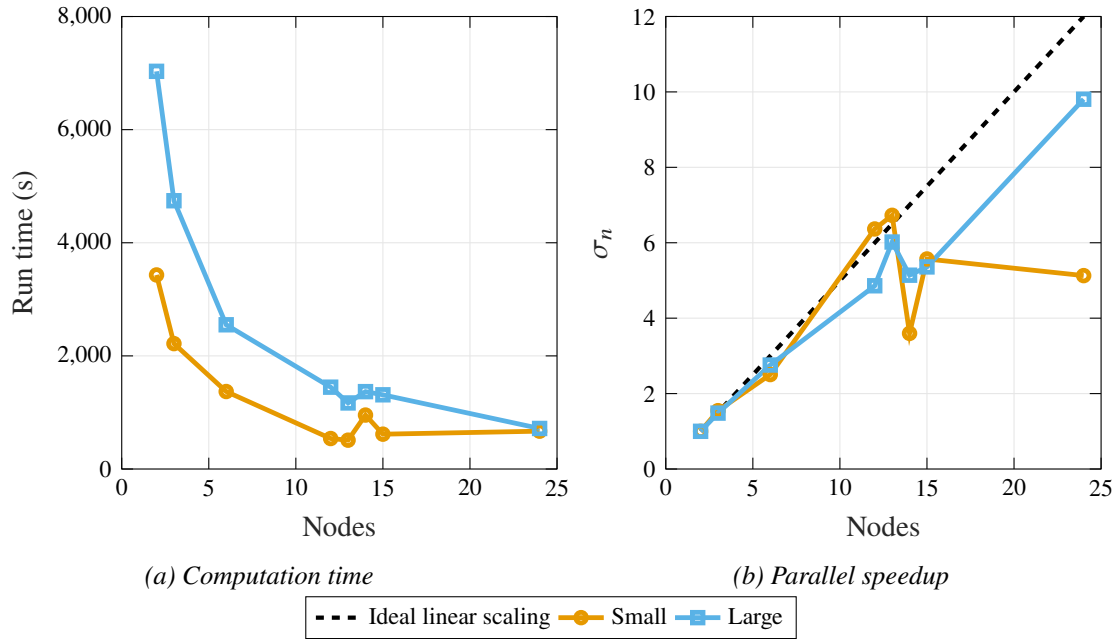


Figure 4.1: Run time and parallel speedup for the Small and Large case.

|            |                                                  |              |                                                     |
|------------|--------------------------------------------------|--------------|-----------------------------------------------------|
| $\sigma_n$ | Parallel speedup for case with $n$ processors    | $T_{ref}$    | Simulation time for the reference case              |
| $T_n$      | Simulation time for the case with $n$ processors | $\epsilon_n$ | Parallel efficiency of the case with $n$ processors |
| $n_{ref}$  | Number of cores in the reference case            | $n$          | Number of cores in the tested case                  |

Based on the above analysis, the optimum number of cells per core for the rhoPimpleFoam solver appears to be  $\approx 20,000$ . This number seems to fit well with that discussed by Axtmann and Rist [139], when testing the pisoFoam solver. Since the PIMPLE algorithm is based on a combination of the PISO and SIMPLE algorithms, it is fitting that the performance is similar for 20,000 cells per core.

## 4.2 Strong Scaling of OpenFOAM v2212 on Archer2

A newer version of OpenFOAM, specifically v2212, was installed on Archer2, and a new series of strong scaling tests were conducted. The original strong scaling tests conducted on OpenFOAM v2112 can be found in Appendix A.3.

Strong scaling tests were carried out on three computational domains, a small, medium and large domain, with the cell counts for each summarised in Table 4.1.

Table 4.1: Parameters of the domains for Archer2 scaling tests

| Mesh   | Cells      |
|--------|------------|
| Small  | 28,253,016 |
| Medium | 42,522,216 |
| Large  | 84,759,048 |

Archer2 contains 128-core nodes, and all further data is provided in terms of nodes. The cases were decomposed into between 1 and 47 nodes and the small case was run for 100 iterations to determine the strong scaling characteristics.

When running parallel cases in OpenFOAM, there are two options for file structure, namely, collated and uncollated. The uncollated file structure is the default in OpenFOAM, in which each processor has a dedicated directory and writes its own files to that directory. This means that the file count scales with the number of processors used. The collated format writes all the data into a single directory, reducing the number of files by a factor of the number of processors used compared to the uncollated format. This is beneficial in terms of keeping the file count manageable on HPC systems. However, OpenFOAM user documentation notes that writing data using the collated format may be slower than the uncollated format. Both formats were compared with each other.

Figure 4.2a shows the run time vs. number of nodes used for the small case, using both the collated and uncollated file structures. It can be seen that both formats behave relatively consistently. Figure 4.2b shows that when using one node as a reference, both collated and uncollated file formats exhibit superlinear speedup, with the uncollated structure generally performing better than the collated structure.

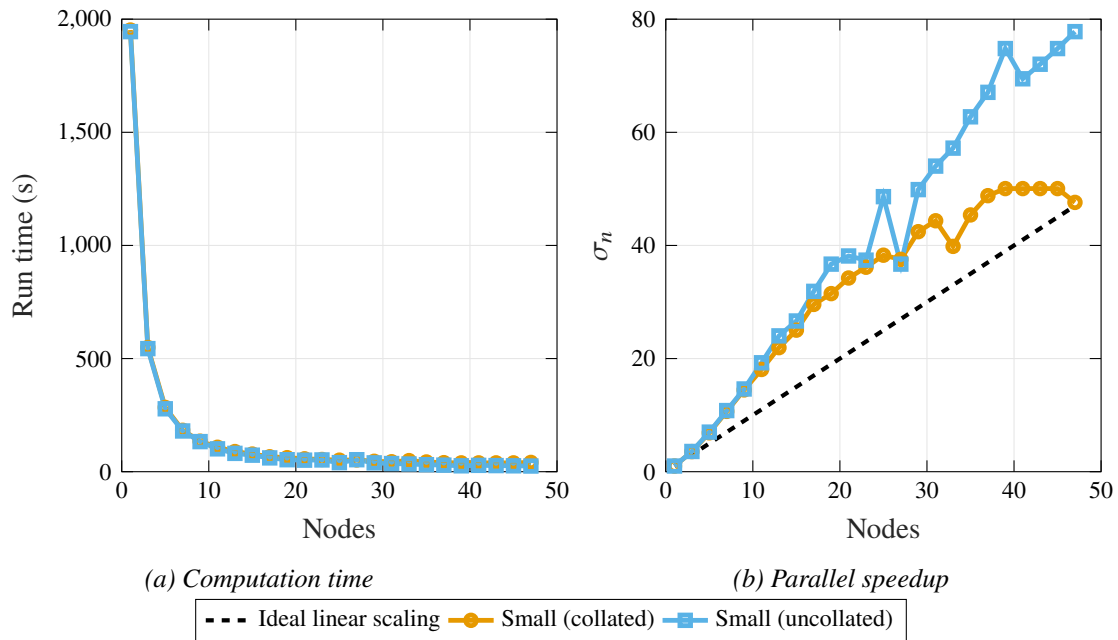


Figure 4.2: Run time and parallel speedup for the Small case.

Mesh partitioning, called decomposition in OpenFOAM is a serial process. On large meshes, this can become both memory intensive and time consuming. An investigation into the time taken for mesh partitioning for all three meshes, using both the collated and uncollated file structures was carried out.

The user time is predominantly the time spent executing the `decomposePar` utility, i.e., the calculation of the mesh partitioning, whereas the system time primarily corresponds to file system operations such as reading and writing mesh data. The total time is the sum of the user and system times.

Figure 4.3a shows the user, system and total decomposition times for collated and uncollated file structure for the small case. Whilst the user time is relatively consistent between the collated and uncollated structures, the collated format shows a significantly higher system time than the uncollated structure, thus also showing a higher total decomposition time. The increased system time can be attributed to increased operating system overhead associated with file I/O and data handling during the collated writing process.



Since the scaling tests were only carried out for 100 timesteps, the run time was extrapolated to 100,000 timesteps to provide an insight into what the time costs of a full simulation might be. Figure 4.3b shows that the simulations run with the uncollated file structure show a significant decrease in run time, which becomes more pronounced when using more computational nodes. Therefore, the uncollated file format shows an overall better performance in both decomposition time and simulation run time, although this comes with the penalty of writing large amounts of files to disk. For the case run at 47 nodes, using the uncollated file format, it can be seen that the decomposition time accounts for approximately 8.5% of the total time, and it is therefore important to accommodate for this when planning computational time.

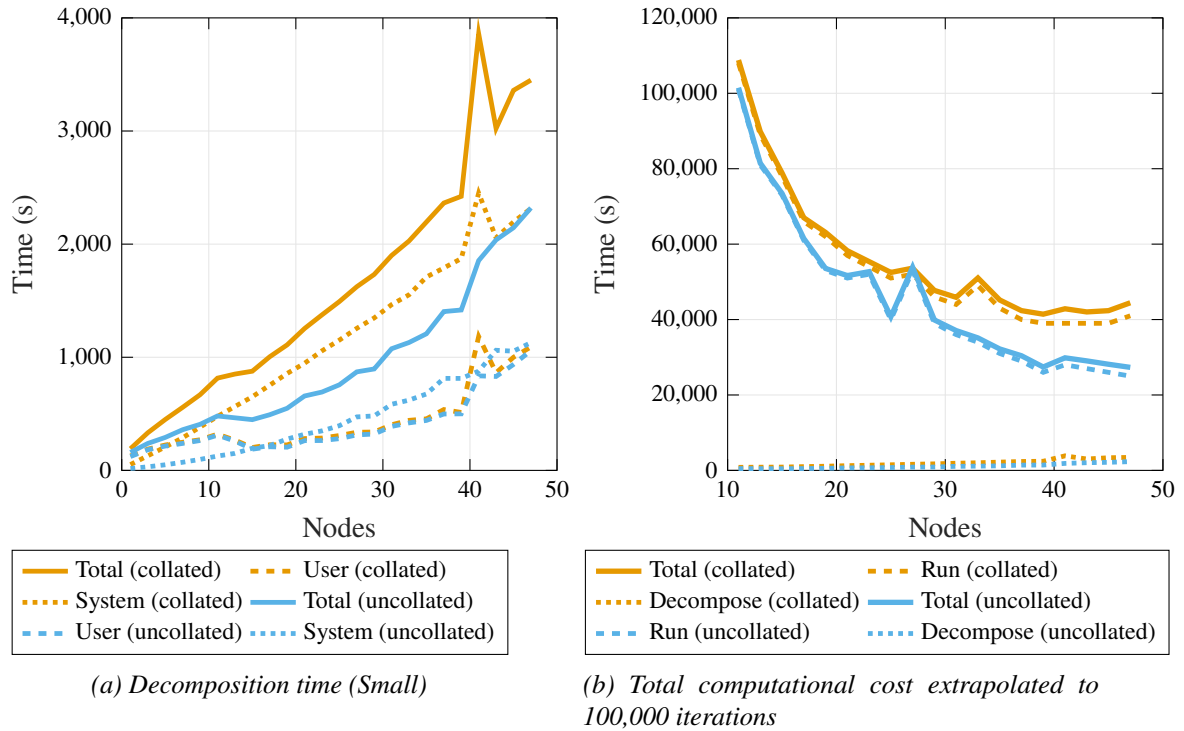


Figure 4.3: Decomposition time and total computational cost assuming 100,000 iterations

Similar to the decomposition of the small mesh (Figure 4.3a), Figure 4.4 shows that for both the medium (Figure 4.4a) and the large (Figure 4.4b) cases, the collated structure also exhibits a higher system (and thus total) decomposition time compared to the uncollated structure. By comparing the total decomposition time for the uncollated medium and large cases, it can be seen that the large case on average takes 1.85 times longer to decompose than the medium case, for a case that is twice as large. Therefore, for the tested case sizes and number of nodes, it can be concluded that decomposition time scales roughly linearly with mesh size, but this may be case specific.

### 4.3 Weak Scaling of OpenFOAM v2112 on MMM Young

The weak scaling test was conducted using the steady-state, compressible solver, rhoSimpleFoam, using the same initial conditions and turbulence model as the case used for the strong scaling test discussed in Section 4.1. Tests were carried out for between 40 and 640 cores (1 and 16 nodes). The initial mesh (corresponding to 1 node), consisted of 1,039,775 cells. The mesh was then sequentially doubled in the spanwise direction to conduct the test. The average cells per core for all cases was approximately 26,000.

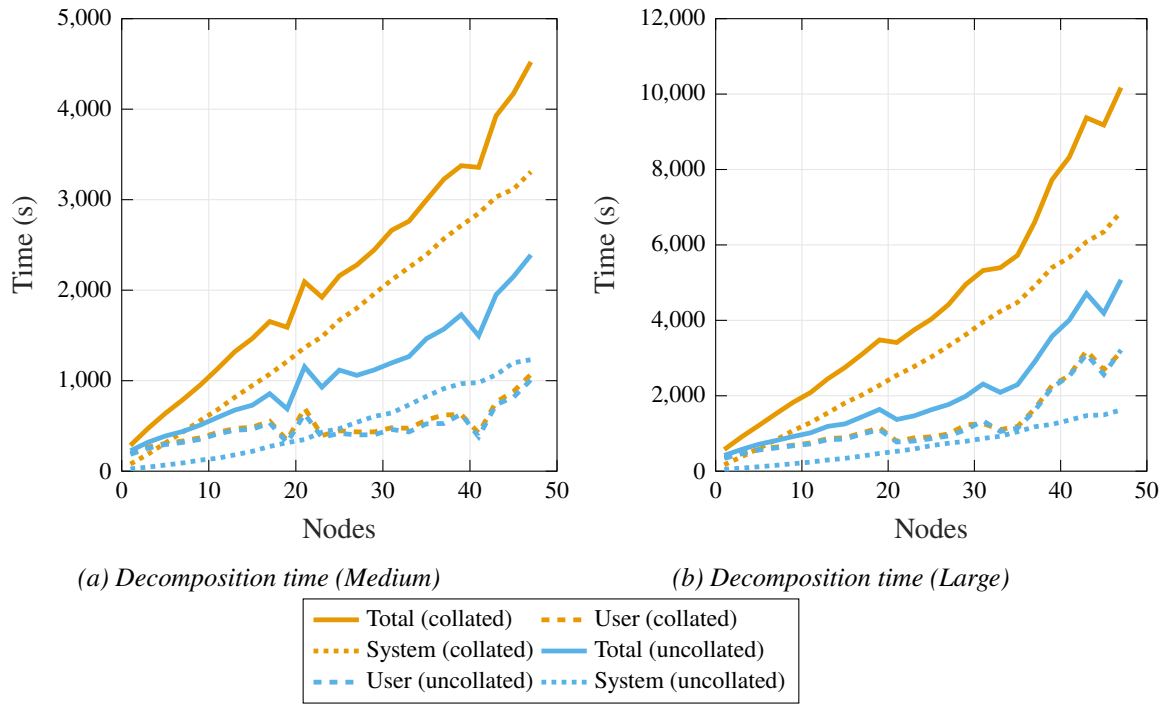


Figure 4.4: Decomposition times for the Medium and Large cases

Similar to the strong scaling test, each simulation was allowed to run for 1,000 iterations.

Another weak scaling test was then carried out using the same conditions, except with double the amount of cells per core, approximately 52,000. The results are shown in Figure 4.5a.

When dealing with weak scaling, the efficiency of the scaling is compared rather than a speedup factor such as with strong scaling. By using the 40 cores case as a reference, the efficiency can then be found according to Equation 4.3.

From Figure 4.5a, it can be seen that the weak scaling trend is similar for both the 26,000 and 52,000 cells per core cases. Both cases show an increase in run time for increased problem size.

Figure 4.5b shows the scaling efficiency as a function of the number of nodes used. From the figure, it can be seen that the scaling efficiency for the larger case is less than that of the smaller case up to 8 nodes, at which point the efficiency of the larger case becomes greater.

$$Efficiency = \frac{T_{40}}{T_N} \quad (4.3)$$

For Equation 4.3:

$$T_{40} \mid \text{Time for simulation to run on 40 cores} \quad T_N \mid \text{Time for simulation to run on N cores}$$

OpenFOAM's rhoSimpleFoam tested on MMM Young exhibits unsatisfactory weak scaling, especially when compared to Orgogozo et al. [140] who achieved 80% parallel efficiency up to 1,024 cores for the RichardsFoam solver. For the small case at 16 nodes, a 25% efficiency was achieved whilst for the larger

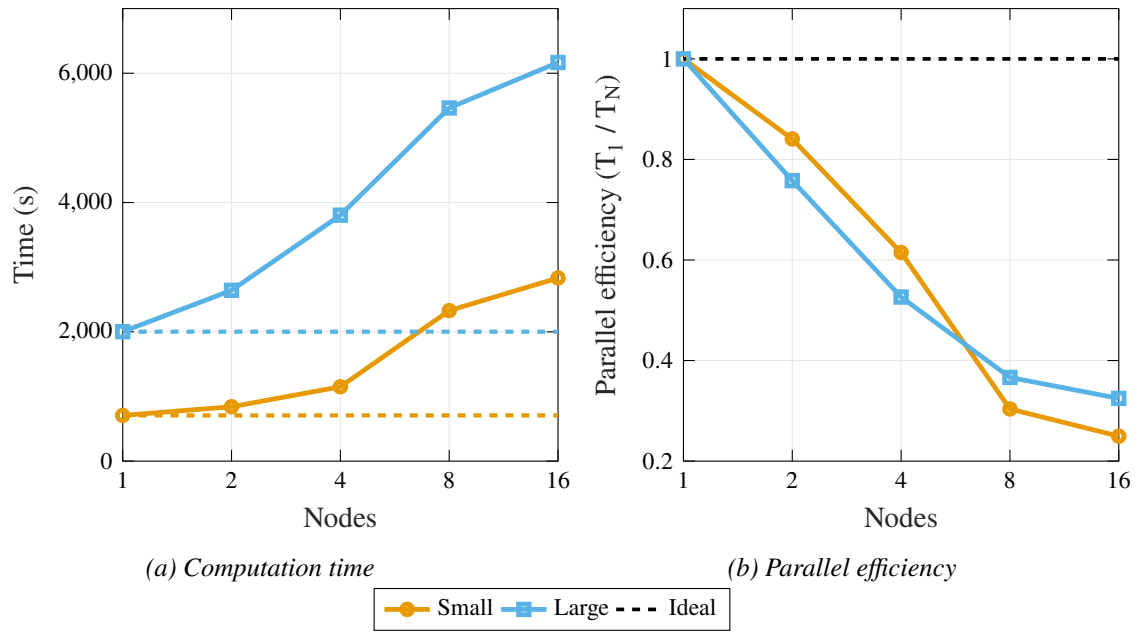


Figure 4.5: Run time and parallel efficiency for the Small and Large case.

case, a 32.5% efficiency was achieved.

# 5

## OPENFOAM SOLVER 2-D RANS VALIDATION

The Turbulence Modelling Resource by the National Aeronautics and Space Administration (NASA) provides a validation case for various turbulence models [141]. In this case, a 2-D NACA 0012 airfoil validation case was used to test the performance of OpenFOAM's steady-state compressible solver, namely, rhoSimpleFoam. The quantities of interest to be compared to are:

- Lift coefficient ( $C_L$ ) vs. angle of attack (AoA) (for  $0^\circ$ ,  $10^\circ$ ,  $15^\circ$ )
- Drag coefficient ( $C_D$ ) vs. AoA (for  $0^\circ$ ,  $10^\circ$ ,  $15^\circ$ )
- Surface pressure coefficient ( $C_p$ ) vs.  $x/c$  (for  $0^\circ$ ,  $10^\circ$ ,  $15^\circ$ )
- Surface skin friction coefficient ( $C_f$ ) vs.  $x/c$  (for  $0^\circ$ ,  $10^\circ$ ,  $15^\circ$ )

The turbulence model chosen for the 2-D RANS simulations was the  $k - \omega$  SST model. The  $k - \omega$  SST model is well studied, and has been found to provide a more accurate prediction of flow separation when compared to most other RANS models. Douvi et al. [142] conducted CFD analysis at  $Re = 3 \times 10^6$  on a NACA 0012 airfoil using three turbulence models, namely, the Realizable  $k - \epsilon$ ,  $k - \omega$  SST and Spalart-Allmaras models. The  $k - \omega$  SST model performed the best when compared with reliable lift and drag coefficient experimental data by Abbott and Von Doenhoff [143]. Similarly, Muiruri et al. [144] used the  $k - \epsilon$  Renormalization Group Theory (RNG), Realizable  $k - \epsilon$ ,  $k - \omega$  SST and Spalart-Allmaras models to determine aerodynamic torque of the NREL 5MW wind turbine rotor. The  $k - \omega$  SST model obtained aerodynamic torque results in closest agreement to the other reliable numerical simulation results. Qu et al. [145] found the  $k - \omega$  SST model to be more accurate than the Realizable  $k - \epsilon$  model when predicting the surface elevations and breaking wave forces for breaking waves past a vertical cylinder. However, the  $k - \omega$  SST model was found to overpredict turbulent kinetic energy. Nevertheless, the relative accuracy of the  $k - \omega$  SST model is evident.

### 5.1 Initial Conditions

The boundary conditions used by NASA for this validation are as follows:

- $Re = 6,000,000$
- Mach no. = 0.15
- Temperature = 300 K

The properties of dry air at  $T = 300$  K is as follows:

- Dynamic viscosity  $\mu = 1.846 \times 10^{-5} \text{ kg/m} \cdot \text{s}$
- Density  $\rho = 1.177 \text{ kg/m}^3$
- Kinematic viscosity  $\nu = 1.568 \times 10^{-5} \text{ m}^2/\text{s}$

The speed of sound for air at  $T = 300$  K is given by Equation 5.1:

$$a = \sqrt{\gamma RT} \quad (5.1)$$

The speed of sound at 300 K was found to be 347.2 m/s. The required freestream velocity for a Mach number of  $M = 0.15$  was found to be 52.08 m/s and is given by Equation 5.2:

$$U_{\infty} = M \times a \quad (5.2)$$

Given that the Reynolds number is given by Equation 5.3:

$$Re = \frac{U_{\infty} \times L}{\nu_{\infty}} \quad (5.3)$$

The required chord length for the case was therefore found to be 1.80645 m. The freestream turbulence intensity was required to be 0.052% and the freestream turbulent viscosity  $0.009\nu_{\infty}$  [141].

Sutherland's law was used to determine the local dynamic viscosity. Sutherland's law is given by Equation 5.4:

$$\mu = \mu_0 \left( \frac{T}{T_0} \right)^{3/2} \left( \frac{T_0 + S}{T + S} \right) \quad (5.4)$$

For Equation 5.4:

|       |                                            |         |                                                |
|-------|--------------------------------------------|---------|------------------------------------------------|
| $\mu$ | Local dynamic viscosity ( $kg/m \cdot s$ ) | $\mu_0$ | Reference dynamic viscosity ( $kg/m \cdot s$ ) |
| $T$   | Local temperature ( $K$ )                  | $T_0$   | Reference temperature (273.15 $K$ )            |
| $S$   | Sutherland temperature (110.4 $K$ )        |         |                                                |

The freestream pressure was found to be 101,360.18 Pa and was determined according to Equation 5.5:

$$p = \rho RT \quad (5.5)$$

## 5.2 The Computational Domain

NASA used the  $897 \times 257$  grid with 513 points on the airfoil surface for all validation cases, subsequently, the same domain was used for this validation. The layout of the computational domain is shown in Figure 5.1. Note that the domain shown in Figure 5.1 was scaled up by a factor of approximately 1.8.

The resulting structured computational domain resulted in 229,376 hexahedral cells, with a first cell height normal to the wall being  $\sim 1.49 \times 10^{-6} m$  and the average growth ratio near the wall being  $\sim 1.08$ . This produced  $\max y^+ < 1$  for the three cases, enabling the boundary layer to be resolved without the use of wall functions. As the  $k - \omega$  SST model is used, the  $k - \omega$  turbulence model is retained near walls whilst  $k - \epsilon$  is used elsewhere. Figure 5.2 shows the computational grid around the airfoil.

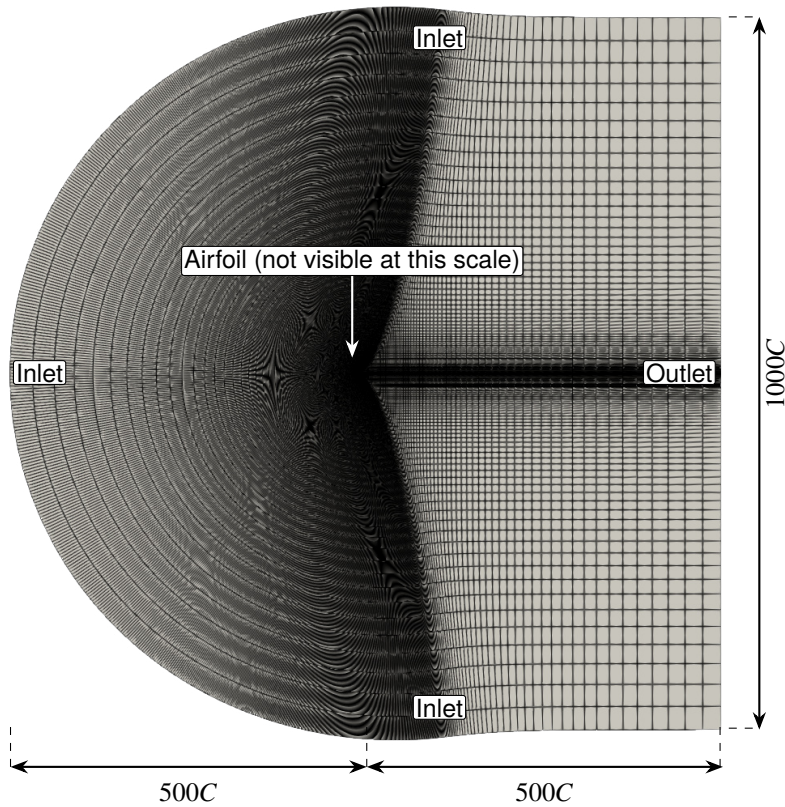


Figure 5.1: Layout of the NASA grid  
[141]

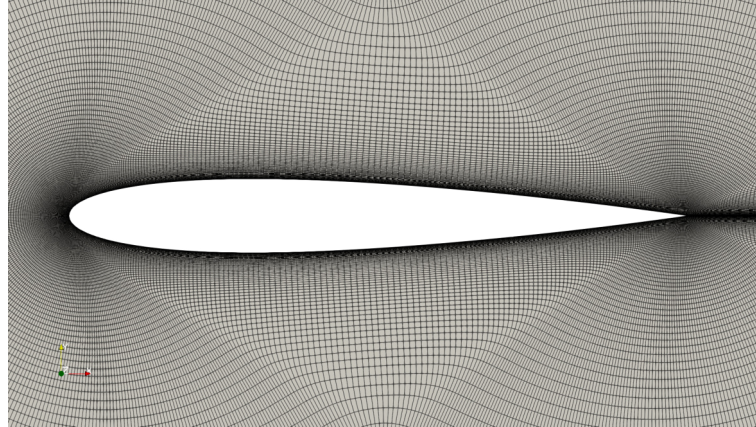


Figure 5.2: Grid structure around the airfoil  
[141]

### 5.3 Spatial Discretisation Schemes

The “Gauss linear” gradient scheme was used with the “Multi-directional cell-limited gradient scheme (cellMDLimited)” gradient limiter. The second order upwind scheme, “bounded Gauss linearUpwind” was used to discretise the divergence of the momentum and energy equations while turbulence was discretised using the first order upwind “bounded Gauss upwind” scheme. According to Jones et al. [146], the discretisation scheme used for the convective divergence term in the turbulence equation has negligible effect on the accuracy of the solution. Therefore, the first order scheme was used to ensure a stable solution

without the loss of accuracy.

## 5.4 2-D Validation Results

The following section compares the quantities of interest mentioned in Chapter 5.

### 5.4.1 Lift and Drag Coefficients

Table 5.1 shows a comparison between the lift and drag coefficients achieved by NASA's CFD codes (CFL3D, FUN3D), NTS and OpenFOAM's rhoSimpleFoam solver. Ladson et al. [147] conducted experiments to determine the aerodynamic characteristics of a NACA 0012 airfoil section for  $0.05 \leq M \leq 0.36$  and Reynolds numbers varying from  $2 \times 10^6 \leq Re \leq 12 \times 10^6$ , for angles of attack ranging from the corresponding minimum to maximum lift coefficient. Therefore, for  $M = 0.15$  and  $Re = 6 \times 10^6$ , these experimental results provide an appropriate comparison, and are also included in Table 5.1.

Table 5.1: Comparison of lift and drag coefficient data at different angles of attack

| Code              | $C_L$ (0°) | $C_L$ (10°) | $C_L$ (15°) | $C_D$ (0°) | $C_D$ (10°) | $C_D$ (15°) |
|-------------------|------------|-------------|-------------|------------|-------------|-------------|
| Ladson experiment | -0.00724   | 1.0586      | 1.4938      | 0.00809    | 0.01191     | 0.01831     |
| CFL3D             | ~0         | 1.0778      | 1.5068      | 0.00809    | 0.01236     | 0.02219     |
| FUN3D             | ~0         | 1.0840      | 1.5109      | 0.00808    | 0.01253     | 0.02275     |
| NTS               | ~0         | 1.0765      | 1.5100      | 0.00809    | 0.01251     | 0.02187     |
| rhoSimpleFoam     | ~0         | 1.0802      | 1.4941      | 0.00809    | 0.01233     | 0.02264     |

Table 5.2 shows the absolute percentage deviation in the lift and drag coefficients obtained by rhoSimpleFoam, from the other CFD codes, as well as the Ladson et al. [147] experimental data, where the absolute percentage deviation is given by:

$$\text{Absolute percentage deviation} = 100 \left| \frac{C_{\text{rhoSimpleFoam}} - C_{\text{reference}}}{C_{\text{reference}}} \right| [\%]. \quad (5.6)$$

From Table 5.2, rhoSimpleFoam performed relatively consistently with all the compared CFD codes, with the maximum deviation being 3.53% for the drag coefficient and 2.04% for the lift coefficient. The results also correspond well with the experimental data for  $AoA = 0^\circ$  and  $10^\circ$ . However, at  $AoA = 15^\circ$ , the drag coefficient obtained by rhoSimpleFoam is 23.64% greater than the experimental data. This is due to the fact that 2-D experiments are difficult to achieve, especially as the AoA increases [141]. Therefore, the performance of rhoSimpleFoam can be considered valid when compared solely with the other CFD codes.

Table 5.2: Absolute percentage deviation of rhoSimpleFoam's lift and drag coefficients from Ladson experimental data and other CFD codes

| Coefficient and AoA | Ladson experiment | CFL3D | FUN3D | NTS   |
|---------------------|-------------------|-------|-------|-------|
| $C_L$ (0°)          | —                 | —     | —     | —     |
| $C_L$ (10°)         | 2.04%             | 0.22% | 0.35% | 0.35% |
| $C_L$ (15°)         | 0.02%             | 0.84% | 1.11% | 1.05% |
| $C_D$ (0°)          | 0.06%             | 0.04% | 0.08% | 0.04% |
| $C_D$ (10°)         | 3.50%             | 0.27% | 1.62% | 1.47% |
| $C_D$ (15°)         | 23.64%            | 2.04% | 0.48% | 3.53% |

### 5.4.2 Surface Pressure and Skin Friction Coefficients

Figure 5.3 shows a comparison of the computational and experimental surface pressure and skin friction coefficients for the NACA 0012 airfoil for  $AoA = 0^\circ, 10^\circ, 15^\circ$ . The leading edge upper surface pressure peak is well resolved by the experiments conducted by Gregory and O'Reilly [148], and is therefore used for comparison [141]. From Figures 5.3a, 5.3c and 5.3e, it can be seen that the results obtained by rhoSimpleFoam correlate well with both the CFL3D computational results and the Gregory and O'Reilly [148] experimental results.

Similar to the surface pressure coefficients, rhoSimpleFoam shows good agreement with the results obtained by CFL3D for the surface skin friction coefficients, as shown in Figures 5.3b, 5.3d and 5.3f. Note that there are currently no known experimental skin friction data available for this case, hence a comparison is only made with the computational CFL3D code.

## 5.5 Summary

In this chapter, the performance of OpenFOAM's compressible solver, rhoSimpleFoam, was validated using the NASA Turbulence Modelling Resource 2-D NACA 0012 airfoil case. The validation was carried out using the  $k - \omega$  SST turbulence model and the structured NASA grid, with the aim of assessing the ability of the selected OpenFOAM setup to predict aerodynamic loading and near-wall flow behaviour under compressible flow conditions.

The lift coefficient obtained using rhoSimpleFoam showed good agreement with the NASA CFL3D and FUN3D codes, the NTS code and the experimental data of Ladson et al., with a maximum deviation of 2.04% from the computational reference data. The drag coefficient also agreed well with the computational results, with a maximum deviation of 3.53%. Agreement with the experimental data was satisfactory at  $0^\circ$  and  $10^\circ$ , although rhoSimpleFoam predicted a noticeably higher drag coefficient at  $15^\circ$ . This discrepancy is likely associated with the difficulty in achieving purely 2-D experimental conditions at higher angles of attack, as also noted by NASA.

The surface pressure coefficient distributions showed strong correlation with both the CFL3D results and the experimental data of Gregory and O'Reilly across all three angles of attack. This included good agreement in the leading edge suction peak, which is particularly important when assessing the accuracy of the predicted pressure field. The surface skin friction coefficient also showed good agreement with the CFL3D data, although comparison was limited to computational results due to the absence of experimental skin friction data.

The results demonstrate that rhoSimpleFoam, when used with the  $k - \omega$  SST turbulence model and the selected numerical setup, is capable of accurately predicting the aerodynamic performance of the NACA 0012 airfoil for this validation case.



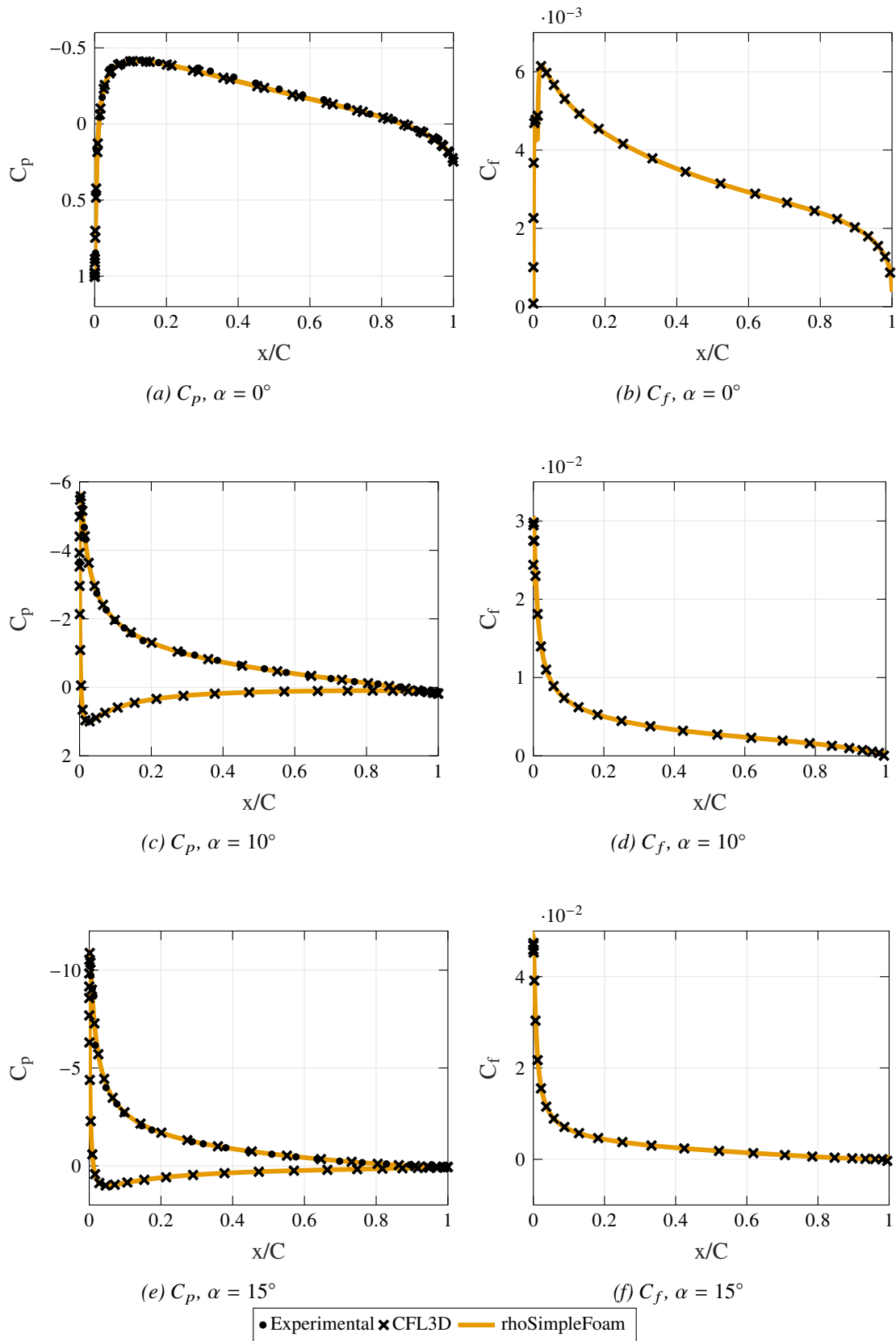


Figure 5.3: Pressure coefficient ( $C_p$ ) and skin friction coefficient ( $C_f$ ) at various angles of attack.

# 6

## OPENFOAM SOLVER 3-D LES VALIDATION

The LES validation was conducted using OpenFOAM's transient, compressible solver, rhoPimpleFoam. The case was initialised using a converged RANS simulation from OpenFOAM's steady, compressible solver, rhoSimpleFoam. For the initialisation, the  $k - \omega$  SST model was used. The mean and Root Mean Square (RMS) velocity profiles at various wake locations and on the airfoil surface are compared to experimental data obtained by Sagrado [149]. LES computational data obtained by Lin et al. [150] is also compared to the experimental data for comparison. Sagrado [149] conducted experiments on a NACA 0012 airfoil to determine the vortex shedding noise. The airfoil used was a NACA 0012 airfoil with a chord of 0.297 m, which was created by reducing the chord of a 0.3 m chord airfoil by 3 mm. The shortened airfoil resulted in a 1.6 mm thick trailing edge. An aspect ratio of 1 was used for the experiments. Lin et al. [150] aimed to validate the same experimental results using a Block Overset Fast Flow Solver (BOFFS) CFD code, with  $span = 0.3 \times chord$ . The validation is discussed in the following sections.

### 6.1 Boundary Conditions

At the inlet, the velocity was set to  $U_\infty = 10$  m/s, corresponding to a chord-based Reynolds number of  $Re = 200,000$  and a Mach number of  $M = 0.029$ , while a zero-gradient condition was set for pressure. At the outlet, a zero-gradient condition was applied for velocity, while the pressure was set to  $p = 101,306.12$  Pa. A no-slip condition was applied at the walls, and spanwise periodicity was used.

### 6.2 Temporal and Spatial Discretisation Schemes

For all cases, temporal discretisation was performed using the second-order backward difference scheme. The simulations were run for  $40T^*$  with temporal statistics conducted over  $20T^*$ , where  $T^* = C/U_\infty$ . The transient timestep for all cases was  $1.01 \times 10^{-4}T^*$ . For spatial discretisation, three schemes were tested, the first being the LUST scheme. The second scheme was the Linear scheme which is a second-order central differencing scheme and the third scheme was the Cubic scheme which is a fourth-order central scheme.

### 6.3 The Computational Domains

The computational domain was chosen to be a structured C-grid extending 4 chord lengths upstream, 20 chord lengths downstream and 5 chord lengths above and below the airfoil. Maximum growth ratios of 5% and 2% were used in the wall-normal and wake directions respectively. Table 6.1 summarises the mesh parameters for the different spans.

Table 6.1: Parameters of the different domains

| Mesh   | Span ( $z/C$ ) | Cells      | Cells in $z$ | $\max(\Delta x^+, \Delta y^+, \Delta z^+)$ | Cores  | Cells/core |
|--------|----------------|------------|--------------|--------------------------------------------|--------|------------|
| Small  | 0.2            | 28,253,016 | 98           | 50, 1, 20                                  | 5,632  | 5,016.5    |
| Medium | 0.3            | 42,522,216 | 149          | 50, 1, 20                                  | 8,448  | 5,033.4    |
| Large  | 0.6            | 84,759,048 | 297          | 50, 1, 20                                  | 17,024 | 4,978.8    |

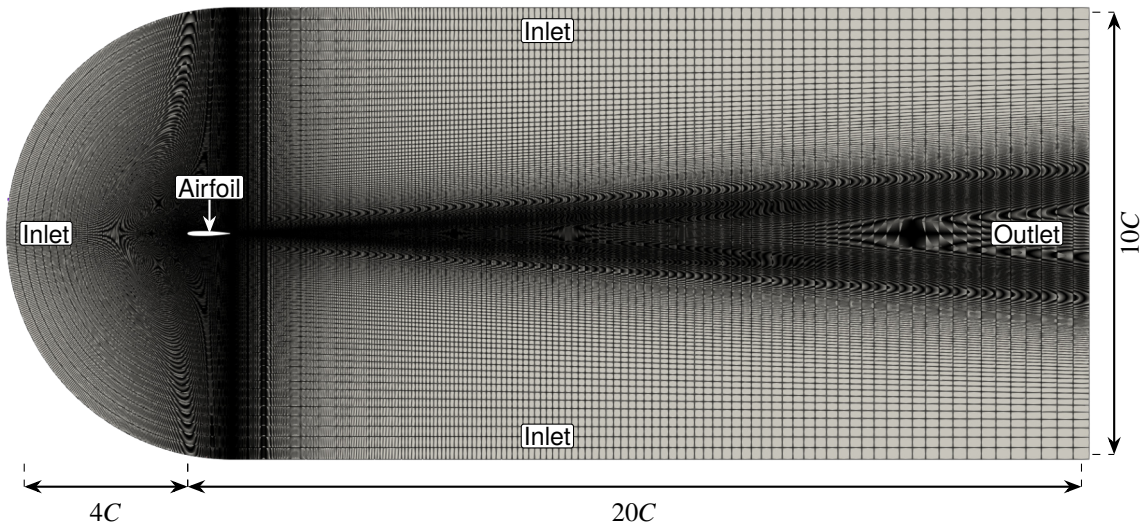


Figure 6.1: Computational domain

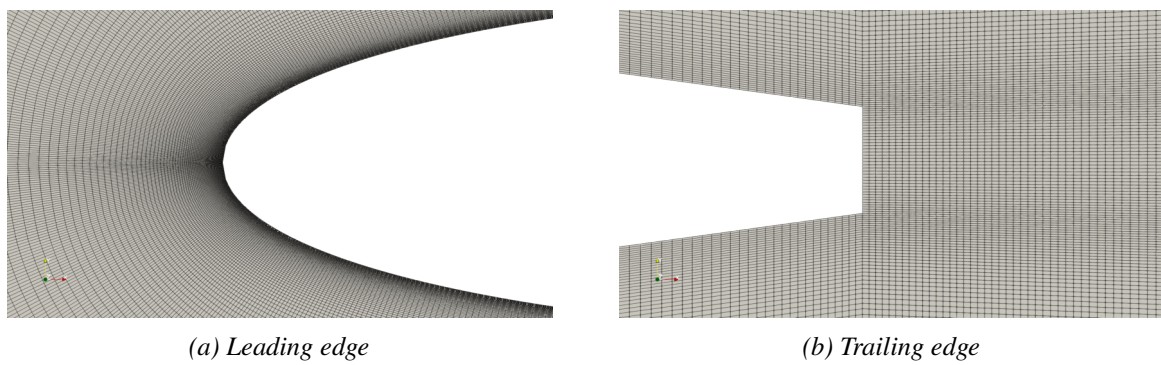


Figure 6.2: Closeup x-y view of the leading and trailing edges.

## 6.4 Wall Spacing

Taking the streamwise, spanwise and wall-normal wall spacing in wall units to be  $\Delta x^+$ ,  $\Delta z^+$  and  $\Delta y^+$  respectively, the following grid resolutions for a wall-resolved LES were recommended by Wagner et al. [18]:

- **High-resolution LES:**  $\Delta x^+ \leq 50$  and  $\Delta z^+ \leq 12$
- **Medium-resolution LES:**  $\Delta x^+ \leq 100$  and  $\Delta z^+ \leq 30$
- **Low-resolution LES:**  $\Delta x^+ \geq 100$  and  $\Delta z^+ \geq 30$

For all cases, the  $\min(\Delta y^+)$  is  $< 1$ . Similarly, Tucker [151] recommended  $50 \leq \Delta x^+ \leq 130$ ,  $15 \leq \Delta z^+ \leq 30$  and  $\min(\Delta y^+) = 1$  for a wall-resolved LES.

## 6.5 Mean Flow and Turbulence Intensity Results

The mean flow and turbulence intensity profiles are only shown for the case with  $Span = 0.3C$ . No significant difference was found between the three tested spans, although the fourth-order central differencing scheme appeared to be more sensitive to the spanwise extent. The mean flow and turbulence intensity profiles for the  $Span = 0.2C$  and  $Span = 0.6C$  can be found in Appendix B.1 and B.2 respectively.

Using the flow fields for the  $Span = 0.2C$  cases, the separation, reattachment and transition zones were determined and compared to the experimental data from Sagrado [149] and the BOFFS solver from Lin et al. [150]. The results are summarised in Table 6.2. From Table 6.2, it can be seen that all three schemes show good agreement with the experimental data with respect to predicting the separation, reattachment and transition regions.

Table 6.2: Separation, reattachment and transition regions

| Case         | Separation ( $x/C$ ) | Reattachment ( $x/C$ ) | Transition ( $x/C$ ) |
|--------------|----------------------|------------------------|----------------------|
| Experimental | 0.65                 | 0.97                   | 0.86 – 0.90          |
| BOFFS        | 0.76                 | 0.90                   | 0.80 – 0.84          |
| LUST         | 0.68                 | 0.97                   | 0.83 – 0.86          |
| Linear       | 0.69                 | 0.97                   | 0.82 – 0.85          |
| Cubic        | 0.67                 | 0.98                   | 0.83 – 0.86          |

Figure 6.3 shows the instantaneous streamwise velocity at the airfoil trailing edge. From Figure 6.3, the vortex shedding emanating from the blunt trailing edge can be seen. In the work done by Sagrado [149], vortex shedding from a blunt trailing edge was identified as the main contributor to narrowband noise and tones in the generation of airfoil trailing edge noise.

The normalised streamwise mean velocity profiles are shown in Figure 6.4.

At  $x/C = 0.65$ , Figure 6.4a shows an overall thicker boundary layer for the three tested numerical schemes and the BOFFS code compared to the experimental data. The three schemes agree well with each other and the BOFFS code up to approximately  $y/C \approx 0.0075$ , at which point the three schemes deviate from the BOFFS code. The three schemes agree well with the experimental data beyond  $y/C \approx 0.01$ .

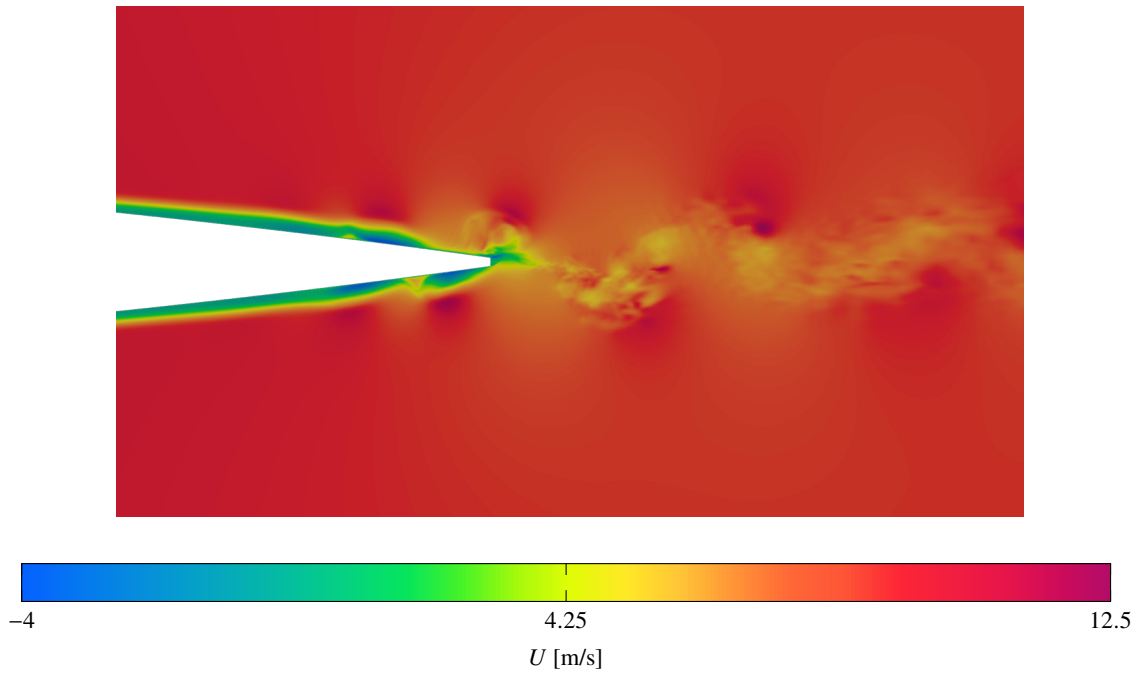


Figure 6.3: Instantaneous streamwise velocity near the trailing edge.

At  $x/C = 0.78$ , Figure 6.4b, the BOFFS code shows good agreement with the experimental data whilst the three tested numerical schemes exhibit a thicker boundary layer, with the fourth-order scheme showing the greatest deviation from the experimental data. All three of the tested schemes also show slight reversed flow near the airfoil surface.

At  $x/C = 0.86$ , Figure 6.4c shows the three tested schemes and the BOFFS code exhibiting a thicker boundary layer compared to the experimental data, with the fourth-order scheme once again showing the greatest deviation. Similar to  $x/C = 0.78$ , all three schemes also show slight reversed flow at this location.

At  $x/C = 0.98$ , Figure 6.4d shows good agreement with the three tested schemes and the experimental data, with the BOFFS code showing the greatest deviation from the experimental data.

In the wake, at  $x/C = 1.02$ , Figure 6.4e shows all three tested schemes in good agreement with each other, but overpredict the minimum peak compared to the experimental data. The BOFFS code showed the closest agreement with the experimental data.

At  $x/C = 1.05$ , Figure 6.4f once again shows the tested numerical schemes overpredicting the minimum, with the second-order central differencing scheme showing the greatest overprediction.

Figure 6.5 shows contours of turbulence intensity near the trailing edge for the cubic numerical scheme. An increase in turbulence intensity is observed in the vicinity of the trailing edge. According to Lin et al. [150], this increase is associated with the development of an adverse pressure gradient, which causes laminar boundary-layer separation. The resulting separated shear layer becomes susceptible to Kelvin–Helmholtz instability, leading to the turbulent transition.

Figure 6.6a shows that the three tested schemes and the BOFFS code underpredict the turbulence intensity compared to the experimental data at  $x/C = 0.65$ . The cubic scheme shows two noticeable peaks at

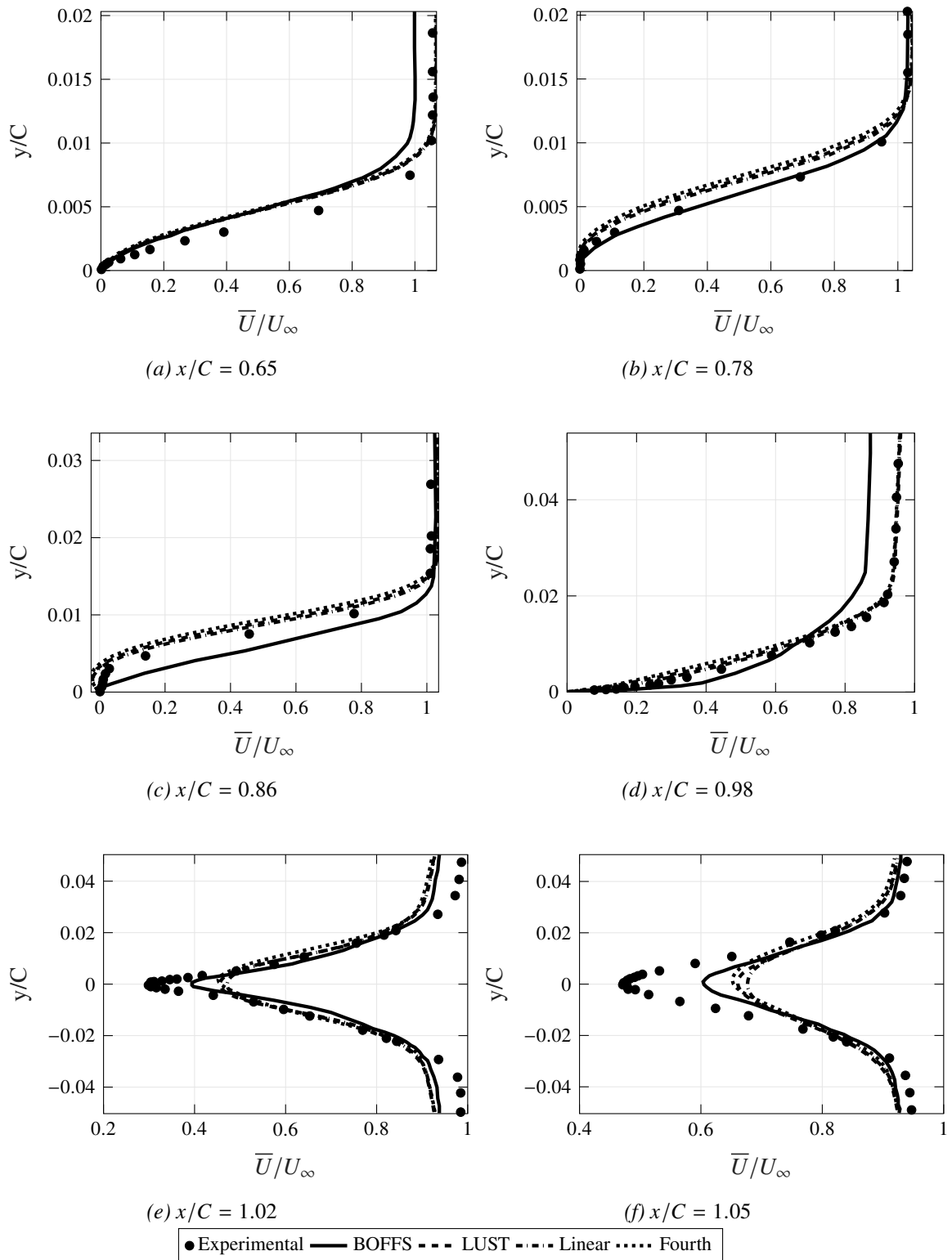


Figure 6.4: Normalised streamwise mean velocity profiles for  $\text{Span} = 0.3C$

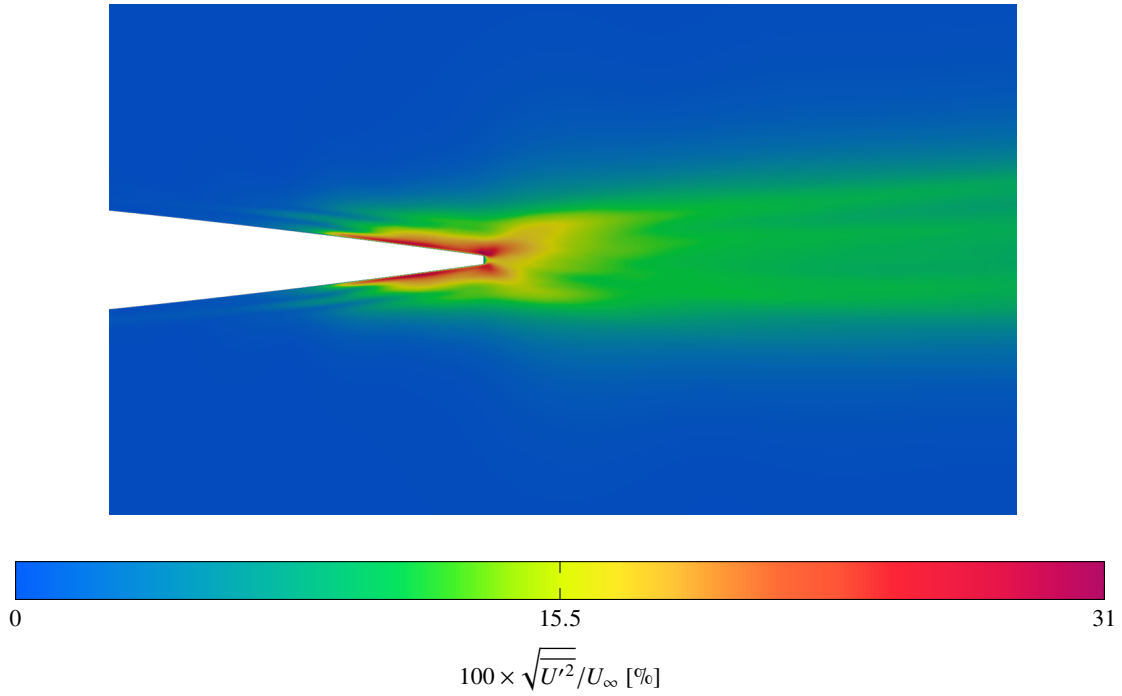


Figure 6.5: Turbulence intensity near the trailing edge for the Cubic scheme.

$y/C = 0.02$  and  $y/C = 0.0047$ . The experimental data also exhibits two peaks in the same vicinity but at a higher turbulence intensity.

Figure 6.6b also shows the three tested schemes underpredicting the turbulence intensity at  $x/C = 0.78$ , with the cubic scheme still showing two noticeable peaks. The BOFFS code overpredicts the turbulence intensity compared to the experimental data.

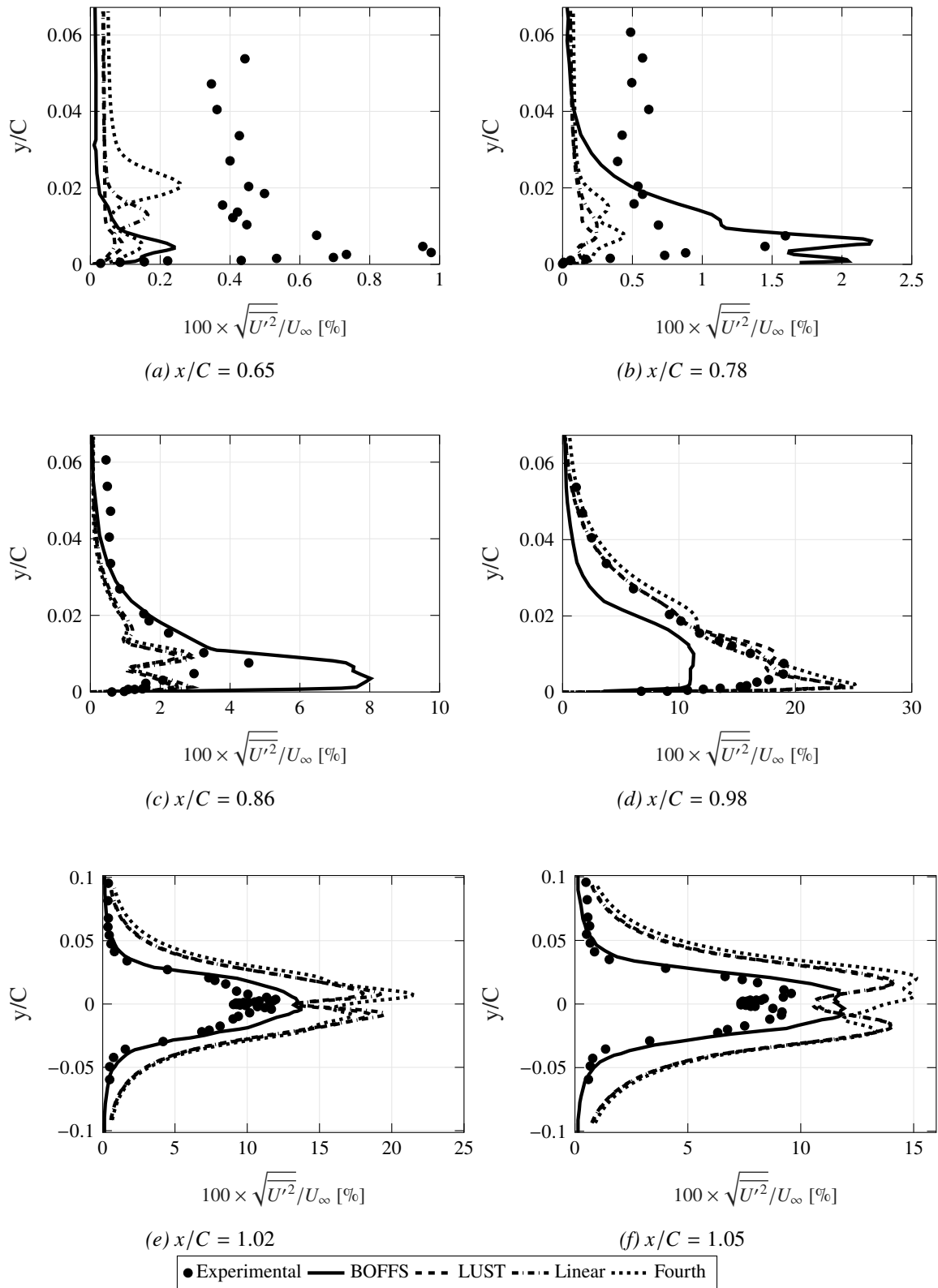
Figure 6.6c shows good agreement between the three tested schemes at  $x/C = 0.86$ , with two noticeable peaks being shown for each. At this location, the peaks start to move closer to the experimental data, although still underpredicting it. The BOFFS code appears to significantly overpredict the peak turbulence intensity.

Figure 6.6d shows the three schemes in good agreement with the experimental data at  $x/C = 0.98$ . The width of the peak is captured well but there is a slight overprediction below  $y/C = 0.0047$ . At this location, the BOFFS code underestimates both the width and magnitude of the peak.

At both wake locations ( $x/C = 1.02$  and  $x/C = 1.05$ ), Figures 6.6e and 6.6f both show that all three schemes overpredict both the width and magnitude of the peak turbulence intensity compared to the experimental data. The BOFFS code also overpredicts the magnitude of the peaks, but to a lesser degree, and also captures the width of the peaks well.

## 6.6 Computational Cost

Figure 6.7a shows the run time for the three numerical schemes and domain sizes, whilst Figure 6.7b shows the computational cost ( $hours \times cores$ ). From Figure 6.7, it can be seen that the LUST scheme on average costs 2.19% more than the Linear scheme. The fourth-order scheme costs 45.73% more than the

Figure 6.6: Turbulence intensity profiles for  $\text{Span} = 0.3C$



LUST scheme and 48.92% more than the Linear scheme on average. The fourth-order scheme did not show any improvement over either the LUST or Linear schemes in Section 6.5, therefore the significant extra cost of using this scheme does not justify the cost, for the tested cases.

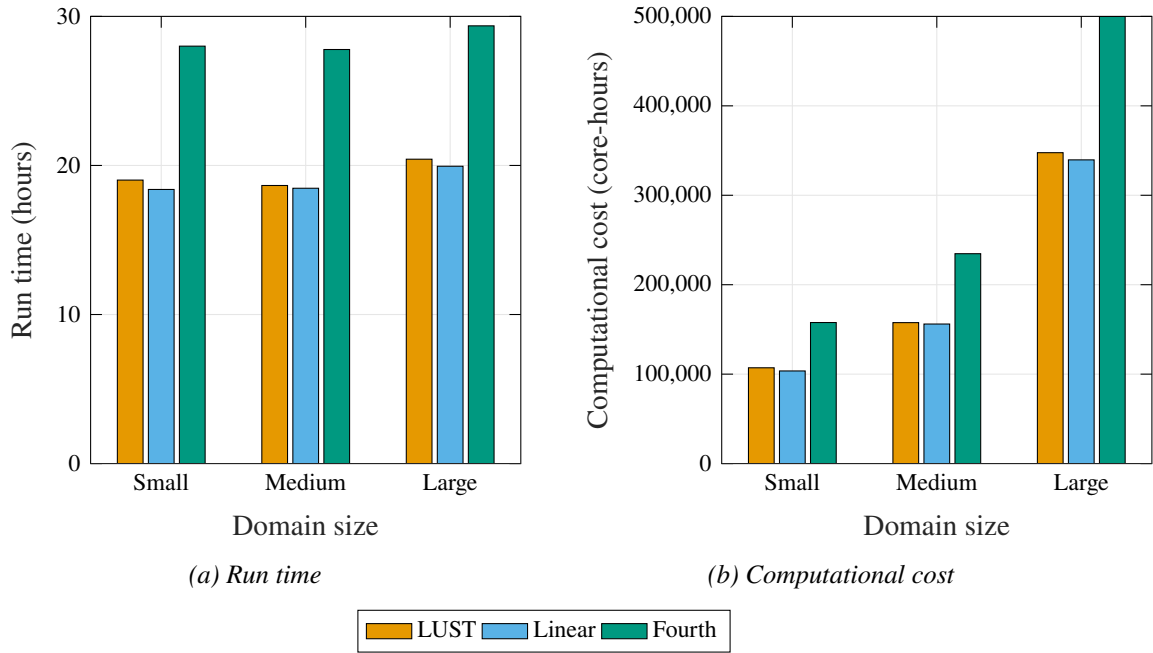


Figure 6.7: Run time and computational cost for different discretisation schemes and domain sizes.

## 6.7 Summary

In this chapter, OpenFOAM's transient compressible solver, rhoPimpleFoam, was validated for a 3-D wall-resolved LES case using the experimental data of Sagrado [149] and the LES results of Lin et al. [150]. The validation focused on assessing whether the selected OpenFOAM setup could capture the main flow features of the blunt trailing edge NACA 0012 case, including separation, reattachment, transition and vortex shedding.

Three spatial discretisation schemes were assessed, namely LUST, second-order Linear central differencing and a fourth-order Cubic scheme. In addition, three spanwise domain sizes were considered. The predicted separation, reattachment and transition locations showed good agreement with the experimental and BOFFS reference data. Separation was predicted between  $x/C = 0.67$  and  $0.69$ , reattachment between  $x/C = 0.97$  and  $0.98$ , and transition between approximately  $x/C = 0.82$  and  $0.86$ . These values compared well with the experimental separation location of  $x/C = 0.65$ , reattachment location of  $x/C = 0.97$ , and transition region of  $x/C = 0.86$  to  $0.90$ . The instantaneous streamwise velocity contours also captured vortex shedding from the blunt trailing edge.

The mean streamwise velocity profiles showed mixed but generally reasonable agreement with the experimental data. The OpenFOAM results tended to predict a thicker boundary layer on the airfoil surface, particularly at  $x/C = 0.78$  and  $x/C = 0.86$ , with the Cubic scheme showing the largest deviation. However, good agreement was obtained near the trailing edge at  $x/C = 0.98$ . In the wake, the schemes agreed well with each other, although the minimum streamwise velocity was overpredicted relative to the experimental measurements.

The turbulence intensity profiles showed larger discrepancies, particularly in the upstream portion of the separated region where the turbulence intensity was underpredicted. Agreement improved closer to the trailing edge, with the profiles at  $x/C = 0.98$  capturing the location and width of the turbulence intensity peak reasonably well. In the wake, the schemes overpredicted both the width and magnitude of the turbulence intensity peaks. Nevertheless, the increase in turbulence intensity near the trailing edge was captured, consistent with the development of an adverse pressure gradient, laminar separation, Kelvin–Helmholtz instability and subsequent transition.

The computational cost comparison showed that the LUST scheme was only 2.19% more expensive than the Linear scheme on average, while the Cubic scheme was significantly more expensive, costing 45.73% more than LUST and 48.92% more than Linear. Since the Cubic scheme did not provide a clear improvement in either the mean velocity or turbulence intensity predictions, the additional cost was not justified for the cases considered.

The results demonstrate that rhoPimpleFoam is capable of capturing the dominant flow features of the 3-D LES validation case. Although discrepancies were observed in the predicted boundary-layer thickness and wake turbulence intensity, the OpenFOAM results showed acceptable agreement with the experimental and BOFFS reference data. Based on both accuracy and computational cost, the lower-order schemes, particularly the Linear and LUST schemes, provide a more practical choice for subsequent LES simulations than the fourth-order Cubic scheme.

# 7 NACA 0012 FLOW AND AEROACOUSTICS

Urban Air Mobility Vehicles (UAMVs), or air-taxis, require installation-noise modelling approaches that account for both rotor and airframe contributions. In this chapter, the flow and acoustic characteristics of a NACA 0012 airfoil at  $Re = 200,000$  and  $0^\circ$  angle of attack are investigated using Large-Eddy Simulation (LES), resolving up to a Strouhal number of 75. Following experimental studies, a zig-zag trip is implemented and compared with straight-trip and untripped configurations. The performance of LES, a new Zonal-LES (ZLES) model, and the  $k - \omega$  SST Improved Delayed Detached Eddy Simulation (IDDES) model is assessed. Curle and Ffowcs Williams-Hawkings (FW-H) acoustic analogies are compared for sound propagation in the low-Mach-number regime expected for UAMVs, with nozzle effects determined using the Curle analogy. For the LES case with the zig-zag trip, the influence of FW-H surface and closure-disc placement is also examined. Finally, Curle, Ffowcs Williams-Hawkings (FW-H) 1A, and GT (Garrick Triangle) analogies are compared alongside the Brooks, Pope, and Marcolini semi-empirical predictions [5].

## 7.1 Methodology

### 7.1.1 Numerical Methods

The simulations were conducted in OpenFOAM v2212 using the rhoPimpleFoam solver. In all cases, temporal discretisation was performed using the second-order backward difference scheme. For spatial discretisation, the LUST scheme was employed. For the standard LES simulations, the WALE [137] LES model was used, with a Subgrid-Scale (SGS) filter width equal to the cube root of the cell volume. For the IDDES simulations, the standard  $k - \omega$  SST IDDES model was used [136]. The simulations were run for  $74.67T^*$  with temporal statistics conducted over  $26.67T^*$ , where  $T^* = C/U_\infty$ . The transient timestep for all cases was  $3.33 \times 10^{-4}T^*$ .

### 7.1.2 Computational Domain

The airfoil used for the simulations was a NACA 0012 with a sharp trailing edge and a chord length of 0.15 m to match the experiments. The leading edge of the airfoil is located at (0,0). Meshing was conducted using Pointwise 2022.2.2. The computational domain extended 10 chord lengths above and below the airfoil and 40 chord lengths in the wake, as shown in Figure 7.1. The computational domains for the IDDES and ZLES cases had approximately half the resolution of the LES computational domains in the spanwise direction (74 grid points across the span compared to 145). All the grids had a maximum stretching ratio of 5% in the wall normal direction and 2% in the wake. The grid sizes and resolutions are summarised in Table 7.1.

### 7.1.3 Trip Geometry

Tripping is generally used in experiments to force the boundary layer to transition to turbulence, which is especially useful for low-speed wind tunnels in which the natural transition may be delayed or inhibited. This allows for the investigation of turbulent boundary layer noise. The zig-zag trip tape used in the experiments is shown in Figure 7.2a. The tape had a width of 9 mm and a height of 0.5 mm. For the CFD,

Table 7.1: Simulation parameters for the different turbulence models and spans.

| Case          | Element Count | max ( $\Delta x^+$ , $\Delta y^+$ , $\Delta z^+$ ) |
|---------------|---------------|----------------------------------------------------|
| LES_untripped | ~40M          | 50, 1, 15                                          |
| LES_straight  | ~40M          | 50, 1, 15                                          |
| LES_zigzag    | ~40M          | 50, 1, 15                                          |
| ZLES_zigzag   | ~21M          | 50, 1, 30                                          |
| IDDES_zigzag  | ~21M          | 50, 1, 30                                          |

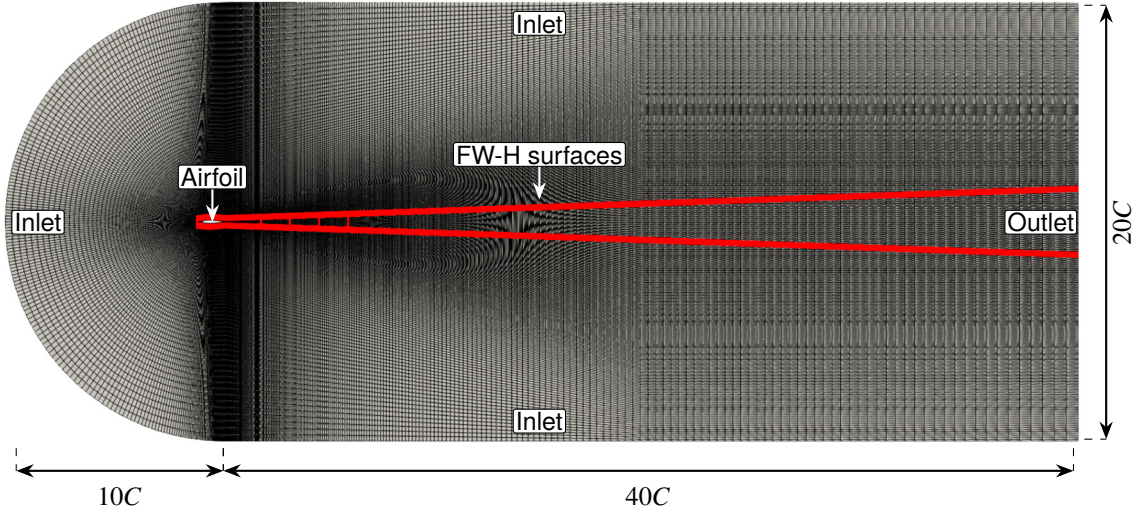


Figure 7.1: Computational domain

this was modelled using an approximated block pattern. To keep  $\Delta z$  constant for the CFD, blocks were arranged to mimic the zigzag pattern of the tape, also shown in Figure 7.2a. As per the experiment, the trip was located at a distance of  $x/C = 0.2$  from the leading edge of the airfoil. The straight trip also had a height of 0.5 mm but a width of 3 mm, corresponding to the width of a single block from the zigzag trip. The trips on the airfoil are shown in Figure 7.2b. The IDDES and ZLES meshes had 4 spanwise points across each block, while the LES resolution meshes had 7 spanwise points. Further coarsening in the spanwise direction would have resulted in too few points across the tripping blocks, hence the IDDES and ZLES cases were only tested with half the spanwise resolution of the LES mesh.

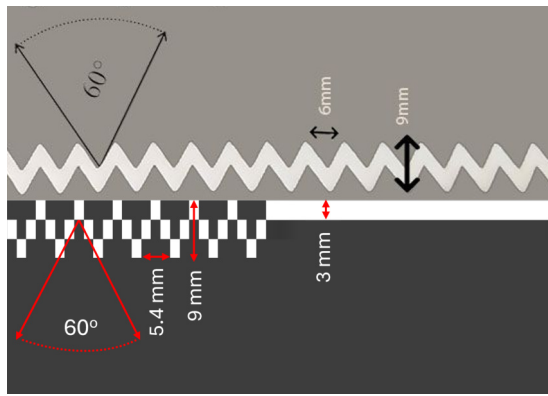
LES performed on the untripped airfoil showed a boundary layer height of 0.8 mm at the trip location, resulting in  $k/\delta_k = 0.625$ , which falls in the range  $0.5 < k/\delta_k < 1.05$  recommended by Santos et al. [54].

#### 7.1.4 Boundary Conditions

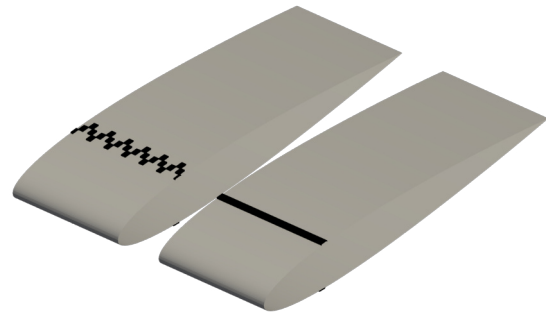
At the inlet, the velocity was set to  $U_\infty = 20$  m/s, corresponding to a chord-based Reynolds number of  $Re = 200,000$  and a Mach number of  $M = 0.058$ , while a zero-gradient condition was set for pressure. At the outlet, a zero-gradient condition was applied for velocity, while the pressure was set to  $p = 101,306.12$  Pa. A no-slip condition was applied at the walls, and spanwise periodicity was used.

#### 7.1.5 Acoustics

For the acoustics section, various acoustic analogies were compared to experimental data obtained from the Brunel University of London Anechoic Chamber [152]. The first was the Curle analogy [27], which is



(a) Trip shapes: Experimental (top), CFD zigzag (bottom left) and CFD straight (bottom right)



(b) Trip shapes shown on airfoil surface: CFD zigzag (left) and CFD straight (right)

Figure 7.2: Comparison of trip geometries and their placement on the airfoil.

an extension of Lighthill's acoustic analogy [72] to include the effects of solid boundaries. The second was the Farassat 1A formulation [30] of the Ffowcs Williams-Hawkings (FW-H) equation [29]. The third was the Garrick Triangle (GT) formulation [31] of the FW-H equation, which assumes a stationary source and observer in uniform flow. Lastly, the Brooks prediction [5] was used to provide a semi-empirical comparison, and the code used for this has been validated by Lee et al. [32].

The microphone array used in the experiments is shown in Figure 7.3. For the CFD, the observer positions were set to coincide with the microphone locations.

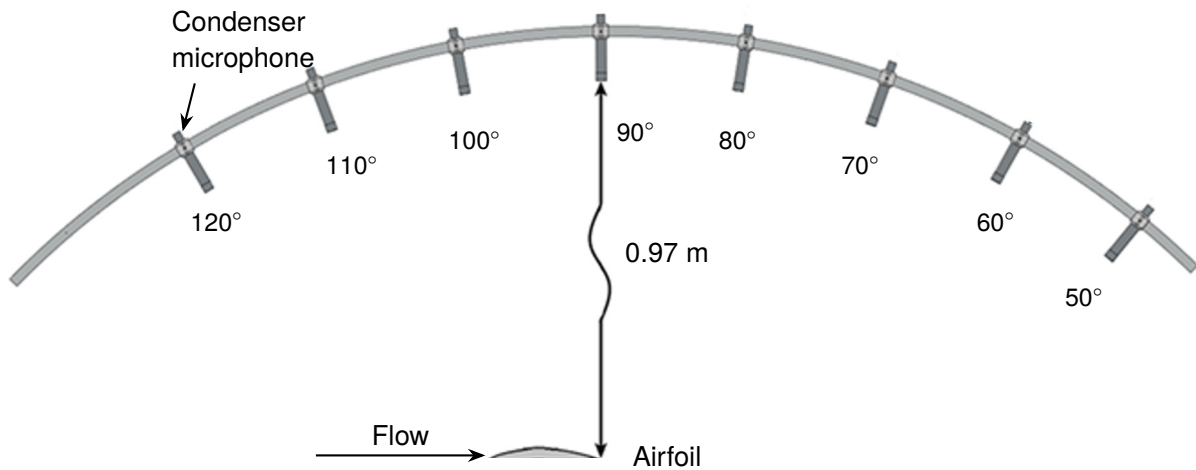


Figure 7.3: Microphone array

Four different heights were used for both the closed and opened surfaces, each separated by a distance of  $y/C = 0.067$ , with the nearest surface also being  $0.067C$  above and below the airfoil surface. All surfaces extend  $2.67C$  upstream of the airfoil leading edge, and as there is no turbulence have an upstream closing disc. The downstream closing disc locations are shown in Figure 7.4, whilst the opened surfaces extend to



the end of the domain (40 chord lengths downstream of the airfoil trailing edge). All surfaces have the same uniform resolution, with every point on the surface being  $C/37.5$  apart.

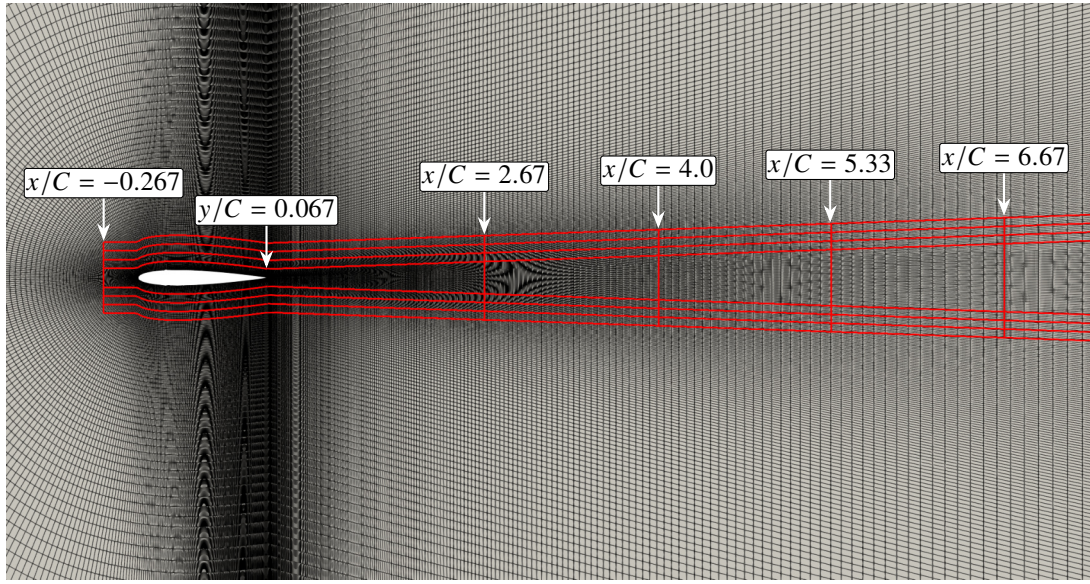


Figure 7.4: FW-H surfaces showing closing disc locations

### 7.1.6 Noise Rectification for Span Variation

Due to the limited span of the CFD which in itself is different to that of the experiment, the total noise emitted per unit span must be accounted for. The corrections applied depend on the coherence of the turbulence at the trailing edge and are detailed below.

According to Kato et al. [153], for every distinct frequency,  $f$ , the coherence length,  $L_c$ , is the length at which the coherence function (Equation 7.1) drops below 0.5. The magnitude-squared coherence function is shown in Equation 7.1:

$$\Gamma_{ij}^2 = \frac{|S_{ij}|^2}{S_i S_j} \quad (7.1)$$

With respect to spanwise coherence,  $S_i$ ,  $S_j$  and  $S_{ij}$  represent the Power Spectral Density (PSD) of the fluctuating surface pressure at the spanwise reference point, point of comparison and cross-spectral density between the two signals respectively.

Based on the coherence function, the relevant corrections can then be applied. The corrections can be found in [153].

$$L_c(f) \leq L_{\text{sim}}, \quad SPL(f) = SPL_{\text{sim}}(f) + 10 \log \left( \frac{L_{\text{exp}}}{L_{\text{sim}}} \right) \quad (7.2a)$$

$$L_{\text{sim}} < L_c(f) < L_{\text{exp}}, \quad SPL(f) = SPL_{\text{sim}}(f) + 20 \log \left( \frac{L_c(f)}{L_{\text{sim}}} \right) + 10 \log \left( \frac{L_{\text{exp}}}{L_c(f)} \right) \quad (7.2b)$$

$$L_c(f) \geq L_{\text{exp}}, \quad SPL(f) = SPL_{\text{sim}}(f) + 20 \log \left( \frac{L_{\text{exp}}}{L_{\text{sim}}} \right) \quad (7.2c)$$

A limitation of the Kato et al. [153] correction is that it assumes that the spanwise coherence is either fully coherent or fully incoherent. Seo and Moon [154] proposed a similar method that offers a smoother transition between coherent and incoherent limits, in which a Gaussian fit can be used to determine the coherence length,  $L_c(f)$ :

$$\Gamma_{ij}^2(\Delta z) = e^{-\left( \frac{\Delta z_{ij}^2}{L_c(f)^2} \right)} \quad (7.3)$$

The relevant corrections can then be applied according to Equation 7.4:

$$\sqrt{\pi} L_c(f) \leq L_{\text{sim}}, \quad SPL(f) = SPL_{\text{sim}}(f) + 10 \log \left( \frac{L_{\text{exp}}}{L_{\text{sim}}} \right) \quad (7.4a)$$

$$L_{\text{sim}} < \sqrt{\pi} L_c(f) < L_{\text{exp}}, \quad SPL(f) = SPL_{\text{sim}}(f) + 10 \log \left( \frac{L_c(f)}{L_{\text{sim}}} \right) + 10 \log \left( \sqrt{\pi} \frac{L_{\text{exp}}}{L_{\text{sim}}} \right) \quad (7.4b)$$

$$\sqrt{\pi} L_c(f) \geq L_{\text{exp}}, \quad SPL(f) = SPL_{\text{sim}}(f) + 20 \log \left( \frac{L_{\text{exp}}}{L_{\text{sim}}} \right) \quad (7.4c)$$

For fully coherent or fully incoherent flows, there should be no difference between the Kato et al. [153] and Seo and Moon [154] corrections. However, for transitional flow that contains laminar regions, laminar separation bubbles and turbulent reattachment regions, such as the work done by du Plessis et al. [155], the Seo and Moon [154] correction provides a more physically consistent and robust approach.

## 7.2 Flow Results

Streamwise mean velocity ( $U_{\text{mean}}$ ) and Turbulence Intensity (TI) profiles are provided for two boundary layer and two wake locations. The boundary layer profiles were measured every 5 mm, starting at 30 mm from the trailing edge ( $x/C = 0.8$ ) and ending at the trailing edge ( $x/C = 1.0$ ). The wake profiles were measured at 1 mm and 5 mm aft of the trailing edge,  $x/C = 1.01$  and  $x/C = 1.03$  respectively. For brevity, results are shown for two boundary layer and two wake locations. All data is normalised by the maximum velocity within the probe range, to accommodate for nozzle effects in the experiment. To reduce the effect of a shorter temporal window in the CFD, the CFD data is also averaged in the spanwise direction. The resolved turbulence intensity is calculated by  $TI = 100 \times \frac{\sqrt{\overline{U'^2}}}{U_\infty}$ , where  $\overline{U'^2}$  is determined by Equation 7.5:

$$\overline{U'^2} = \frac{1}{N} \sum_{i=1}^N (U_i - \bar{U})^2 \quad (7.5)$$

The modelled TI for the IDDES case is determined by  $TI = 100 \times \frac{\sqrt{\frac{2}{3}k}}{U_\infty}$ .

### 7.2.1 Effect of Tripping and Trip Shape

Figure 7.5 shows the normalised Q-criterion iso-surfaces ( $Q_{norm}$ ), coloured by mean velocity magnitude, where  $Q_{norm} = Q \times C^2/U^2$ . The value of  $Q_{norm}$  was chosen to be 5. Both tripped cases look visually similar except near the tripping location, with the zig-zag trip (Figure 7.5c) giving more spatially coherent structures which exist until approximately  $C = 0.75$ . The untripped case (Figure 7.5a) shows laminar spanwise structures until transition near the trailing edge.

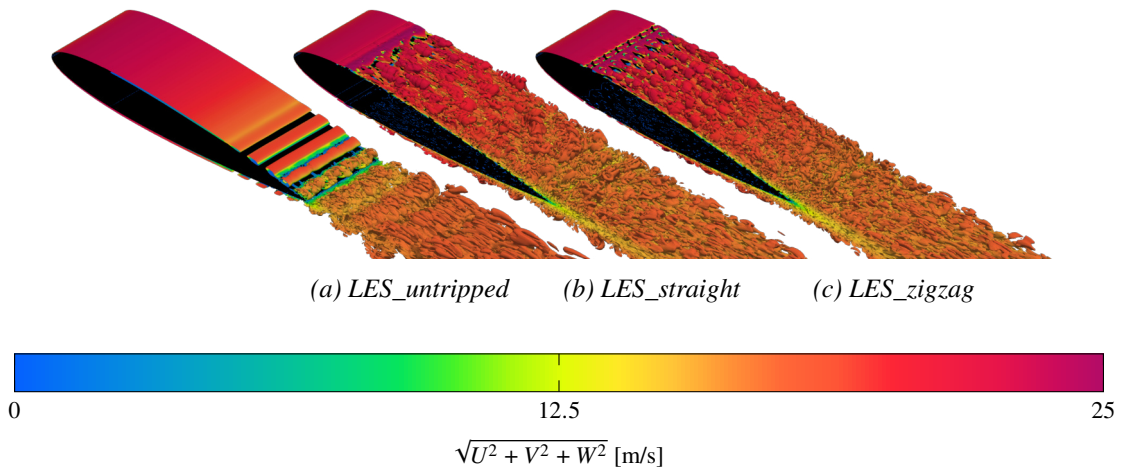


Figure 7.5:  $Q$ -criterion iso-surfaces coloured by mean velocity magnitude,  $Q_{norm} = 5$ .

Figure 7.6a shows a good agreement for mean velocity profiles between the experimental, LES\_straight and LES\_zigzag cases, whilst the untripped case shows a much flatter profile with slightly reversed flow near the wall. Figure 7.6b shows a thickening of the boundary layer at the trailing edge, with good agreement shown between all four cases. Figures 7.6c and 7.6d broadly show similar agreement with the untripped profile making small departures from the measurements at  $y/C > 0.025$ .

The  $\delta^*$  shown in Figure 7.7a shows good agreement between the LES\_straight and LES\_zigzag cases, and the trend of these two cases also agrees well with the experimental data. There is an increase in  $\delta^*$  towards the trailing edge. Figure 7.7b shows a sharp decrease in shape factor,  $H = \delta^*/\theta_{BL}$  for the LES\_untripped at  $x/C = 0.9$ , reaching similar levels as the experimental data at  $x/C = 1$ . For flat plate boundary layers, Schlichting and Gersten [156] found a shape factor of  $H = 1.4$  for fully turbulent boundary layers, agreeing well with the tripped cases towards the trailing edge.

Figure 7.8a shows an overprediction in Turbulence Intensity (TI) for both the LES\_straight and LES\_zigzag cases compared to the experiment, whilst the LES\_untripped case shows very low levels of TI. At the trailing edge, there is once again higher TI for the LES\_straight and LES\_zigzag, whilst the LES\_untripped case shows much higher turbulence intensity over a wider region above the airfoil surface. This indicates an energetic but delayed laminar-turbulent transition as seen in Figure 7.5. Note that the experiment TI does not tend to zero due to presence of the nozzle.



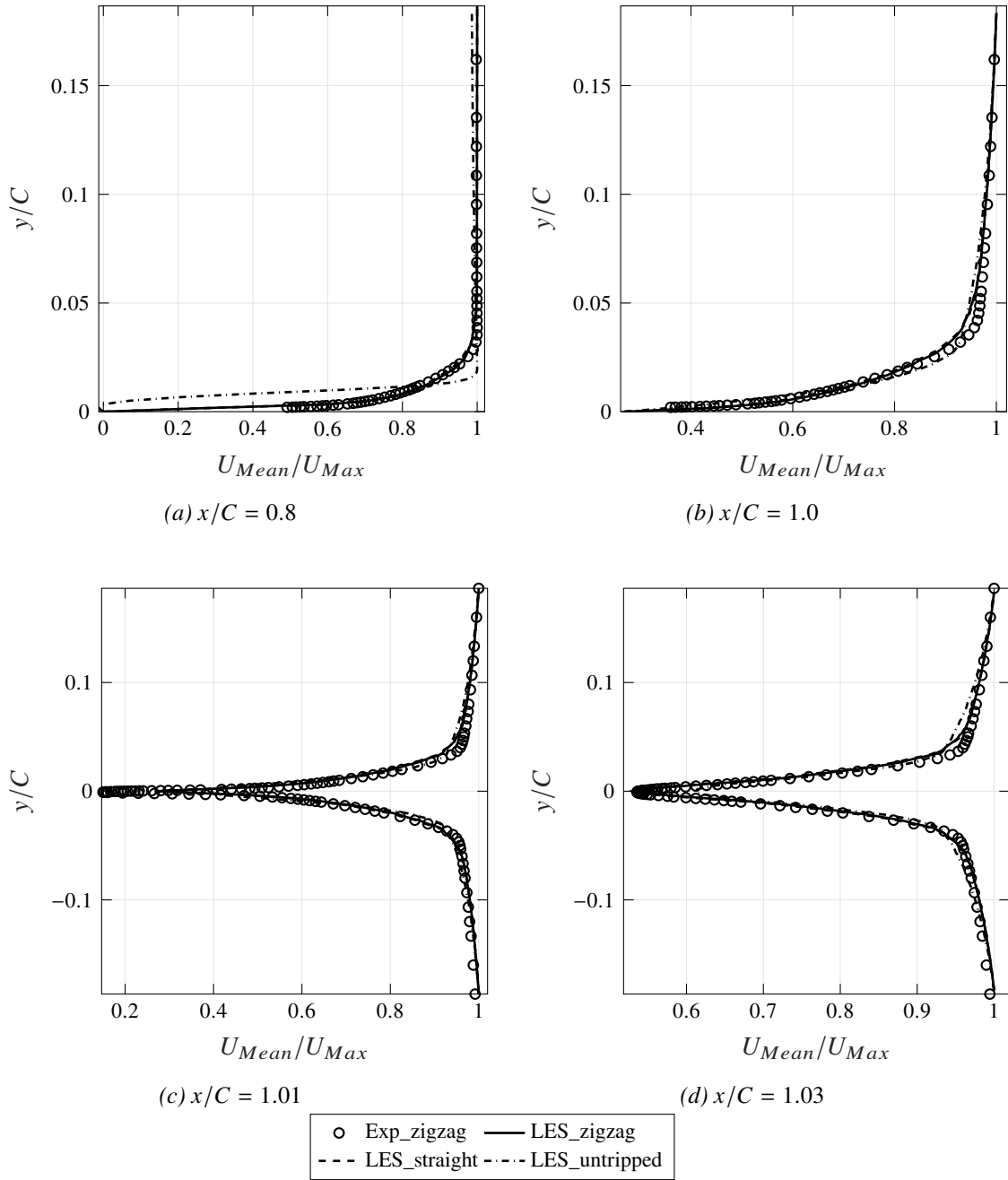


Figure 7.6: Normalised streamwise mean velocity showing the effect of tripping and trip shape.

The wake profiles (Figure 7.8c) show a trend of two peaks and a central minimum between the two peaks. The LES\_straight and LES\_zigzag overpredict the peaks and the central minimum compared to the experiment whilst the LES\_untripped shows much higher TI and wider peaks. The trend remains the same further downstream (Figure 7.8d), although the LES\_straight and LES\_zigzag appear to predict the minimum between the peaks much better.

Figure 7.9 shows the comparison of the mean wall shear stress between the LES\_zigzag and the LES\_straight cases. Figure 7.9a shows an imprint of the zigzag shape extending downstream of the trip, resulting in alternating regions of high and low wall shear stress. Figure 7.9b shows a near uniform wall shear stress across the span for the LES\_straight case.

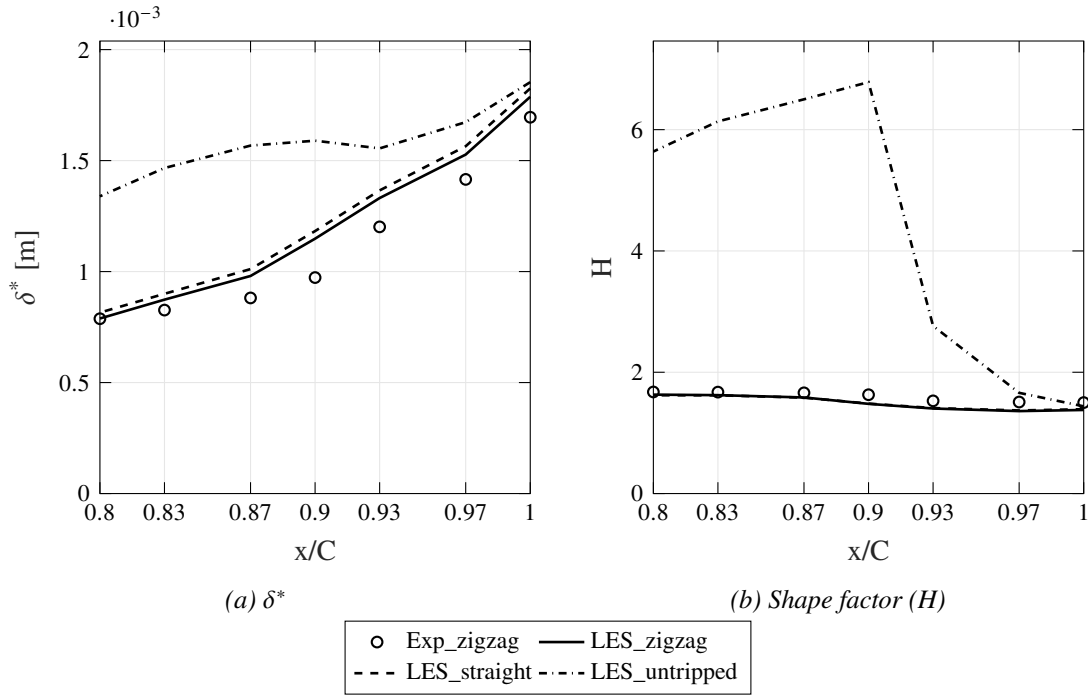


Figure 7.7: Displacement thickness and shape factor at different  $x/C$  locations

The variation of mean wall shear stress ( $\overline{\tau_w}$ ) and RMS surface pressure ( $p_{RMS}$ ) is shown in Figure 7.10. Figure 7.10a shows that  $\overline{\tau_w}$  fluctuates by 1.59 Pa for the LES\_zigzag case whilst only fluctuating by 0.11 Pa for the LES\_straight case at  $x/C = 0.3$ . At  $x/C = 0.9$ ,  $\overline{\tau_w}$  fluctuates by 0.26 Pa for LES\_zigzag and 0.09 Pa for LES\_straight. However, the average  $\overline{\tau_w}$  along the span only differs by 0.05 Pa and 0.0085 Pa between the two types of trip at  $x/C = 0.3$  and  $x/C = 0.9$  respectively.

A similar trend is noted for the surface  $p_{RMS}$ , where the LES\_zigzag has a range of 12.32 Pa and the LES\_straight has a range of 1.26 Pa at  $x/C = 0.3$ . At  $x/C = 0.9$ , the range drops to 0.33 Pa and 0.31 Pa for the LES\_zigzag and LES\_straight cases respectively.

The examination of  $\overline{\tau_w}$  and surface  $p_{RMS}$  indicate that the flow becomes relatively similar towards the trailing edge, regardless of the trip shape, however this will be case-dependent.

## 7.2.2 Effect of Turbulence Modelling

For the zonal LES cases, the following values were used:  $y_{RANS}^+ = 30$  and  $y_{LES}^+ = 60$ . At  $x/C = 0.9$ , the computational domain has 24 wall-normal grid points in the RANS region and 12 in the blending region, showing sufficient resolution between the zones. For  $42.67T^*$ , the ZLES\_zigzag costed 37,598.4 CPU-hours, while the LES\_zigzag costed 69,724.24 CPU-hours, approximately 1.85 times more. Therefore, the ZLES should offer at least a 46% computational saving compared to LES. Tyacke and Tucker [157] showed that for airframes, the number of grid points required for LES scales with  $Re^2$ , whilst for hybrid RANS-LES, it scales with  $Re^{0.4}$ . At the higher  $Re$  of a full-scale UAMV (assuming  $Re = 1 \times 10^6$ ), and assuming this scaling remains applicable, the saving is expected to be approximately 96%.

Figures 7.11a and 7.11b show good agreement between the experimental, LES\_zigzag and ZLES\_zigzag cases, whilst the IDDES\_zigzag shows a thicker boundary layer at both locations. In the wake, Figure

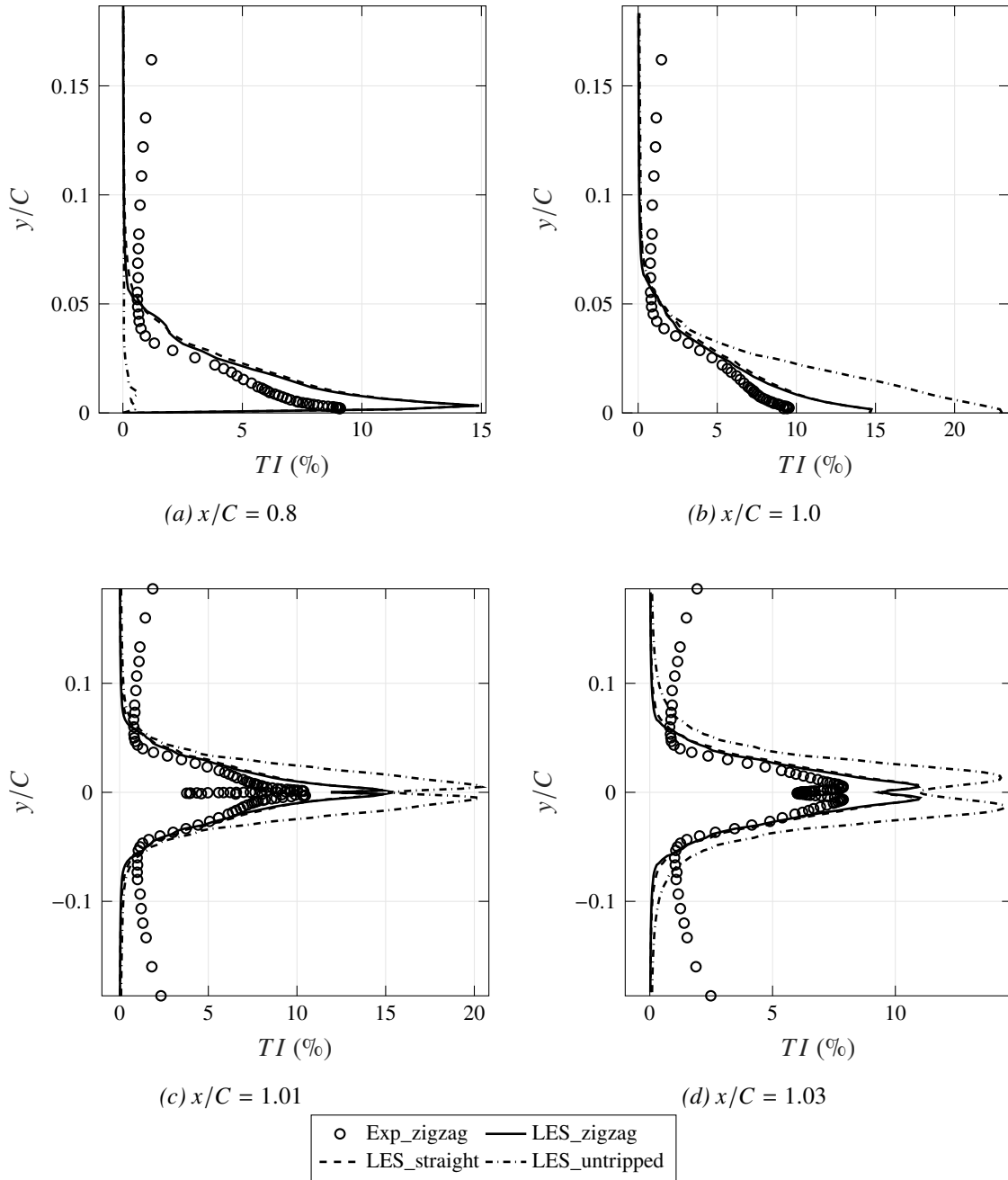


Figure 7.8: Turbulence intensity showing the effect of tripping and trip shape.

7.11c shows wider peaks for the IDDES\_zigzag, with a much lower minimum shown in Figure 7.11d. Figures 7.12a and 7.12b show that the IDDES\_zigzag captures the trend in  $\delta^*$  and  $H$  but at a higher level.

The TI profiles shown in Figure 7.13 show good agreement between the LES\_zigzag and ZLES\_zigzag with overprediction in the peaks by the LES\_zigzag and ZLES\_zigzag at  $x/C = 1.0$ ,  $1.01$  and  $1.03$ . The modelled TI for the IDDES\_zigzag underpredicts the peaks compared to the experiment, whilst the resolved TI remains near 0 at all locations. This lack of resolved turbulence is responsible for poor acoustic results. The peak in the IDDES\_zigzag also appears at an elevated position relative to the wall leading to an overprediction of boundary layer thickness.

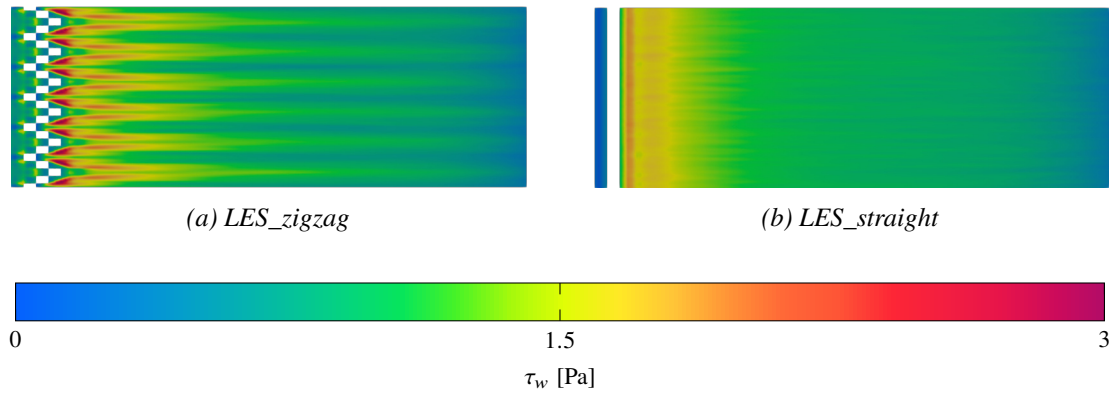


Figure 7.9: Comparison of wall shear stress for the two different trip shapes.

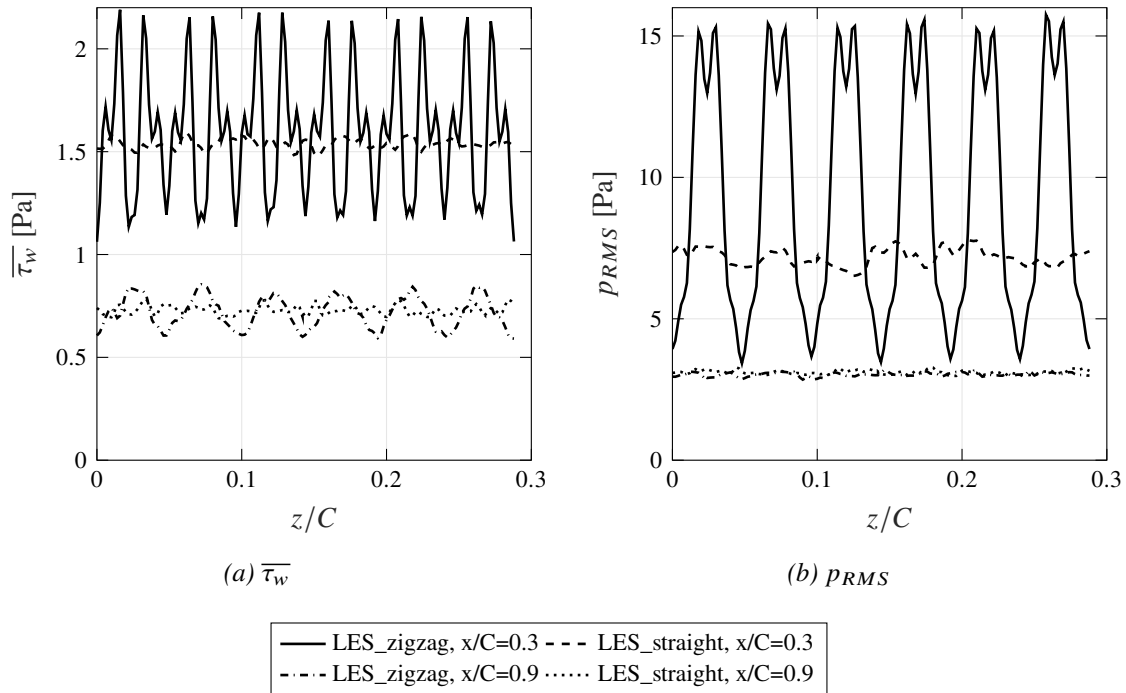


Figure 7.10: Mean wall-shear stress and surface pressure RMS distributions along  $z/C$  for different trip geometries.

Figure 7.14a shows a transition to turbulence at  $x/C \approx 0.86$ , which agrees well with experimental data from Sagrado [149], at the same  $Re$  for a blunt trailing-edge NACA 0012. Figure 7.14b shows a high turbulence intensity aft of the trip for the LES\_straight case, which is not apparent in Figure 7.14c and Figure 7.14d, which show a more controlled transition to turbulence for the LES\_zigzag and ZLES\_zigzag cases respectively. However, further downstream, all three cases exhibit similar levels of turbulence intensity, indicating a tendency to relaminarise at the low Reynolds number studied.

Figure 7.14e shows no resolved turbulence. However, it should be noted that from the IDDES\_zigzag case, the boundary layer thickness slightly upstream of the trip, at  $x/C = 0.18$ , was found to be 0.8 mm, which is greater than the trip height of 0.5 mm. The  $y^+$  at the trip height at  $x/C = 0.18$  was found to be 36, which lies in the log-law region. Therefore, since the IDDES aims to model the entire boundary layer using RANS, it is expected that there would be no resolved turbulence. Figure 7.14f, however, does show

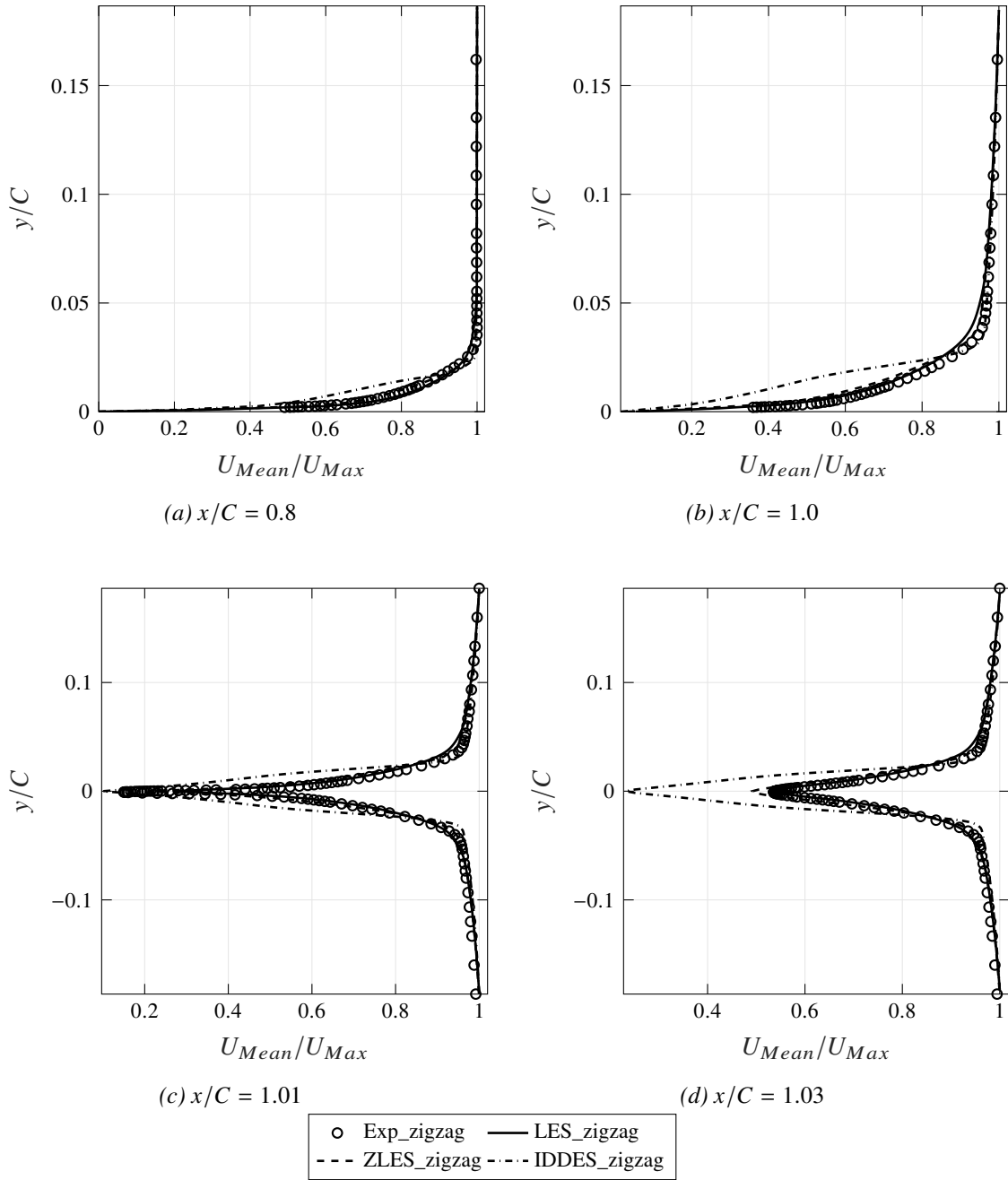


Figure 7.11: Normalised streamwise mean velocity showing the effect of turbulence modelling.

a turbulent boundary layer, albeit at a lower modelled intensity compared to the LES and ZLES cases.

Figure 7.15 shows contours of the empirical blending function  $\tilde{f}_d$ . From the figure, it can be seen that the boundary layer is modelled using RANS (as expected), however, upstream and downstream of the trip, there is a secondary increase in the blending function towards RANS modelling. The blending function is thus considered the primary cause of resolved turbulence attenuation.

### 7.3 Aeroacoustic Results

The CFD acoustic data was recorded over 0.2 s, or  $26.67T^*$ , where  $T^* = C/U$ , with a sampling frequency of 44,000 Hz. Welch's method was then applied to the raw pressure data using a Hann window. The

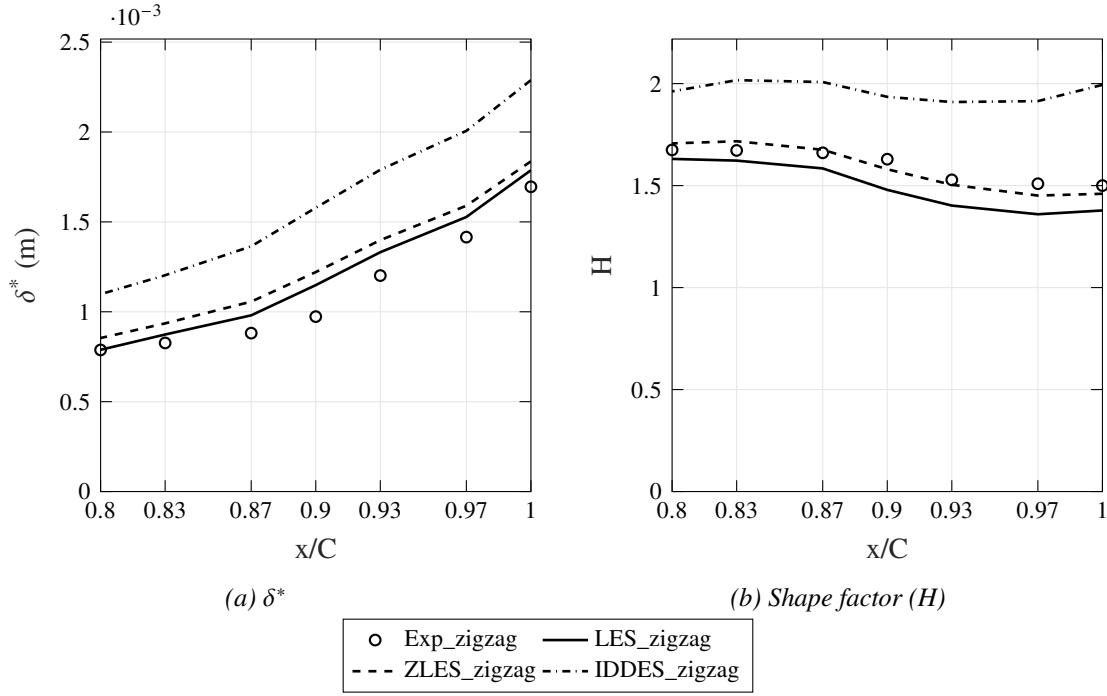


Figure 7.12: Displacement thickness and shape factor at different  $x/C$  locations

analysis was performed with a Fast Fourier Transform (FFT) length of 1000 and a 50% overlap between consecutive segments. The acoustic pressure power spectral density  $\text{PSD}_p(f)$  was first calculated in linear units of  $\text{Pa}^2/\text{Hz}$ . It was then expressed on a logarithmic decibel scale as  $\phi_{pp}(f)$  [dB/Hz], using a reference acoustic pressure of  $P_{\text{ref}} = 2 \times 10^{-5}$  Pa according to Equation 7.6:

$$\phi_{pp}(f) = 10 \log_{10} \left( \frac{\text{PSD}_p(f)}{P_{\text{ref}}^2} \right), \quad P_{\text{ref}} = 2 \times 10^{-5} \text{ Pa.} \quad (7.6)$$

### 7.3.1 Spanwise Discrepancy Acoustics Correction

To account for the spanwise discrepancy between the experimental data ( $2C$ ) and the CFD data ( $0.288C$ ), the Seo correction factor was applied to the CFD results. The spanwise coherence was determined at the streamwise location  $x/C = 0.9$ . The normalised coherence length ( $\sqrt{\pi}L_c/L_{\text{sim}}$ ) at each discrete frequency is shown in Figure 7.16. From Figure 7.16a, it can be seen that for LES\_zigzag, LES\_straight and ZLES\_zigzag, the normalised coherence length is less than 1, resulting in a constant shift in the spectra by  $10 \log(L_{\text{exp}}/L_{\text{sim}})$ , or 8.42 dB/Hz. The IDDES\_zigzag case (Figure 7.16b), however, does not correspond to this constant shift (due to eddy-viscosity induced lack of 3-dimensionality). For the IDDES\_zigzag case, the coherence length,  $\sqrt{\pi}L_c(f)$ , never exceeds  $L_{\text{exp}}$ , therefore, for the frequencies with  $\sqrt{\pi}L_c(f)/L_{\text{sim}} \leq 1$ , the correction in Equation 7.4a was used, and for all other frequencies, the correction in Equation 7.4b was used. All results shown further have been corrected to account for the spanwise discrepancy.

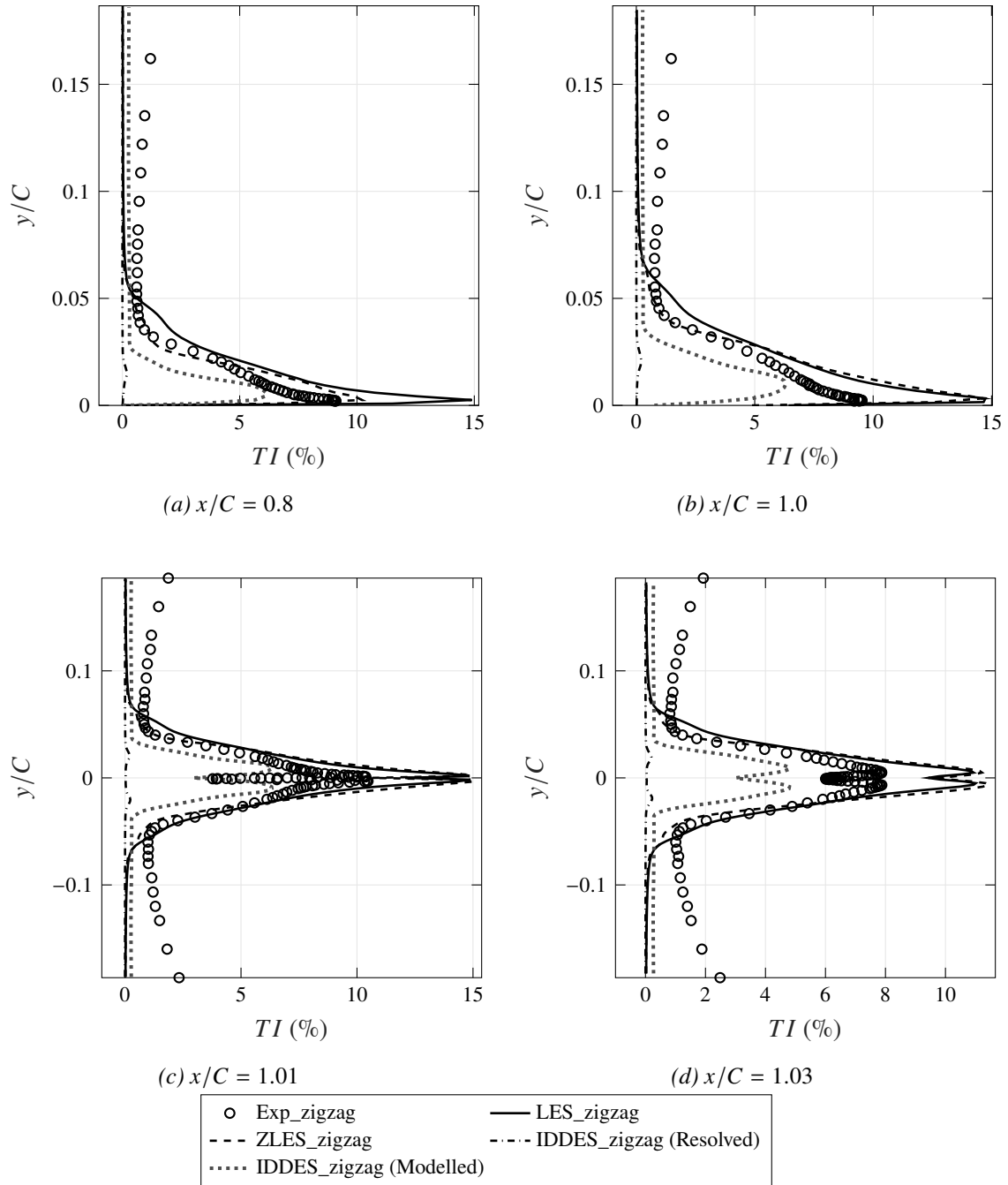


Figure 7.13: Turbulence intensity showing the effect of turbulence modelling.

### 7.3.2 Effect of Turbulence Modelling and Trip Shape

Figure 7.17b shows a comparison between the different trip shapes and turbulence models using the Curle analogy. Good agreement is observed between the experimental data and the LES\_straight, LES\_zigzag and ZLES\_zigzag results using the Curle analogy. The ZLES\_zigzag starts to deviate from the experimental data at around 5000 Hz whilst both the LES cases agree well up to 10 kHz. The IDDES\_zigzag exhibits a significantly lower  $\phi_{pp}$  compared to the other cases, as well as a significantly lower Overall Sound Pressure Level (OASPL), as shown in Figure 7.17d, with a 25.71 dB lower OASPL compared to the experimental data at the 120° polar angle. The maximum difference in OASPL for the LES\_zigzag and

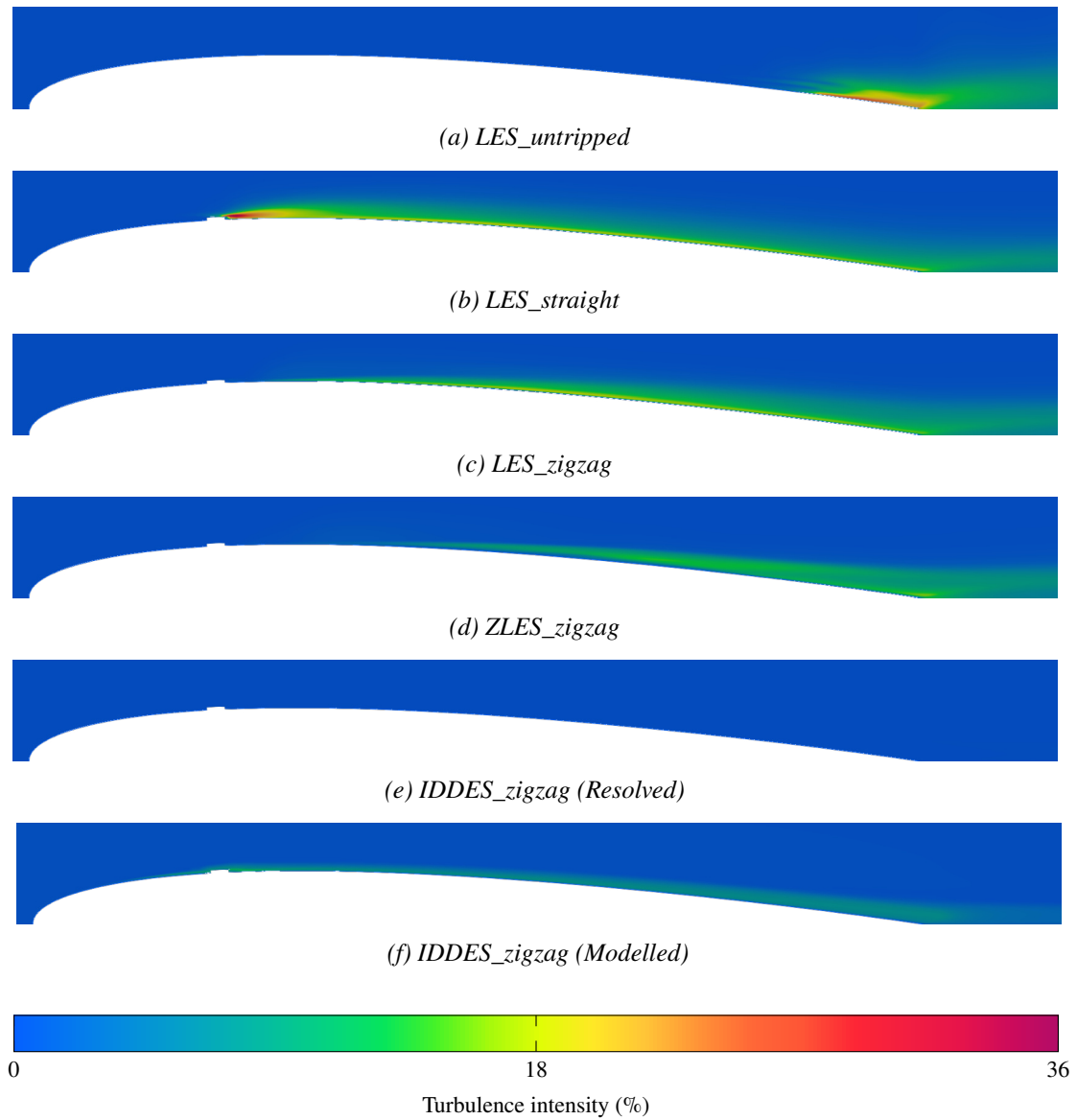


Figure 7.14: Comparison of turbulence intensity for different trip shapes and turbulence models (half airfoil shown).

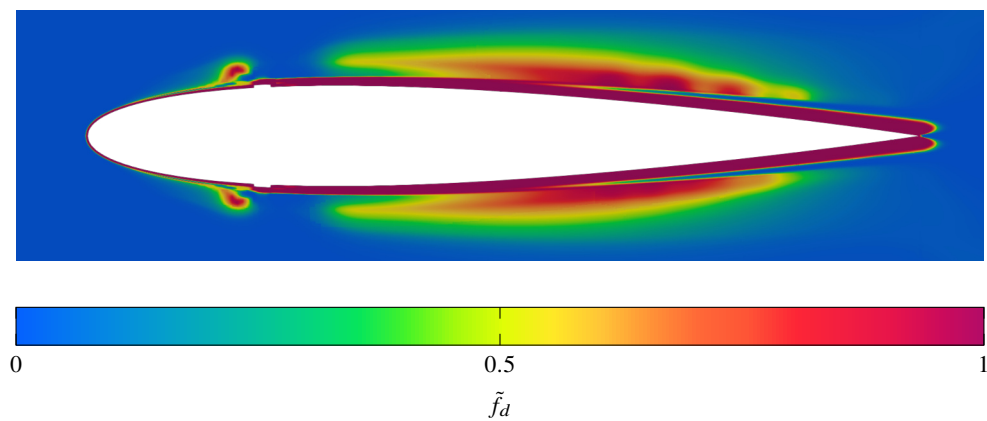


Figure 7.15: Contours of the empirical blending function,  $\tilde{f}_d$  (see Equation 3.14).



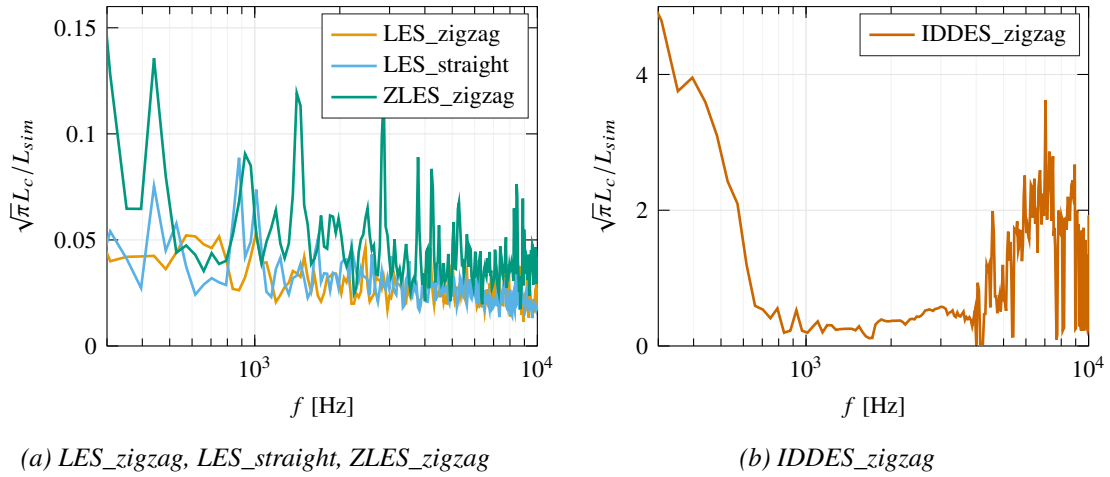


Figure 7.16: Normalised coherence length  $\sqrt{\pi}L_c/L_{sim}$  as a function of frequency  $f$  [Hz].

LES\_straight cases compared to the experimental data are a 0.518 dB and 0.427 dB decrease respectively at the  $120^\circ$  polar angle, whilst for the ZLES\_zigzag case, the maximum difference is a 0.85 dB increase at the  $60^\circ$  polar angle.

### 7.3.3 Effect of FW-H Surface and Closing Disc Location

The following section compares the effect of FW-H surface locations (with respect to distance from the airfoil turbulence), as well as location of closing discs (or absence of) downstream of the airfoil. Figure 7.4 shows the location of the FW-H surfaces and Figure 7.18 shows these with contours of  $dp/dt$ . The following naming convention is used for the CFD results: <turbulence model>\_<trip shape>\_<formulation>\_<y/C location of FW-H surface from airfoil>\_<open or closed surface>\_<x/C location of closure disc if closed>.

For Figures 7.19–7.23, the FW-H surfaces located at  $y/C = 0.067$  show a significantly higher  $\phi_{pp}$  and OASPL compared to the other surface locations and the experiment. Figure 7.18 shows that this surface lies within a region of strong hydrodynamic pressure fluctuations, hence the strong overprediction of noise.

Figure 7.19 shows results using FW-H with an open disc. The surface located at  $y/C = 0.13$  appears to provide the closest match to the experiments. However, there is a noticeable depression between approximately 500 to 2,000 Hz at the  $120^\circ$  polar angle shown in Figure 7.19a. This depression reduces progressively with downstream polar angle, for example at  $\theta = 50^\circ$  (Figure 7.19c). At this location, the depression nearly disappears. In order to test the mesh resolution effect, a reduced axial grid stretching ratio of 1% was tested but results were nearly identical.

For a closing disc located at  $x/C = 2.67$ , Figure 7.20b shows good agreement between the surfaces located at  $y/C = 0.13$ , 0.2, and 0.27. However, the  $\phi_{pp}$  for these three locations are significantly higher than the experimental data, especially below 1000 Hz. Figure 7.18 shows hydrodynamic pressure fluctuations passing through the closing disc located at  $x/C = 2.67$ , which explains the increase in low frequency noise. The OASPL plot in Figure 7.20d also shows a much higher OASPL for all surfaces closed at  $x/C = 2.67$  compared to the experiments.

Figure 7.21b shows results for a closure disc at  $x/C = 4.0$ . Similar to the surfaces with closure discs at

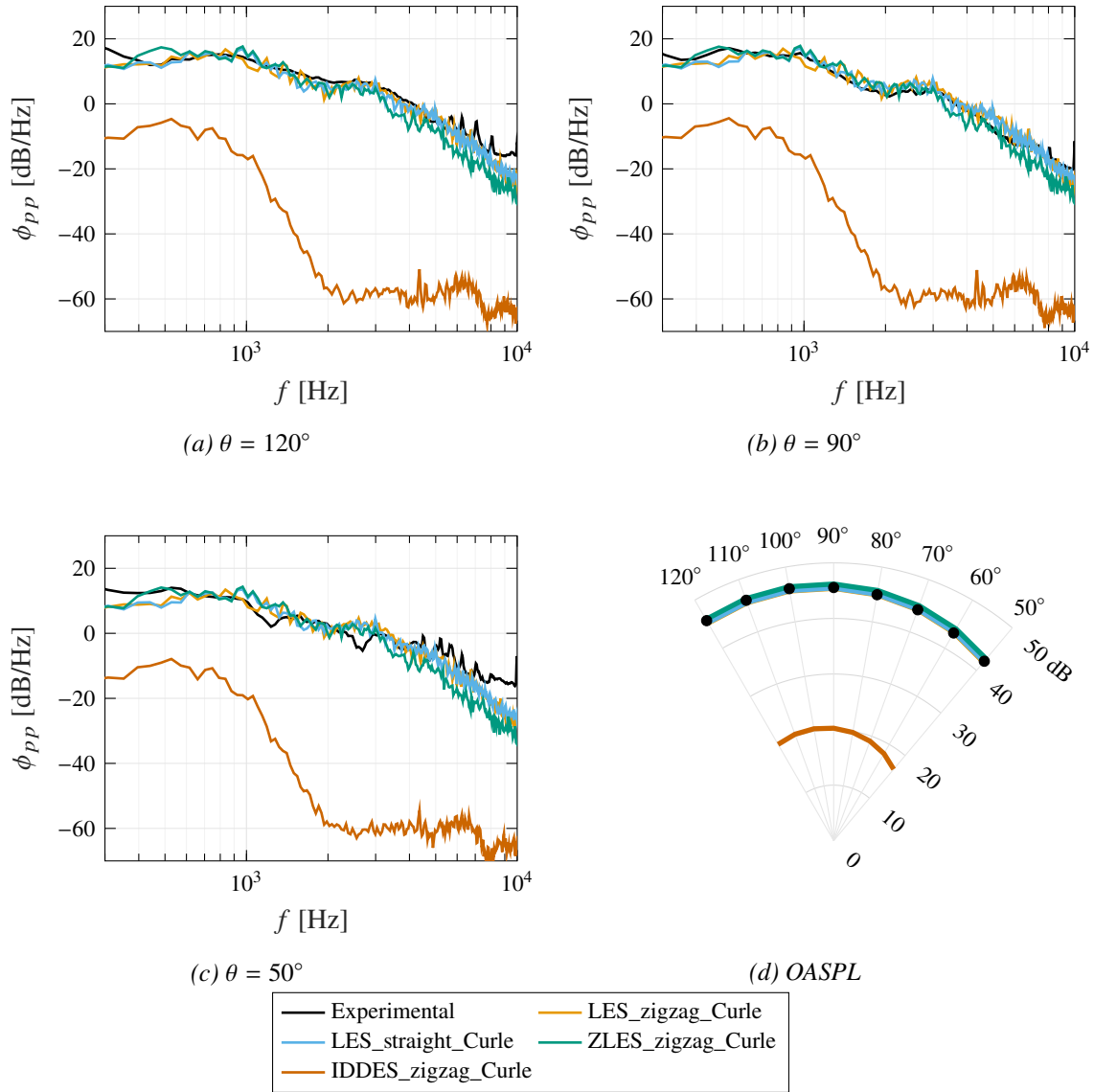


Figure 7.17: Acoustic PSD ( $\phi_{pp}$ ) [dB/Hz] as a function of frequency  $f$  [Hz] comparing different trip shapes and turbulence models.

$x/C = 2.67$ , the surfaces located at  $y/C = 0.13$ ,  $0.2$ , and  $0.27$ , also show good agreement with each other. In this case, however, they also show very good agreement with the experimental data above 500 Hz, as shown in Figure 7.21b. This deviation below 500 Hz becomes more pronounced at the higher polar angles, with an increase of 5.17 dB in OASPL compared to the experiment at  $\theta = 50^\circ$  for the surface located at  $y/C = 0.27$ , shown in Figure 7.21d.

For the surfaces with a closing disc at  $x/C = 5.33$ , the surface located at  $y/C = 0.27$  shows good agreement with the experimental data, as shown in Figure 7.22b. The maximum deviation in the OASPL for this surface occurs at  $\theta = 120^\circ$ , a 1.06 dB larger OASPL compared to the experiment.

Figure 7.23 shows the surfaces with a closing disc at  $x/C = 6.67$ . From Figure 7.23b, it appears that the surface located at  $y/C = 0.27$  compares best with the experimental data. However, Figure 7.23d shows that all surfaces behave inconsistently at various polar angles.

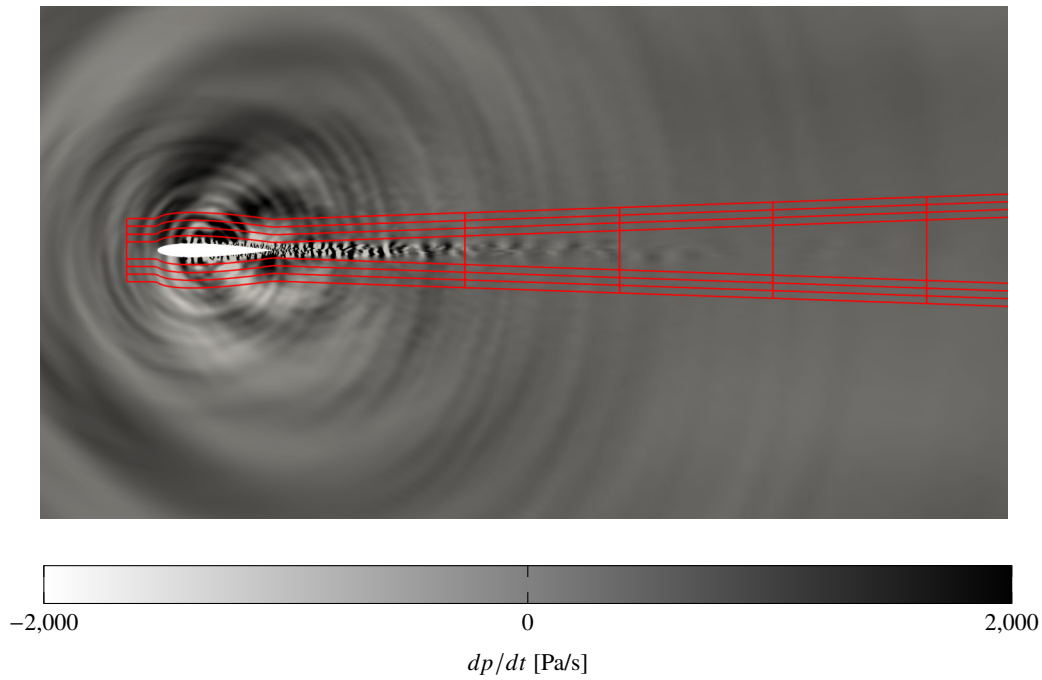


Figure 7.18: FW-H surface locations shown on a contour of  $dp/dt$ .

### 7.3.4 A priori prediction of FW-H surface position

Following examination of FW-H surface shape and location, we attempt to provide guidance for future simulations. Prior estimates for jet flow by Wang et al. [158] were made using wall distance, turbulence intensity and vorticity. For low Reynolds number flow over airfoils, noise is dominated by trailing edge scattering caused by pressure fluctuations at the surface. Figure 7.24 shows contours of  $p_{RMS}$  around the airfoil surface. By examining  $dp_{RMS}/dx$  and  $dp_{RMS}/dy$  from the trailing edge, and based on the results shown in Figure 7.21 and Figure 7.22, it can be conjectured that the surfaces should be placed at a distance where  $dp_{RMS}/dx$  and  $dp_{RMS}/dy$  drop to 0.1% of their respective values close to the wall. For this case, that corresponds to a surface placed at  $y/C = 0.29$  above and below the airfoil surface, with a closing disc at  $x/C = 4.6$ . These approximately correspond to the best FW-H position in the current study.

### 7.3.5 Comparison of Acoustics Models and Prediction

Figure 7.25 shows a comparison between the experimental data, the semi-empirical Brooks predictions, the Curle analogy, and the 1A and GT formulations of the FW-H equations with the FW-H surfaces being placed at the best locations found in Section 7.3.4. Excellent agreement is shown between the experimental, Curle, 1A and GT formulations across the frequency spectrum, as shown in Figure 7.25b. Figure 7.25d also shows good agreement between the CFD data and experimental data, with the largest difference in OASPL being for the 1A formulation at the  $120^\circ$  polar angle, where a 1.057 dB increase in OASPL is observed over the experiment. The Brooks prediction shows better agreement with the experiment at the downstream polar angles, with only a 0.439 dB larger OASPL at the  $50^\circ$  polar angle and a 2.57 dB larger OASPL compared to the experiments at  $120^\circ$ . As noted by Lee et al. [32], the Brooks prediction only provides 1/3 octave band predictions, and in the case of a tripped boundary layer, the 1/3 octave band does not capture the oscillations in the spectrum caused by interference between coherent sources along the airfoil chord.

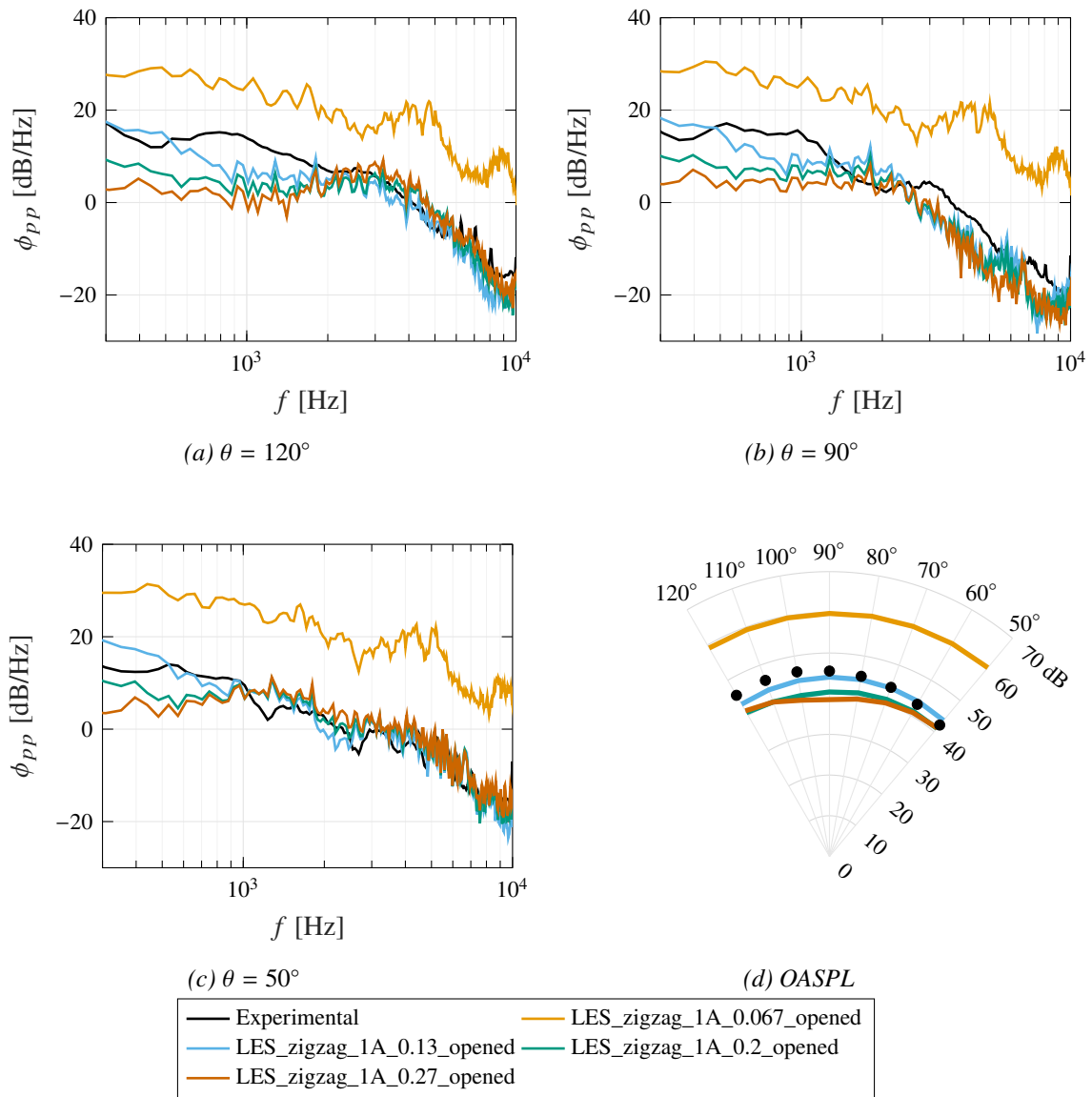


Figure 7.19: Acoustic PSD ( $\phi_{pp}$ ) [dB/Hz] as a function of frequency  $f$  [Hz] comparing FW-H surface locations with no closing discs.

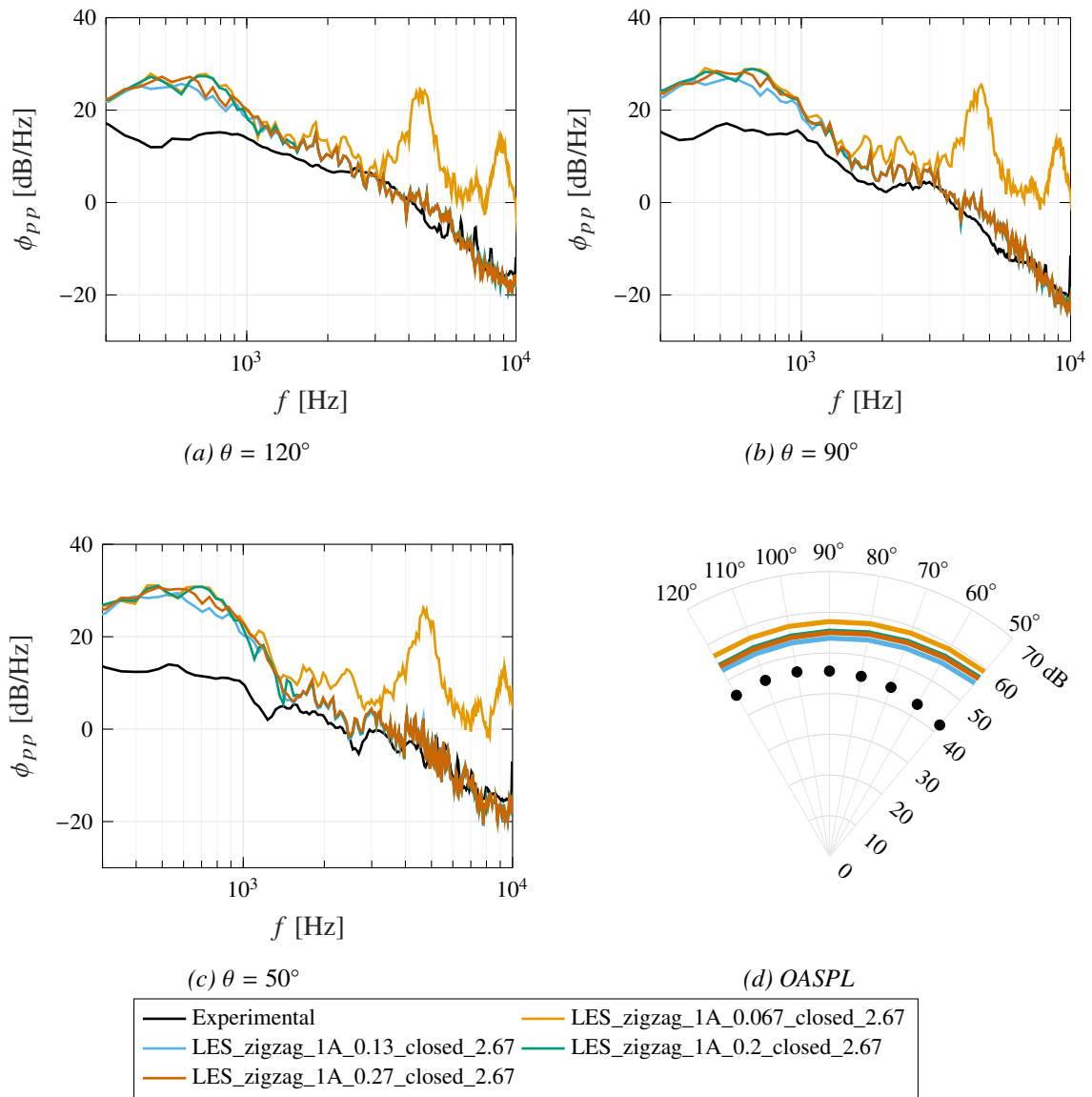


Figure 7.20: Acoustic PSD ( $\phi_{pp}$ ) [dB/Hz] as a function of frequency  $f$  [Hz] comparing FW-H surface locations with a closing disc at  $x/C = 2.67$ .

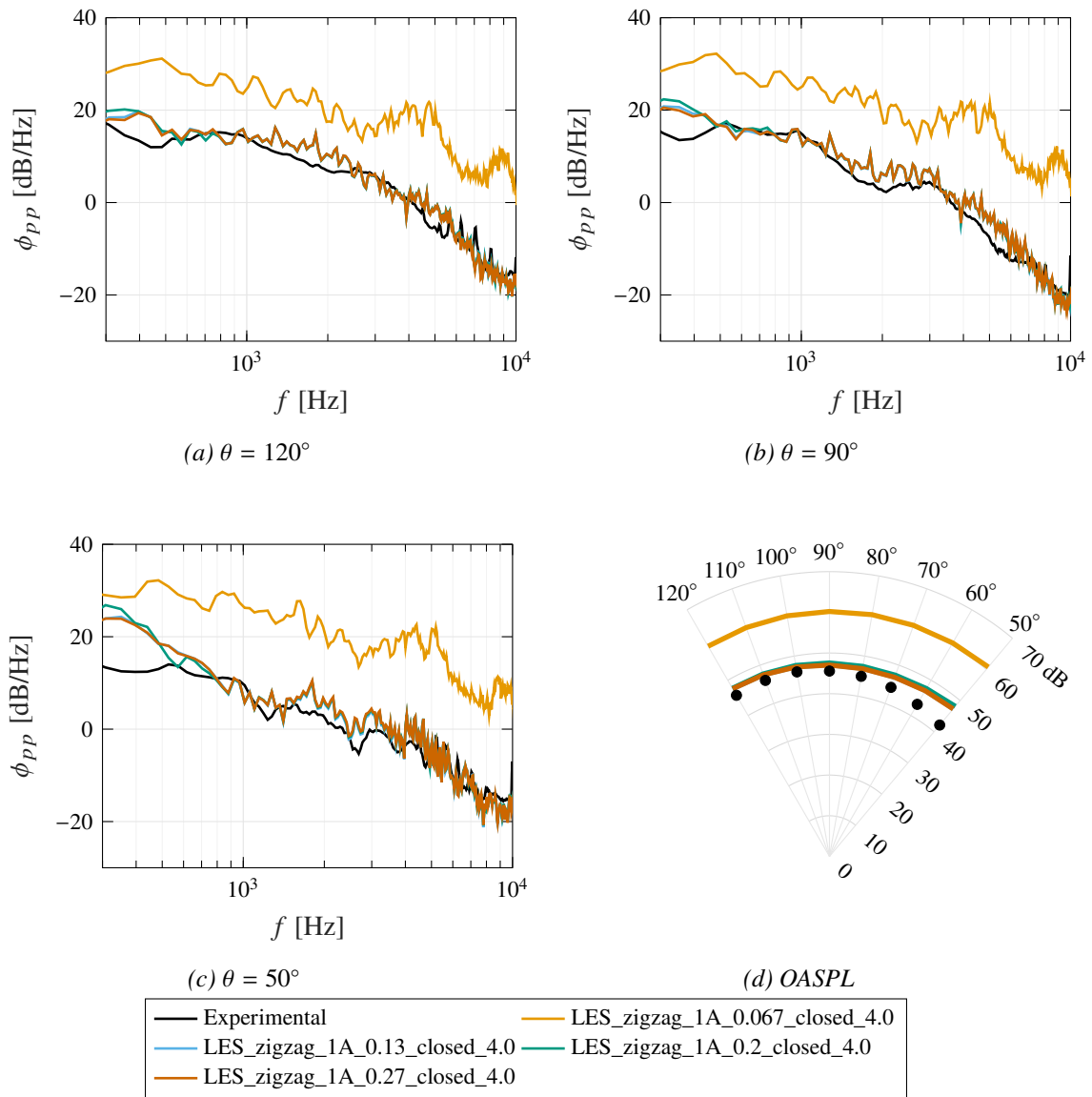


Figure 7.21: Acoustic PSD ( $\phi_{pp}$ ) [dB/Hz] as a function of frequency  $f$  [Hz] comparing FW-H surface locations with a closing disc at  $x/C = 4.0$ .

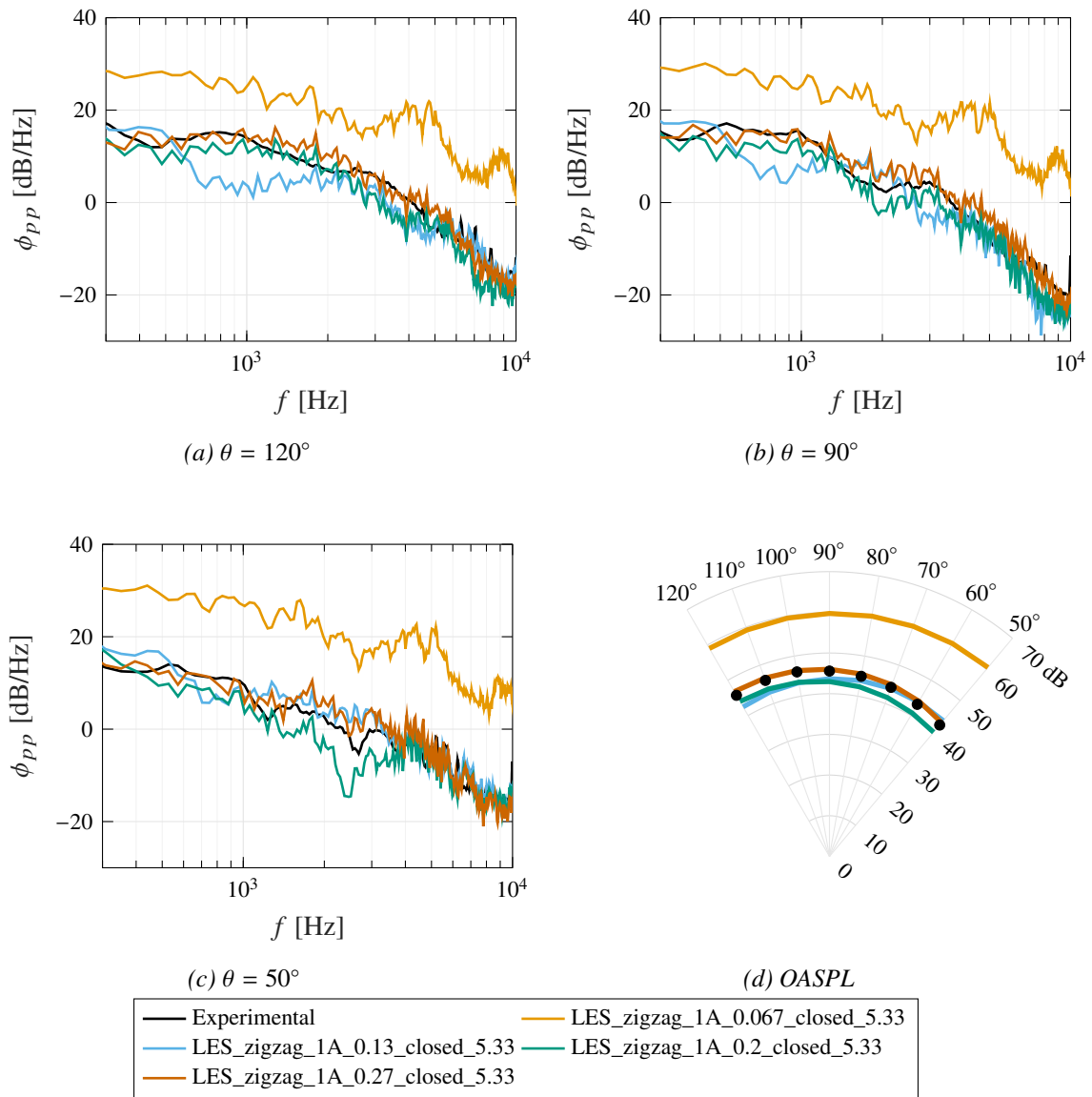


Figure 7.22: Acoustic PSD ( $\phi_{pp}$ ) [dB/Hz] as a function of frequency  $f$  [Hz] comparing FW-H surface locations with a closing disc at  $x/C = 5.33$ .

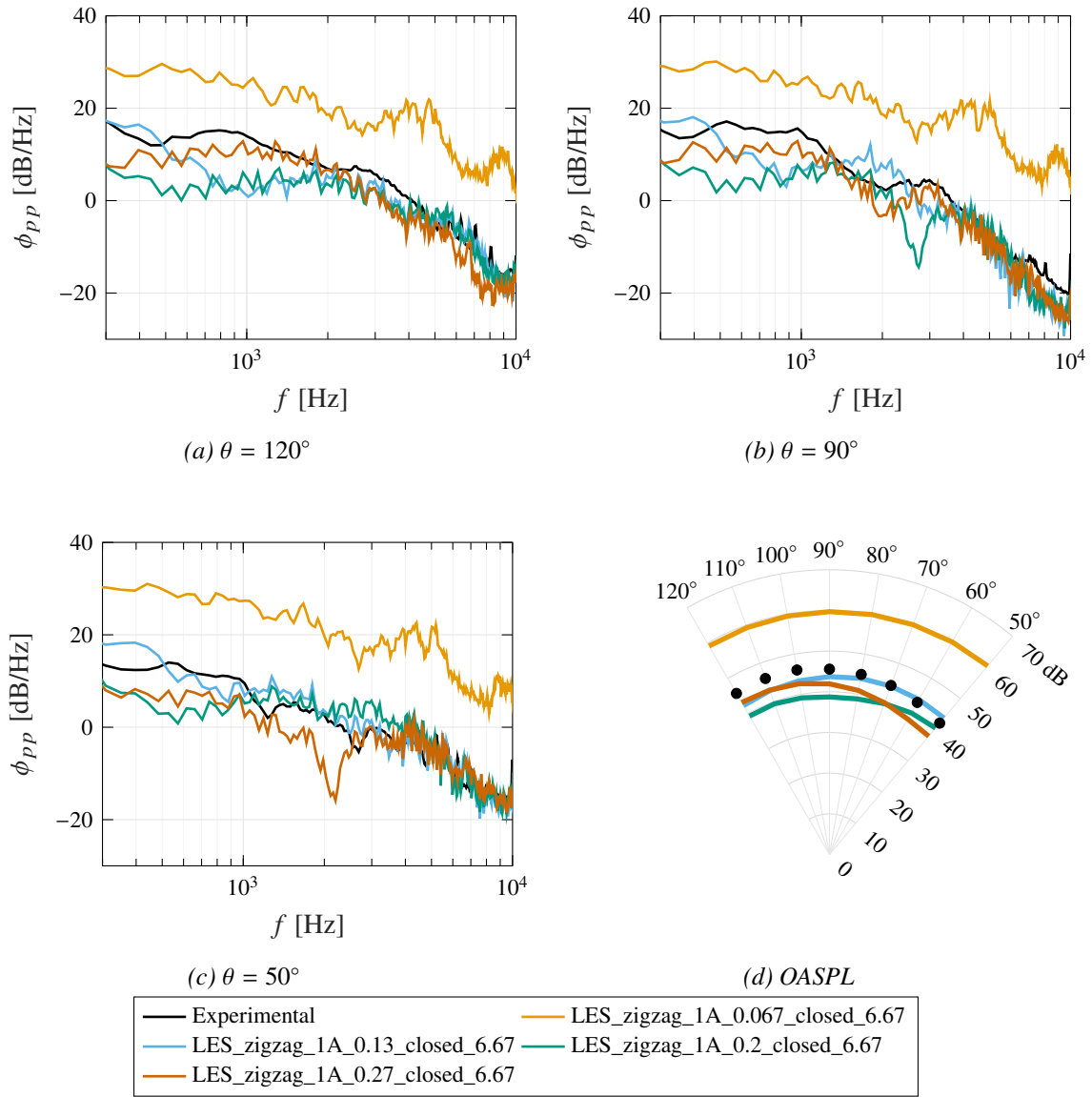
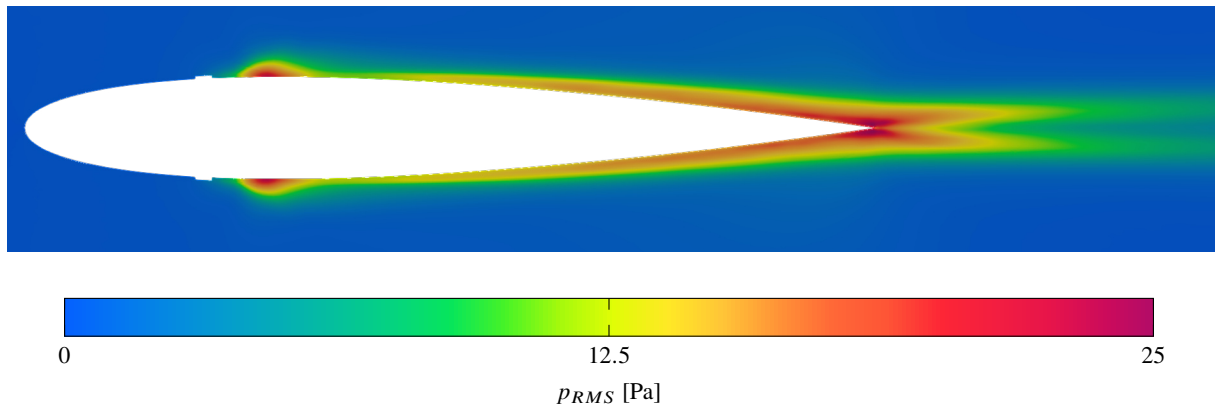
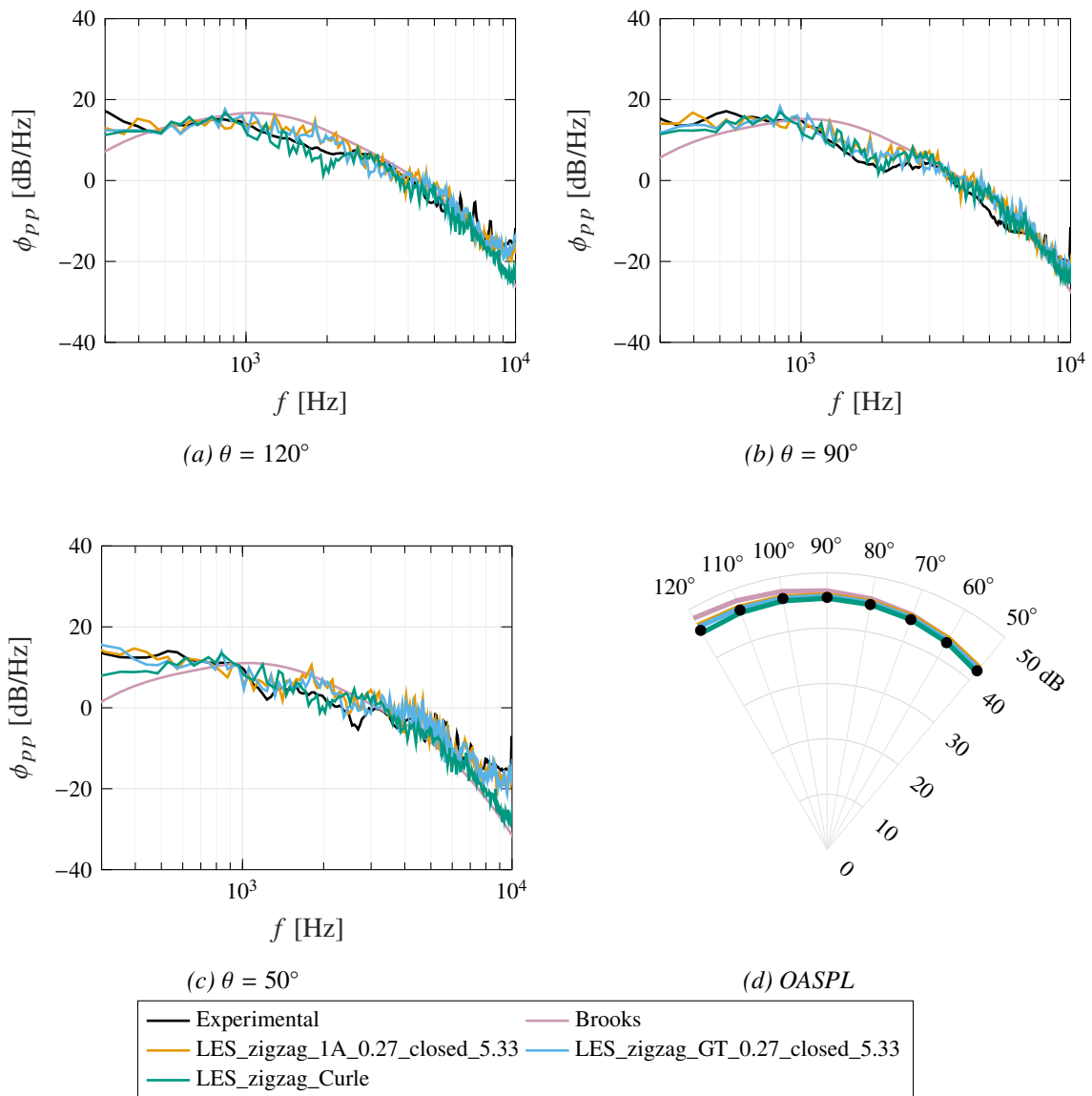


Figure 7.23: Acoustic PSD ( $\phi_{pp}$ ) [dB/Hz] as a function of frequency  $f$  [Hz] comparing FW-H surface locations with a closing disc at  $x/C = 6.67$ .



Figure 7.24: Contour of  $p_{RMS}$ .Figure 7.25: Acoustic PSD ( $\phi_{pp}$ ) [dB/Hz] as a function of frequency  $f$  [Hz] comparing experimental, numerical, and analytical data.

## 7.4 Nozzle Effects on NACA 0012 Flow and Acoustics

To assess effects of the nozzle, the geometry was represented in CFD.

### 7.4.1 Methodology

#### 7.4.1.1 Computational Domain

Figure 7.26 shows a part of the computational domain, focussing on the nozzle and airfoil. The entire computational domain consisted of approximately 111M hexahedral elements, extended  $0.8C$  upstream of the airfoil,  $50C$  above, below and downstream of the airfoil and had a span of  $0.288C$ .

The nozzle exit was modelled to replicate the nozzle at the Brunel University of London anechoic chamber [152]. The nozzle exit had a height of  $0.67C$  and the thickness of the lip was  $0.0157C$ . The actual nozzle was modelled as two parallel flat plates with a length of  $6.67C$ .

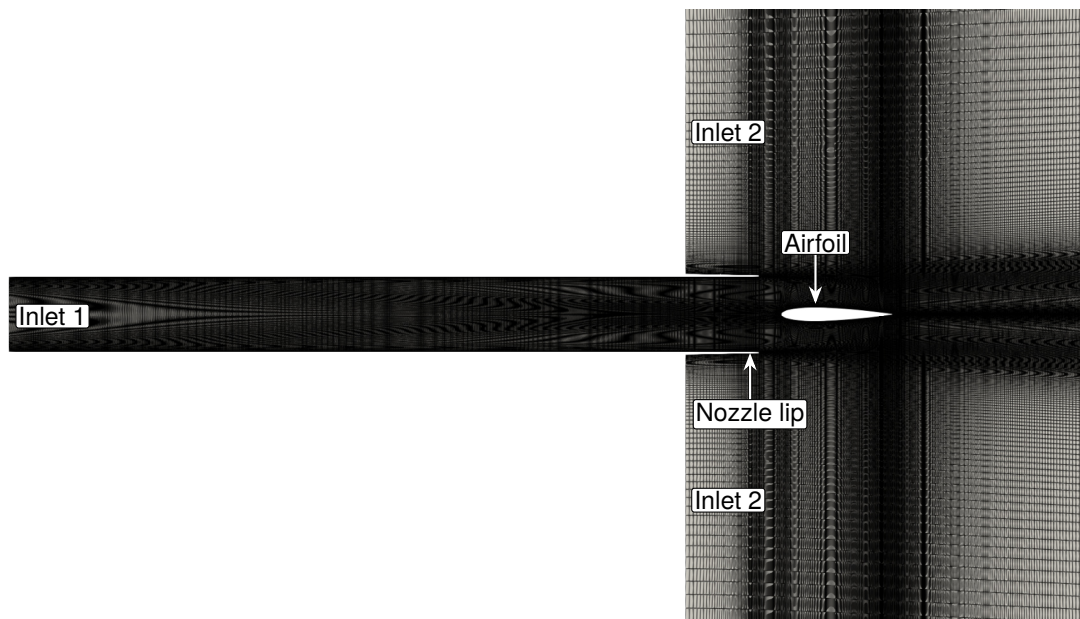


Figure 7.26: Part of the computational domain showing the nozzle and airfoil.

#### 7.4.1.2 Boundary Conditions

At the Inlet 1, the velocity was set to  $U_\infty = 20$  m/s, corresponding to a chord-based Reynolds number of  $Re = 200,000$  and a Mach number of  $M = 0.058$ . At Inlet 2, the velocity was set to  $0.01U_\infty$  to stabilise the solver and ensure the generation of the shear layer. A zero-gradient condition was set for pressure at both inlets. At the outlet, a zero-gradient condition was applied for velocity, while the pressure was set to  $p = 101,306.12$  Pa. A no-slip condition was applied at the walls, and spanwise periodicity was used.

#### 7.4.1.3 Numerical Methods

The simulations were conducted in OpenFOAM v2212 using the rhoPimpleFoam solver. The turbulence model used was the WALE LES model. Temporal discretisation was performed using the second-order backward difference scheme. For spatial discretisation, the Linear Upwind Stabilized Transport (LUST) scheme was employed. The cases were run for  $58.67T^*$ , where  $T^* = C/U_\infty$ , with temporal statistics and acoustics being recorded from  $32T^*$ .

### 7.4.2 Results

Figure 7.27a shows the normalised streamwise mean velocity profile at the nozzle exit. From Figure 7.27a, it can be seen that the velocity profile at the nozzle exit is relatively flat, achieving a value of  $\bar{U}/U_\infty \approx 0.98$  at  $y/C = 0$ . The nozzle also exhibits a low turbulence intensity at the exit (shown in Figure 7.27b), achieving a value of  $\approx 0.17\%$  at  $y/C = 0$ .

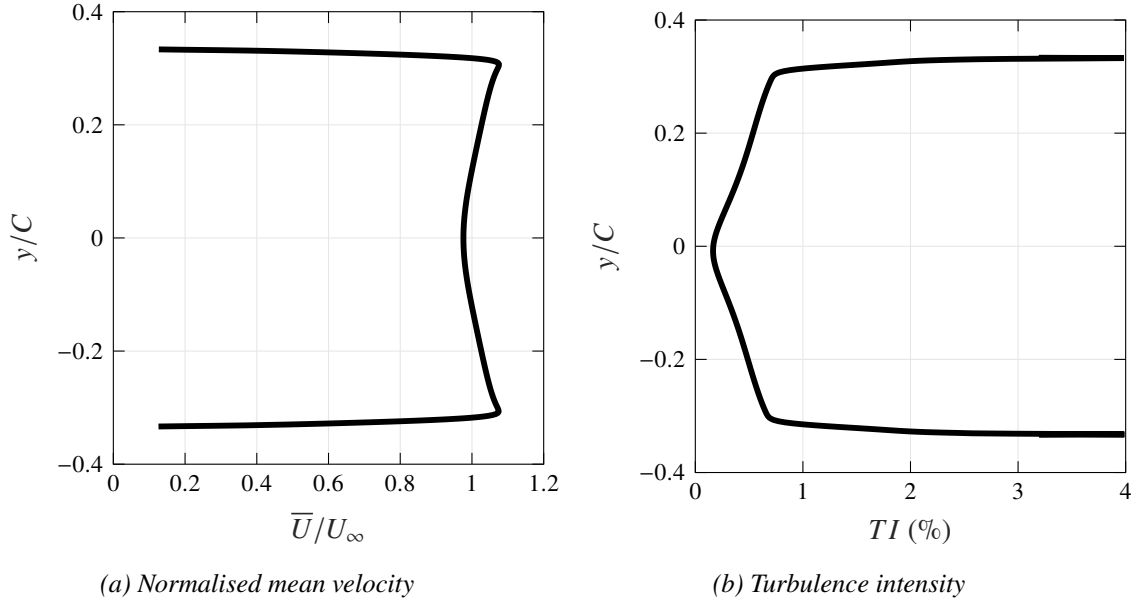


Figure 7.27: Normalised streamwise mean velocity and turbulence intensity at nozzle exit

Figure 7.28 shows contours of spanwise instantaneous vorticity. From the figure, large-scale vortices being shed from the nozzle lip are observed. Further downstream, in the airfoil wake, these large-scale structures appear to interact with the smaller scale structures shed from the turbulent boundary layer of the airfoil.

#### 7.4.2.1 Mean Velocity and Turbulence Intensity

Figure 7.29 shows the contours of normalised streamwise mean velocity. The shear layer emanating from the nozzle lip is clearly visible, separating the high-speed flow from the nozzle exit with the surrounding fluid.

Figure 7.30a shows excellent agreement between both LES\_zigzag cases (with and without the nozzle) and the experimental data at  $x/C = 0.8$ . At  $x/C = 1.0$ , Figure 7.30b shows that the two LES\_zigzag cases agree closely with each other, but exhibit a slightly thicker boundary layer compared to the experimental data.

In the wake, at  $x/C = 1.01$ , Figure 7.30c also shows good agreement between the LES\_zigzag cases, although the simulations predict a slightly wider velocity deficit and a small overprediction of the minimum velocity compared to the experimental data.

A slightly wider deficit is also observed at  $x/C = 1.03$  (Figure 7.30d), where the LES\_zigzag case without the nozzle captures the minimum velocity well, while the case with the nozzle shows a slight overprediction.

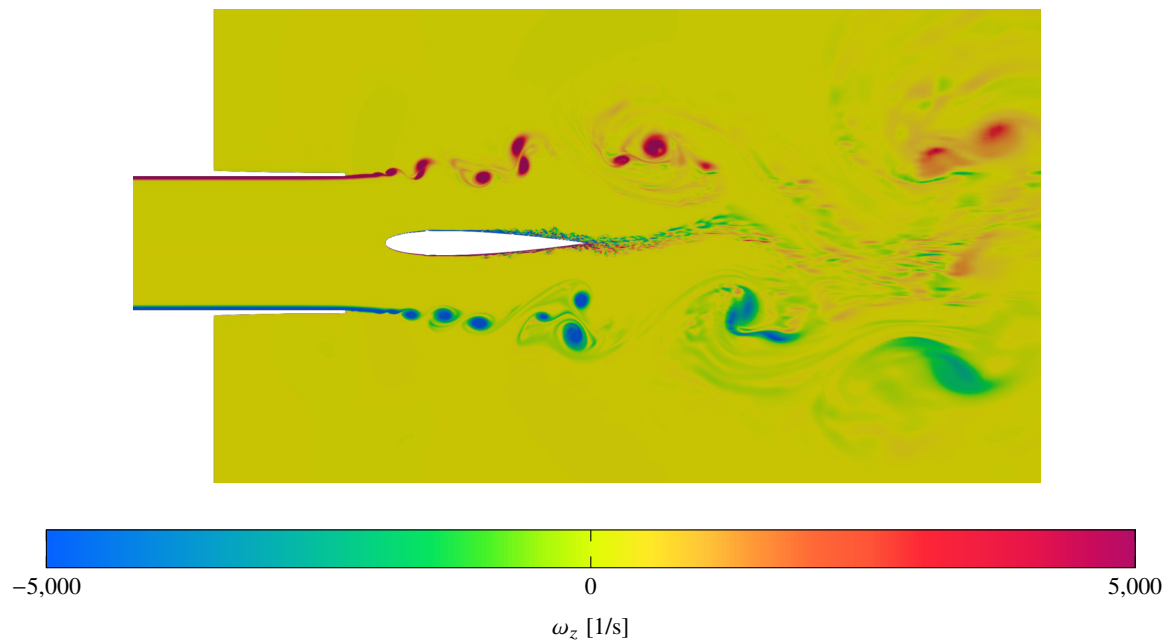


Figure 7.28: Contours of instantaneous spanwise vorticity showing the formation of vortices near the nozzle lip.

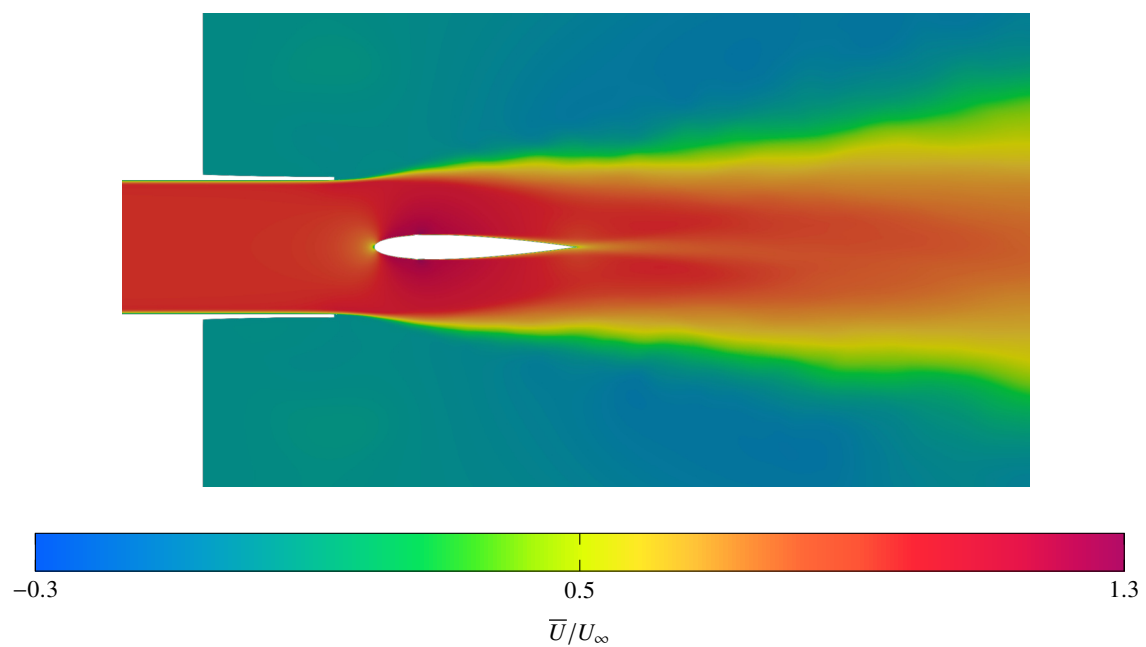


Figure 7.29: Contours of normalised streamwise mean velocity showing the nozzle shear layer.

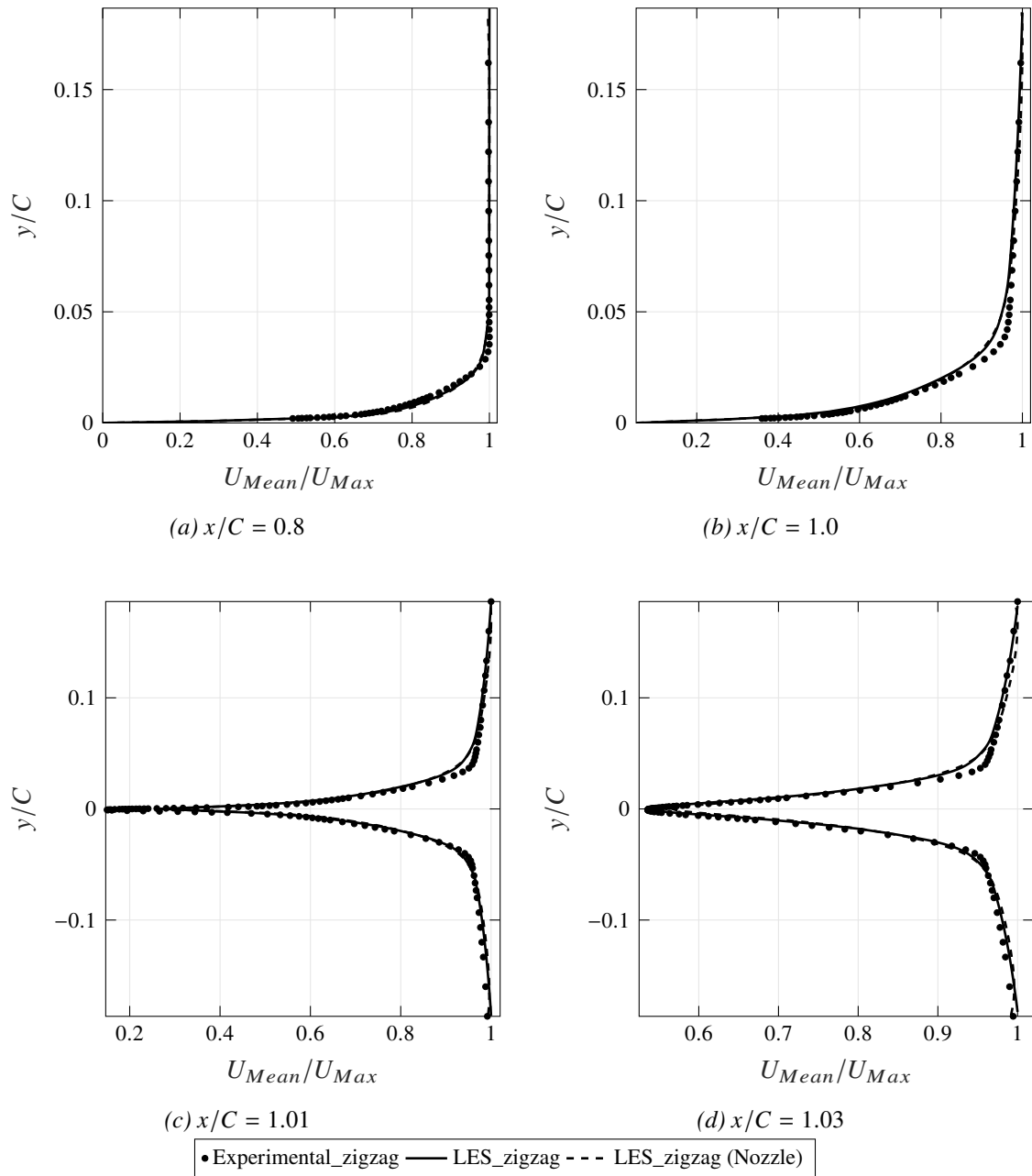


Figure 7.30: Normalised streamwise mean velocity showing the effect of the nozzle.

Figure 7.31a shows close agreement in displacement thickness, both in terms of magnitude and trend, between the two LES\_zigzag cases and the experimental data. The LES\_zigzag case with the nozzle shows slightly better overall agreement with the experimental measurements.

The shape factor (Figure 7.31b) shows excellent agreement between the two LES\_zigzag cases, and both follow the trend of the experimental data, although at slightly lower values.

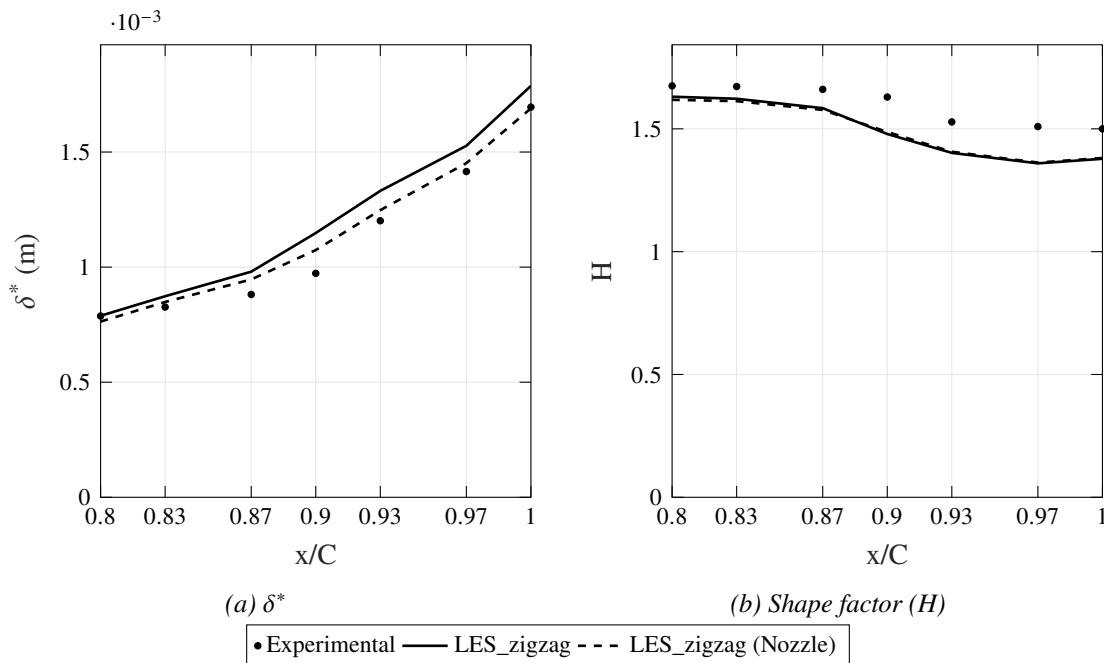


Figure 7.31: Displacement thickness and shape factor at different  $x/C$  locations

Figure 7.32 shows contours of turbulence intensity. The contours indicate that turbulence generated by the shear layers emanating from the nozzle lip diffuses into the airfoil boundary layer.

This effect is evident by examining the turbulence intensity profiles (shown in Figures 7.33a, 7.33b, 7.33c and 7.33d). At all four of these locations, the entire turbulence intensity profile for the LES\_zigzag with the nozzle appears to be offset from the experimental data. Evaluating this difference at  $y/C = 0.05$ , it was found that the average offset in turbulence intensity over the tested locations was 8.52% for the boundary layer locations and 9.64% for the wake locations.

Figure 7.34 shows the normalised Q-criterion iso-surfaces ( $Q_{norm} = 10$ ), coloured by mean velocity magnitude, where  $Q_{norm} = Q \times C^2 / U_\infty^2$ . Quasi-2-D coherent structures shed from the nozzle can be seen. These structures do not break down into 3-D turbulence, but instead persist downstream and appear to affect the flow around the airfoil. Therefore, whilst a reasonable attempt had been made to reconcile the experimental and CFD data, the limited span of the CFD domain appears to have increased the turbulence intensity at both the boundary-layer and wake locations.

#### 7.4.2.2 Acoustic Power Spectral Density

Figure 7.35 shows the acoustic PSD ( $\phi_{pp}$ ) at the  $120^\circ$  (Figure 7.35a),  $90^\circ$  (Figure 7.35b), and  $50^\circ$  (Figure 7.35c) polar angles, along with the OASPL (Figure 7.35d). At all three polar angles, a strong low-frequency deviation from the experimental data is observed for the LES\_zigzag nozzle case below 900 Hz, where

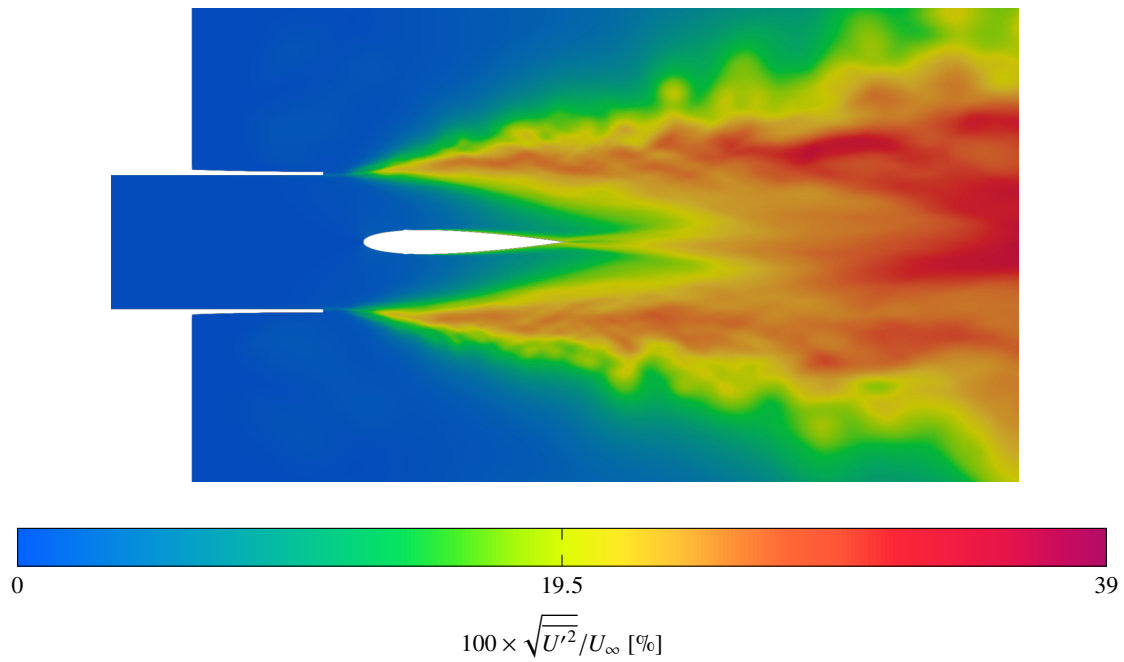


Figure 7.32: Contours of turbulence intensity showing shear-layer interaction with the airfoil boundary layer.

a significant increase in  $\phi_{pp}$  occurs. An additional increase in the high-frequency contribution is also present for the LES\_zigzag nozzle case, although the frequency at which this deviation occurs varies with polar angle. The divergence from the experimental data occurs at approximately 4,000 Hz for the 120° and 90° polar angles, and at approximately 6,000 Hz for the 50° polar angle. Consequently, the OASPL for the LES\_zigzag nozzle case is significantly higher than the experimental values, as shown in Figure 7.35d.

In the mid-frequency range (900–4,000 Hz), the LES\_zigzag nozzle case shows good agreement with both the LES\_zigzag case without the nozzle and the experimental data.

An attempt was made to filter the high and low frequency noise of the LES\_zigzag nozzle case by applying a second-order Butterworth filter with a high-pass cutoff of 700 Hz and a low-pass cutoff of 6,000 Hz. As shown in Figure 7.35, applying this filter results in much better agreement between the LES\_zigzag nozzle case, the experimental data, and the LES\_zigzag case without the nozzle at all polar angles, leading to a closer match in OASPL as well. However, it should be noted that the high- and low-pass cutoff values were selected through trial and error. At the time of running the simulation, the upSuite Library was not yet implemented, and therefore unsteady data was not recorded.

It is proposed that probing the velocity spectral density at key flow regions could help identify the appropriate frequency ranges for applying such filters.

### 7.4.3 Summary

In this chapter, the flow and acoustic characteristics of a tripped NACA 0012 airfoil at  $Re = 200,000$ ,  $M = 0.058$  and 0° angle of attack were investigated using OpenFOAM's rhoPimpleFoam solver. The study assessed the influence of trip configuration, turbulence modelling approach, acoustic analogy, FW-H surface placement and nozzle effects, with the numerical results compared against experimental

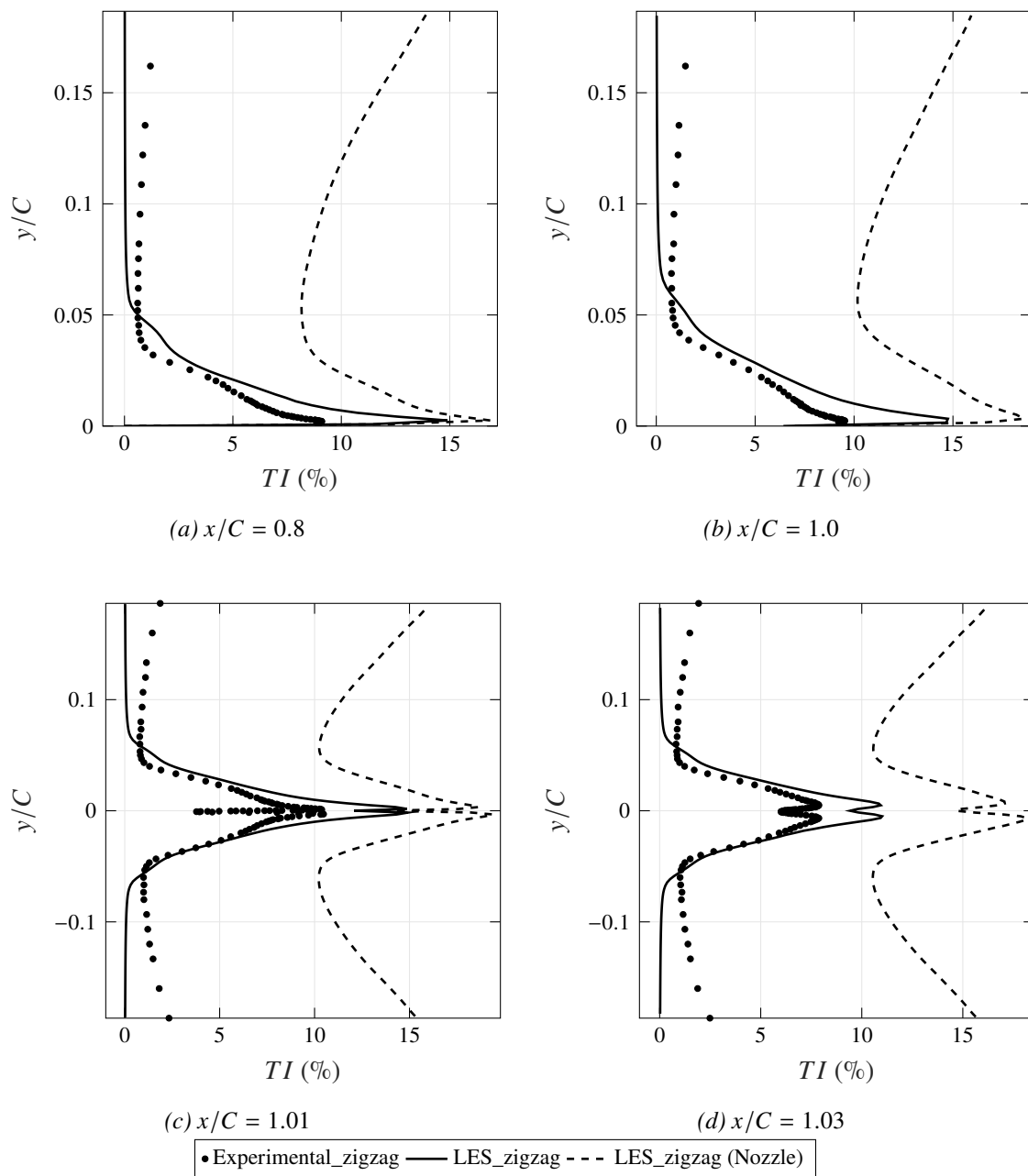


Figure 7.33: Turbulence intensity showing the effect of the nozzle.



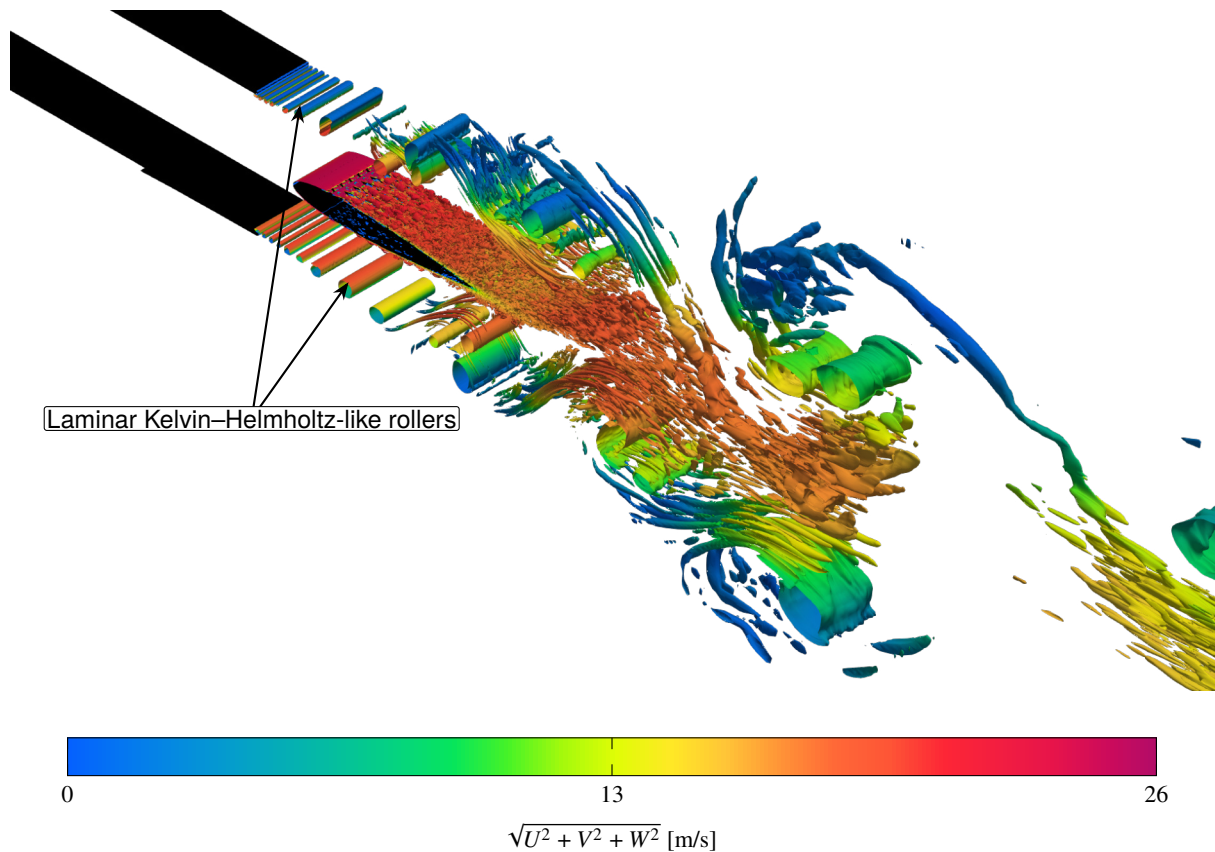


Figure 7.34:  $Q$ -criterion iso-surfaces coloured by mean velocity magnitude,  $Q_{norm} = 10$ .

measurements from the Brunel University of London anechoic chamber.

The comparison between the untripped, straight-trip and zigzag-trip LES cases showed that tripping was necessary to reproduce the experimental boundary-layer development, whilst the trip shape was less important. The tripped cases showed good agreement with the measured mean velocity profiles, displacement thickness and shape factor, particularly towards the trailing edge. In contrast, the untripped case showed delayed transition and produced a different turbulence distribution near the trailing edge and in the wake. Although the straight and zigzag trips produced different local behaviour immediately downstream of the trip, these differences reduced further downstream, and both cases produced similar trailing-edge flow conditions.

The turbulence modelling comparison showed that LES and ZLES provided similar mean velocity and turbulence intensity predictions, with both approaches agreeing reasonably well with the experimental data. The ZLES case achieved this using approximately half the number of cells in the spanwise direction, suggesting that it can reduce computational cost while maintaining acceptable accuracy. The IDDES case, however, predicted a thicker boundary layer and showed very limited resolved turbulence in the near-wall region. This led to poor acoustic predictions, with the OASPL underpredicted by up to 25.71 dB at the  $120^\circ$  polar angle.

The acoustic predictions from the LES and ZLES cases showed good agreement with the experimental spectra and OASPL directivity when the Curle analogy was used. The LES\_zigzag, LES\_straight and

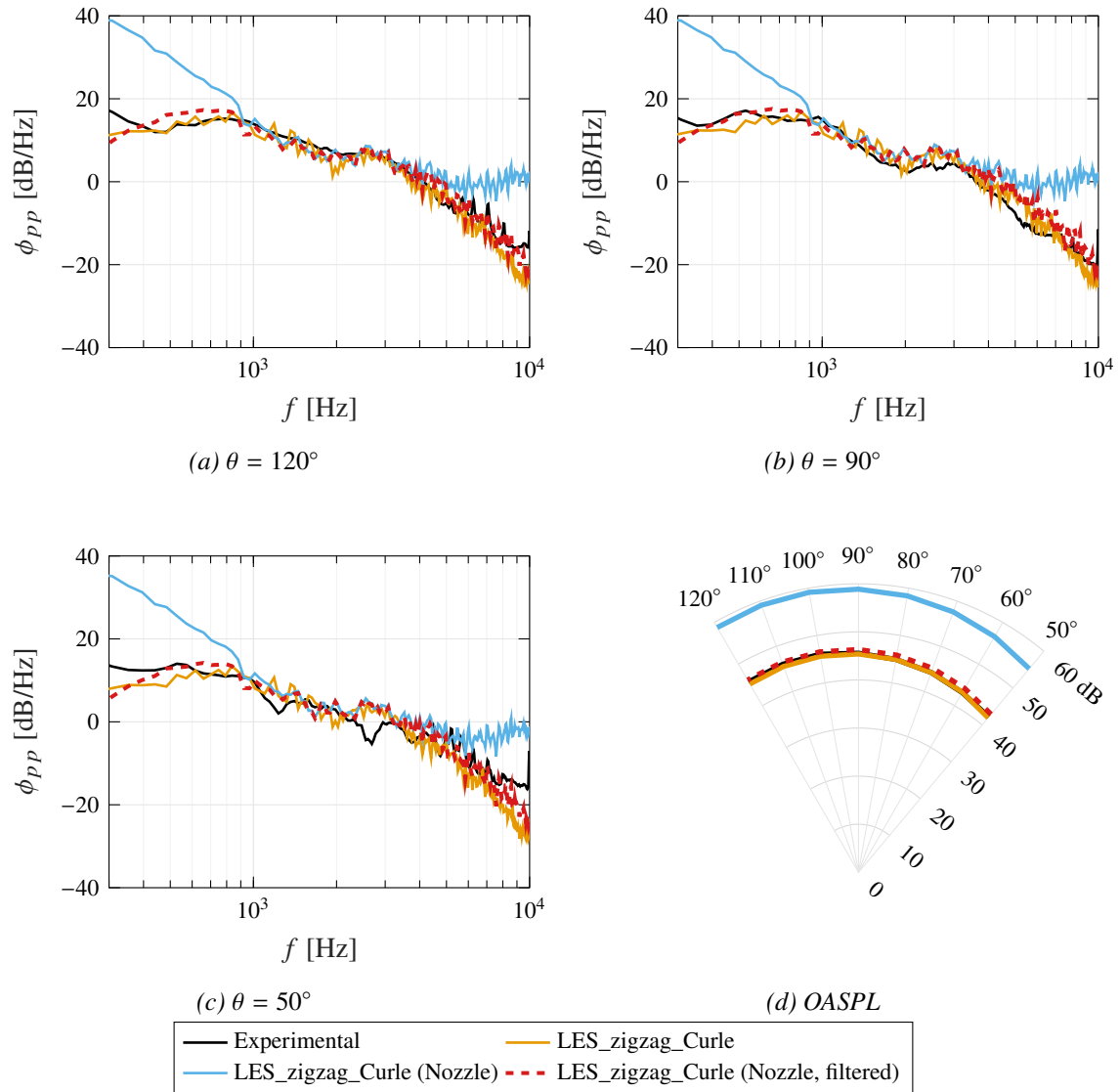


Figure 7.35: Acoustic PSD ( $\phi_{pp}$ ) [dB/Hz] as a function of frequency  $f$  [Hz] comparing the tripped airfoil with and without the nozzle.

ZLES\_zigzag cases produced maximum OASPL differences of 0.518 dB below, 0.427 dB below and 0.85 dB above the experimental values respectively. This demonstrates that both LES and ZLES were able to capture the main acoustic behaviour of the tripped airfoil, whereas the lack of resolved turbulence made IDDES unsuitable for this case.

The FW-H surface study showed that the placement of both the permeable surface and closure disc had a strong influence on the predicted spectra. Surfaces placed too close to the airfoil were affected by hydrodynamic pressure fluctuations and therefore overpredicted the noise, while closure discs placed too close to the trailing edge introduced excessive low-frequency content. The most consistent agreement was obtained using a surface at approximately  $y/C = 0.27$  with a downstream closure disc at  $x/C = 5.33$ . Based on this, an a priori placement guideline was proposed using the decay of  $p_{RMS}$  gradients away from the trailing edge.

The comparison of acoustic analogies showed excellent agreement between the Curle, FW-H 1A and GT formulations when suitable FW-H surfaces were used. The largest OASPL difference among these CFD-based methods was approximately 1.057 dB above the experiment for the FW-H 1A formulation at the  $120^\circ$  polar angle. The Brooks semi-empirical method also showed reasonable agreement, particularly at downstream polar angles, although its one-third-octave-band formulation limited its ability to reproduce the detailed narrowband spectral behaviour.

Including the nozzle had little effect on the mean velocity predictions, which remained close to the experimental data and the baseline LES\_zigzag case. However, the nozzle shear layers introduced quasi-2-D turbulence into the airfoil boundary layer and wake, producing a clear offset in the turbulence intensity profiles. Acoustically, the nozzle case showed increased low-frequency noise below approximately 900 Hz and additional high-frequency deviations above approximately 4000–6000 Hz, depending on polar angle. Applying a second-order Butterworth filter between 700 Hz and 6000 Hz improved agreement with the experimental spectra and OASPL, although the cutoff frequencies were selected through trial and error.

The results demonstrate that LES and ZLES are capable of accurately predicting the flow and acoustic behaviour of the tripped NACA 0012 airfoil at low Mach number, provided that transition is properly represented and the acoustic surfaces are carefully positioned. The Curle analogy remains effective for this low-Mach-number case, while FW-H formulations can provide similarly accurate results when hydrodynamic contamination is avoided. The nozzle study aimed to reconcile experimental and CFD data, but the results were contaminated by the short span of the CFD domain.

# 8

## STRUCTURED POROUS CYLINDER COATING

Structured porous coated cylinders have gained increasing interest due to their potential application in fields such as landing gear struts, wind turbine structures, and heat exchangers. Their geometric similarity to airframe structures such as propeller mounting pylons suggests a potential application to UAMVs. Recent studies have considered both randomised and structured porous coatings, although the flow mechanisms associated with these configurations remain an active area of investigation, particularly within the porous region itself. To investigate this, this chapter presents the first known fully resolved Large-Eddy Simulation (LES) of a Structured Porous Coated Cylinder (SPCC), modelling a spanwise section at  $Re = 73,000$ , based on the outer diameter. The computational setup, wake development, boundary-layer behaviour on the inner cylinder, and shear-layer trajectory are examined.

### 8.1 Methodology

#### 8.1.1 Computational Domain

The computational domain (shown in Figure 8.1) contained approximately 208M hexahedral cells, and extended  $6.67D$  upstream of, above and below the cylinder centre and  $26.67D$  in the wake. The span of the domain was  $0.23D$ . The domain was constructed using a hybrid meshing technique, with prism layers and unstructured meshing being used within the SPCC and structured meshing being used for the far-field.

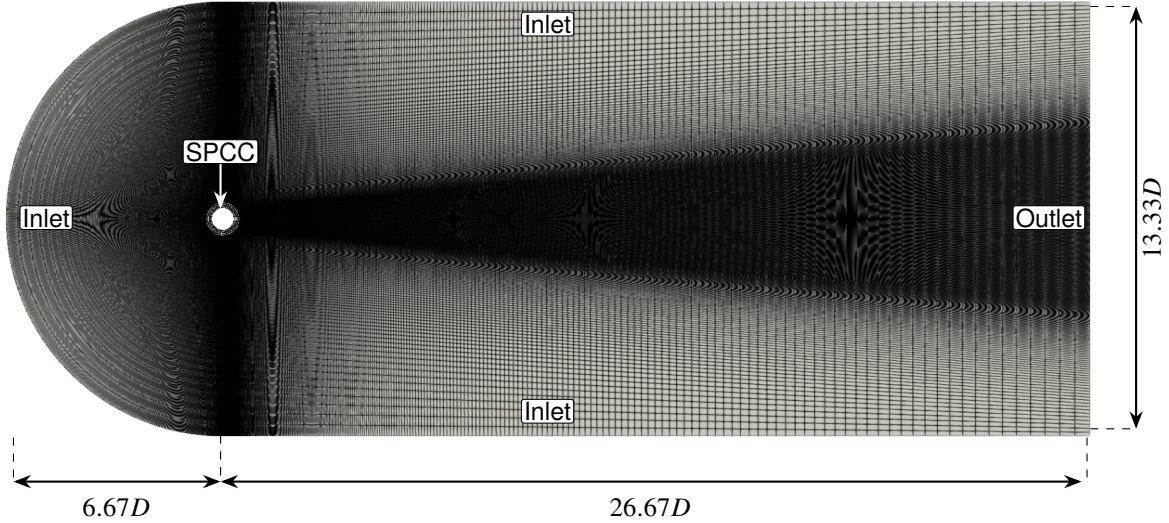


Figure 8.1: Computational domain

Figure 8.2 shows the front view of the SPCC (Figure 8.2a), with cut planes through the centre of a structure (A–A) and through the centre of the pores (B–B). Figures 8.2b and 8.2c show the computational domain around the SPCC at cut-planes A–A and B–B respectively.

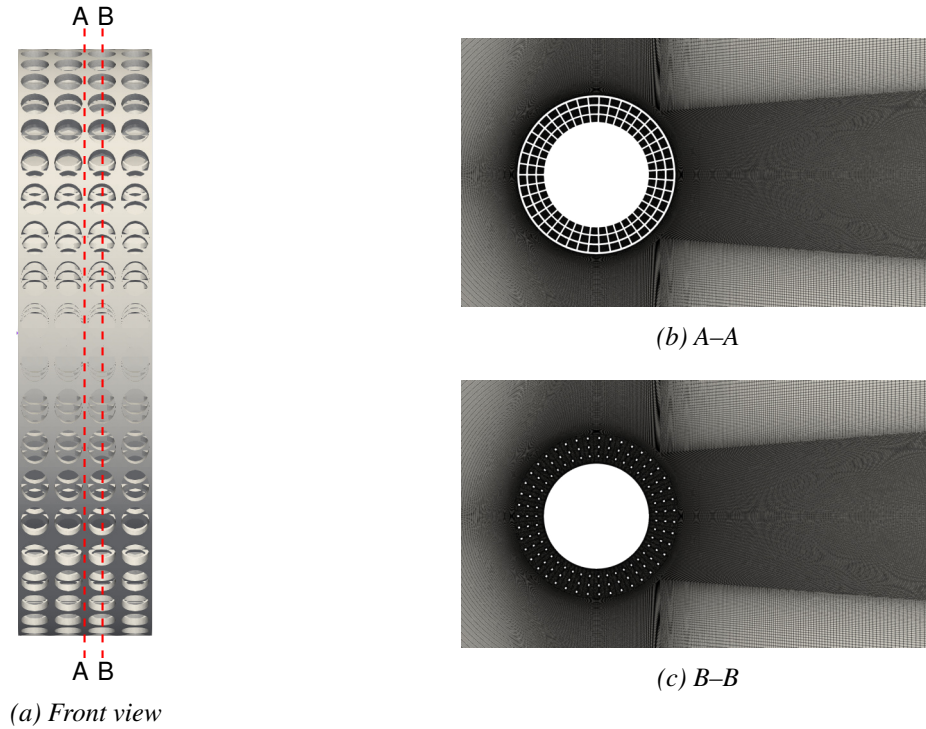


Figure 8.2: Computational domain through different cut planes

### 8.1.2 Boundary Conditions

At the inlet, the velocity was set to  $U_\infty = 11.3$  m/s, corresponding to an outer diameter based Reynolds number  $Re_D = 73,000$  and a Mach number of  $M = 0.033$ , while a zero-gradient condition was set for pressure. At the outlet, a zero-gradient condition was applied for velocity, while the pressure was set to  $p = 101,325$  Pa. A no-slip condition was applied at the walls, and spanwise periodicity was used.

### 8.1.3 Numerical Methods

The simulations were conducted in OpenFOAM v2212 using the rhoPimpleFoam solver. The turbulence model used was the WALE LES model. Temporal discretisation was performed using the second-order backward difference scheme. For spatial discretisation, the LUST scheme was employed. The cases were run for  $35.3T^*$ , where  $T^* = D/U_\infty$ , with temporal statistics being recorded from  $14.13T^*$ . The transient timestep was  $4.71 \times 10^{-5}T^*$ .

## 8.2 Results

### 8.2.1 Instantaneous and Mean Flow Fields

Figure 8.3 shows contours of the instantaneous (Figure 8.3a) and mean (Figure 8.3b) spanwise vorticity. From the figures, the shear layer trajectory can be seen. It can also be seen that there is an alternating positive and negative vorticity around the structural members of the SPCC. A similar trend was found by Arcondoulis et al. [121], and the alternating vorticity was attributed to flow separation from the internal structures. Due to this manner in which fluid flows around the internal structures of the SPCC, there is a noticeably high vorticity at the windward side. The separation from the outermost structures, and the flow exiting through the circumferential openings, appears to diffuse the shear layer, causing it to separate early



from the SPCC.

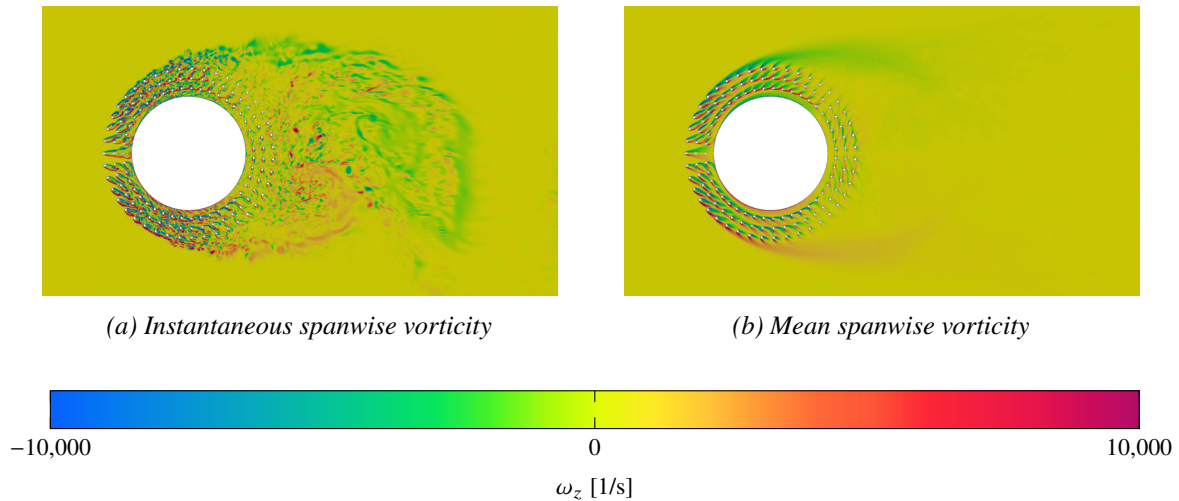


Figure 8.3: Contours of instantaneous and mean spanwise vorticity.

Figure 8.4 shows contours of the instantaneous (Figure 8.4a) and mean (Figure 8.4b) streamwise velocity. From Figure 8.4a, the flow exiting the outer circumferential pores diffusing the shear layer can be seen. There is also a region of reversed flow that appears to force fluid back upstream in the SPCC. In Figure 8.4b, there is evidence of boundary layer development on the inner cylinder. For the same geometry, water-tunnel tests conducted by Arcondoulis et al. [159] at  $Re = 7,000$ , boundary layer development was also observed on the inner cylinder, although the boundary layer appears to separate earlier in the current LES.

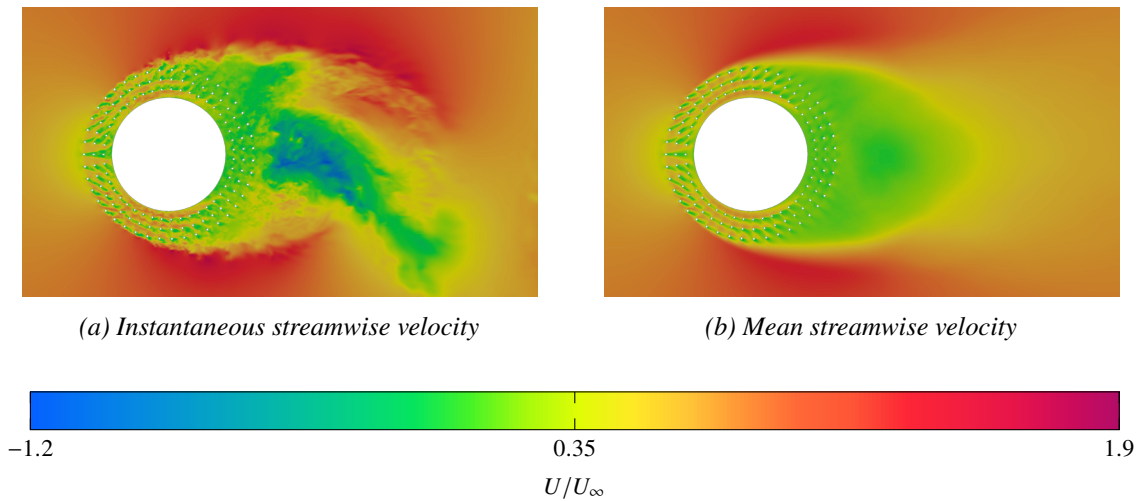


Figure 8.4: Contours of instantaneous and mean streamwise velocity.

Figure 8.5 illustrates how several initially separate shear layers merge, forming sub-channels of fluid with moderate velocity. Near the upstream stagnation zone and within the downstream separated regions associated with the porous structures, the flow inside the porous layer reverts to distinct wake-like formations. As in typical flow around a cylinder, the fluid accelerates over the outer surface. A significant portion of the flow also exits through the circumferential pores before reaching the leeward side of the

SPCC.

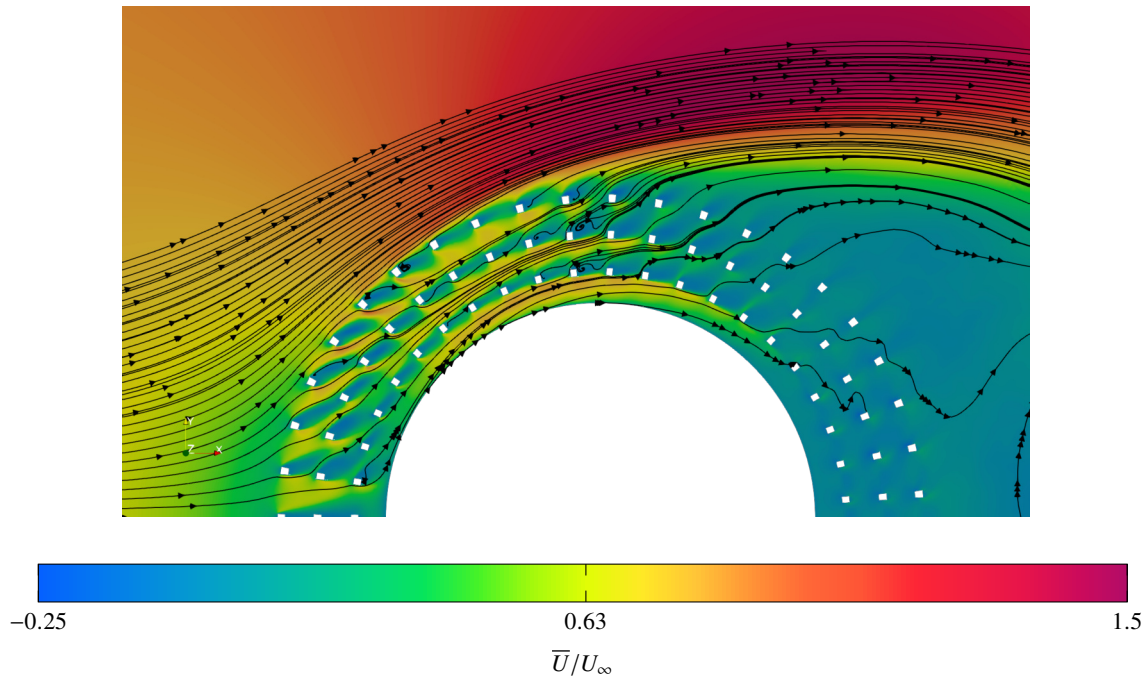


Figure 8.5: Contours of mean streamwise velocity showing streamlines.

### 8.2.2 Mean Velocity and Turbulence Intensity Wake Profiles

Figure 8.6 shows a comparison in normalised streamwise mean velocity between the resolved LES SPCC geometry and the experimental data at various wake locations. At  $x/D = 0.75$ , Figure 8.6a shows good agreement between the LES and experimental data, with the LES showing a slightly deeper wake deficit. However, this is expected since the hotwire measurements are unable to detect negative velocity, whilst the CFD can.

Further downstream, at  $x/D = 1.5$  and  $x/D = 3.0$ , Figures 8.6b and 8.6c both show the LES underpredicting the wake deficit compared to the experiments, with both showing narrower and shallower peaks.

At  $x/D = 7.0$ , Figure 8.6d shows relatively good agreement between the LES and experimental data, both with respect to the peak and the width of the deficit. There is a slight asymmetry to the LES data, possibly due to the short temporal statistics.

Figure 8.7 shows a comparison in turbulence intensity between the resolved LES SPCC geometry and the experimental data at various wake locations.

At  $x/D = 0.75$ , Figure 8.7a shows that the LES significantly overpredicts the turbulence intensity by over 30%. There are also four distinct peaks related to the vortex shedding from the outer and inner diameters.

At  $x/D = 1.5$ , Figure 8.7b shows the LES overpredicting the turbulence intensity, although to a lesser degree than at  $x/D = 0.75$ .

At  $x/D = 3.0$ , Figure 8.7c shows that the LES slightly overpredicts both the width and the peak of the turbulence intensity compared to the experimental data.

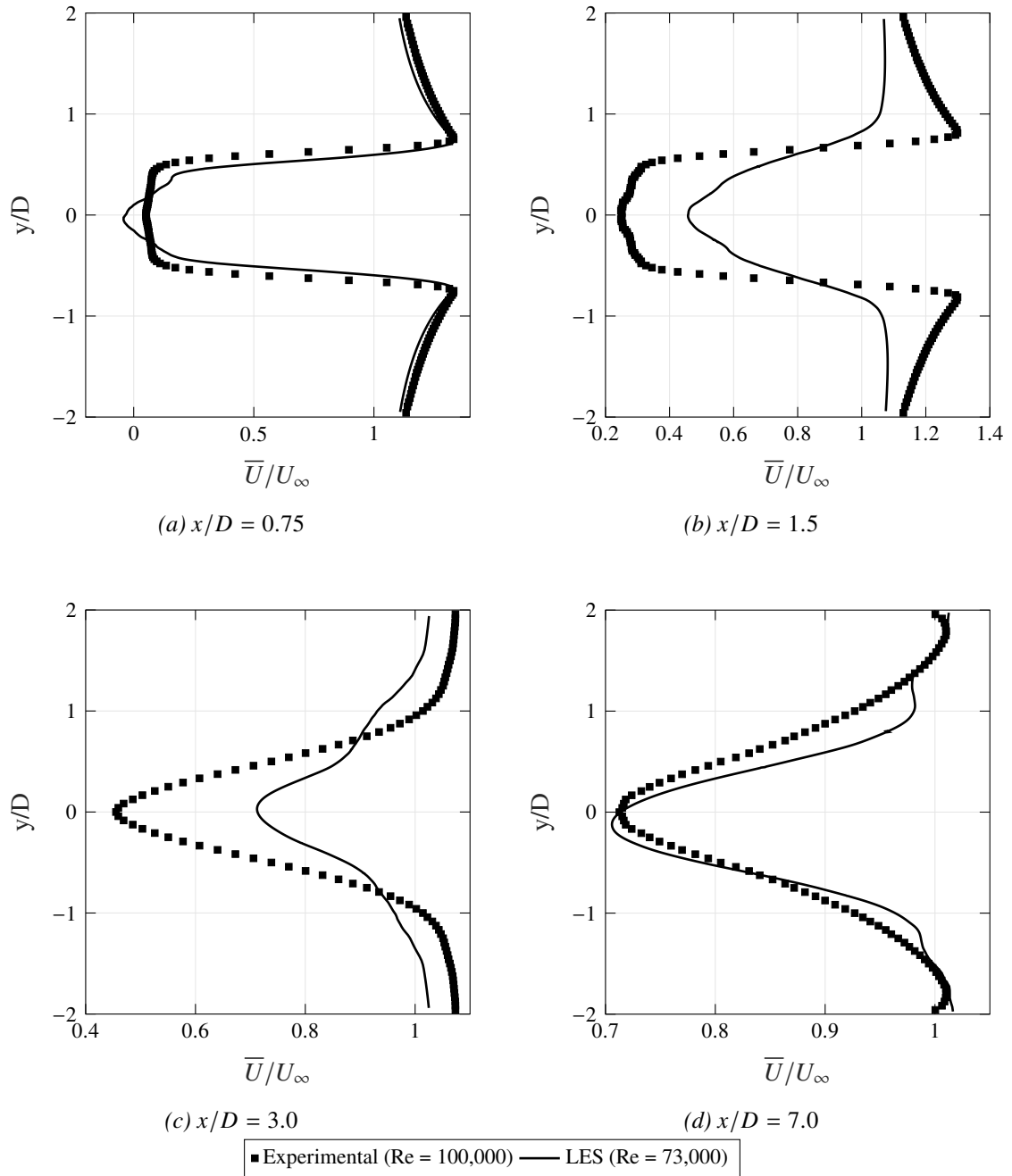


Figure 8.6: Normalised streamwise mean velocity profiles in the wake comparing the experimental and CFD SPCCs.



At  $x/D = 7.0$ , Figure 8.7d shows a considerable overprediction of turbulence intensity by the LES, and evidence of asymmetry, similar to that noticed in Figure 8.6d.

At  $x/D = 3.0$ , the LES results begin to exhibit a slight asymmetry in the turbulence intensity profile, which becomes more pronounced further downstream at  $x/D = 7.0$ . This behaviour is likely associated with the shorter temporal sampling period, as the downstream regions may not have fully reached a statistically steady state.

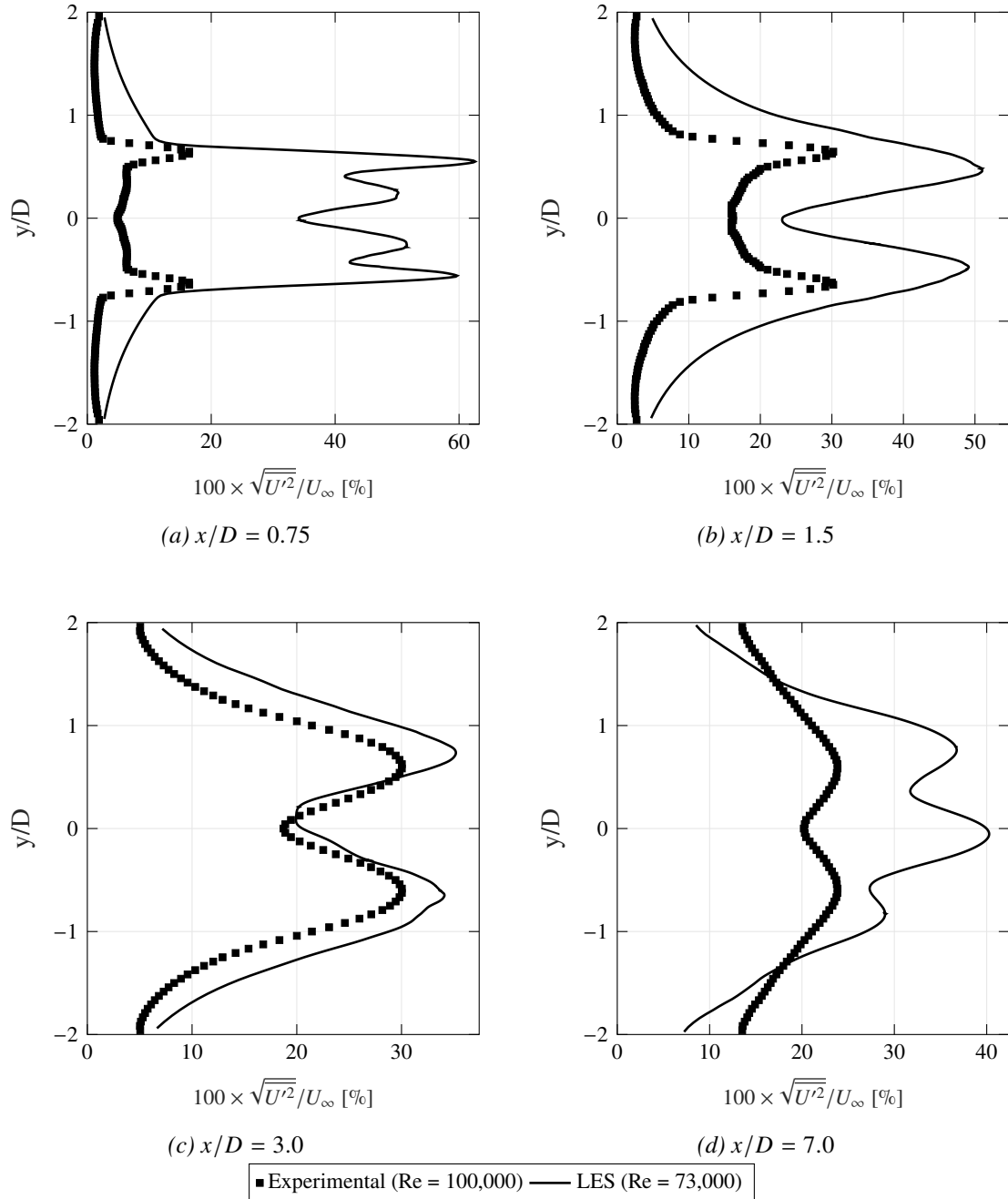


Figure 8.7: Turbulence intensity profiles in the wake comparing the experimental and CFD SPCCs.

Considering that the resolved LES SPCC investigation was conducted at a lower Reynolds number than the experiments, but both still within the sub-critical Reynolds number regime, it would be expected that

the resolved LES exhibit lower turbulence intensity in the wake compared to the experiments. However, Figure 8.7 showed that this was not the case.

Figure 8.8 shows normalised  $Q$ -criterion iso surfaces ( $Q_{norm} = 10$ ) coloured by mean velocity magnitude, where  $Q_{norm} = Q \times D^2/U_\infty^2$ . From the figure, large scale quasi-2-D coherent structures can be seen persisting in the wake. Given that the tested span was only  $0.23D$ , it appears that coherent structures are unable to break down into 3-D turbulence, hence resulting in the increased turbulence intensity observed in the wake.

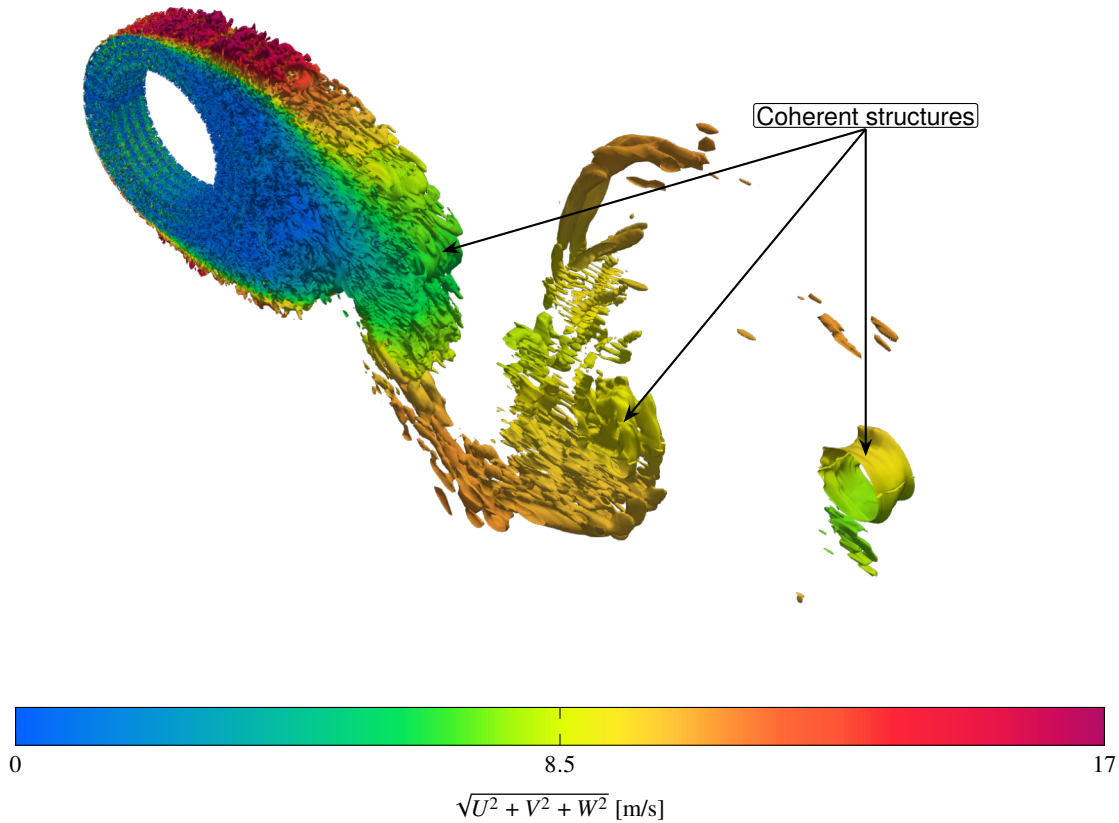


Figure 8.8:  $Q$ -criterion iso-surfaces coloured by mean velocity magnitude,  $Q_{norm} = 10$ .

### 8.3 Summary

In this chapter, the first known fully resolved LES of a Structured Porous Coated Cylinder was conducted using OpenFOAM's rhoPimpleFoam solver. The study examined the ability of the numerical setup to capture the complex coupled flow between the outer shear layer and the internal porous structure at  $Re_D = 73,000$  and  $M = 0.033$ .

The instantaneous and mean flow fields showed that the SPCC strongly modifies the near-cylinder flow compared with a smooth cylinder. Alternating regions of positive and negative spanwise vorticity were observed around the internal structural members, indicating local separation within the porous coating. Flow exiting through the circumferential pores also interacted with the outer shear layer, helping to diffuse the shear layer and promote earlier separation from the coated cylinder.

The streamwise velocity and streamline results showed clear interaction between the external flow and the

internal porous structure. Regions of reversed flow were observed within the porous coating, while the mean velocity field showed evidence of boundary-layer development on the inner cylinder. The flow also formed sub-channels within the porous layer, with some of the flow exiting through the circumferential pores before reaching the leeward side of the SPCC. These features demonstrate the complex redistribution of momentum caused by the structured porous coating.

The wake profiles showed mixed agreement with the experimental data. At  $x/D = 0.75$ , the LES predicted the mean streamwise velocity profile reasonably well, although with a slightly deeper wake deficit. Further downstream, at  $x/D = 1.5$  and  $x/D = 3.0$ , the wake deficit was underpredicted, producing narrower and shallower profiles than the experiments. By  $x/D = 7.0$ , better agreement was obtained in terms of both peak deficit and wake width, although a slight asymmetry remained, likely due to the limited temporal averaging window.

The turbulent kinetic energy was consistently overpredicted at all wake locations, with the largest discrepancy occurring at  $x/D = 0.75$ . Four distinct peaks were observed in the near wake, associated with vortex shedding from both the inner and outer diameters of the porous coating. The Q-criterion iso-surfaces suggested that large-scale quasi-2-D coherent structures persisted far into the wake. Since the computational span was limited to  $0.23D$ , these structures were likely unable to fully break down into 3-D turbulence, contributing to both the overprediction of turbulent kinetic energy and the asymmetry in the far wake.

The fully resolved LES captured the main flow mechanisms associated with the SPCC, including internal separation, circumferential pore outflow, shear-layer diffusion, inner-cylinder boundary-layer development and wake formation. While the mean wake predictions showed reasonable agreement with the experimental data at selected locations, the turbulent kinetic energy was consistently overpredicted. This was most likely caused by the limited spanwise extent of the computational domain.

# 9 Darcy–Forschheimer Modelling of a Structured Porous Cylinder Coating in Uniform Flow

The results presented in Chapter 8 highlighted the need to increase the spanwise extent of the SPCC. However, fully resolving the geometry over a larger span would incur a significantly greater computational cost. Therefore, alternative modelling approaches needed to be investigated. One of the most common approaches for modelling porous materials is the Darcy–Forschheimer model [160]. The Darcy–Forschheimer model is an extension of the linear viscous-resistance relationship proposed by Darcy [161], to accommodate for the non-linear inertial-resistance at higher flow velocities. A calibration method to revise the typical Darcy–Forschheimer coefficient is proposed and tested.

## 9.1 The Darcy–Forschheimer Model

The Darcy–Forschheimer model introduces a sink term into the momentum equation:

$$\frac{\partial (\rho \mathbf{U})}{\partial t} + \nabla \cdot (\rho \mathbf{U} \mathbf{U}) = -\nabla p + \nabla \cdot \boldsymbol{\tau} + \mathbf{S}_m \quad (9.1)$$

Where the sink term is given by:

$$\mathbf{S}_m = - \left( \mu \mathbf{D}_m + \frac{1}{2} \rho |\mathbf{U}| \mathbf{F} \right) \cdot \mathbf{U} \quad (9.2)$$

The pressure drop as a function of velocity is given by:

$$\frac{\Delta p}{\Delta x} = A u + B u^2 \quad (9.3)$$

Resulting in:

$$A = \mu D_m \quad (9.4)$$

and:

$$B = \frac{1}{2} \rho F \quad (9.5)$$

The coefficients can then be determined by

$$D_m = \frac{A}{\mu} \quad (9.6)$$

and

$$F = \frac{2B}{\rho} \quad (9.7)$$

The SPCC used for the modelling is shown in Figure 9.1a, with each section that composes the SPCC shown in Figure 9.1b. The SPCC was designed by Arcondoulis et al. [117], [118] and has been studied extensively in [121], [119], [120], [162].

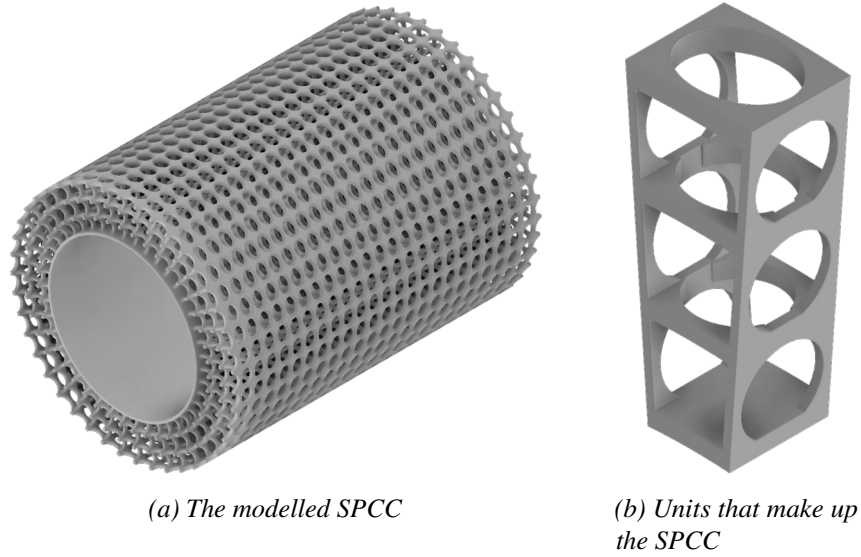


Figure 9.1: The modelled SPCC and the units that compose it

Given the porosity of  $\phi = 0.87$ , the SPCC can be approximated by an isotropic cubic structure, as shown in Figure 9.2a, with the unit cell shown in Figure 9.2b. The structure thickness of the idealised shape had a thickness of  $0.0125D$  and a spacing between structures of  $0.043D$ .

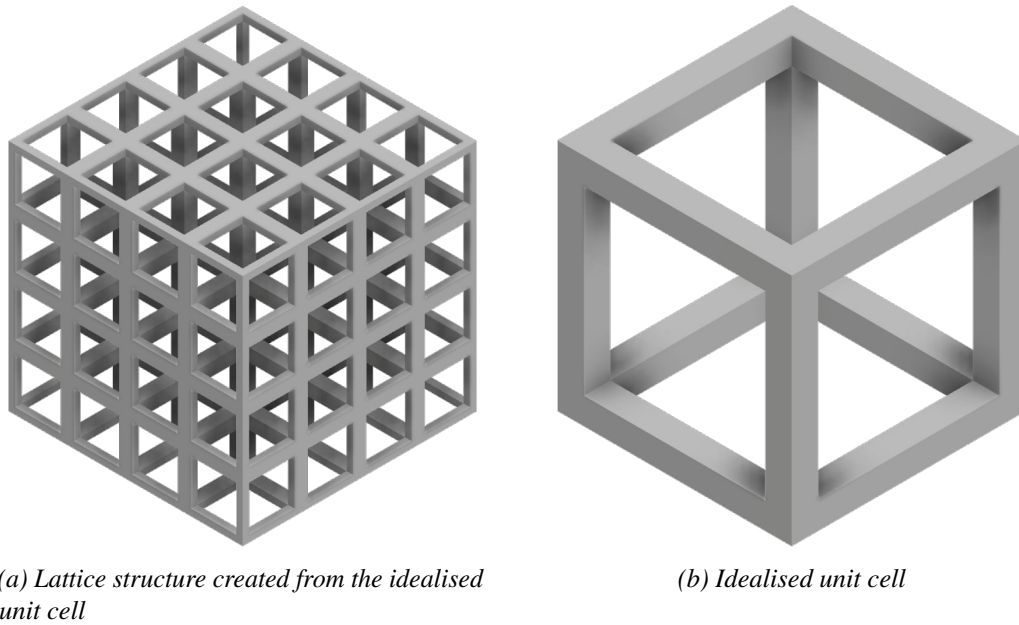


Figure 9.2: Idealised lattice structure and unit cell.

Figure 9.3 shows the pressure gradient across the idealised unit cell as a function of velocity. A line was fitted to the curve in the form of  $dp/dx = BU^2 + AU$ . From the fitted curve, the value of  $B$  was used to determine  $F$  according to Equation 9.7. The value of  $F$  was found to be 203 [1/m].

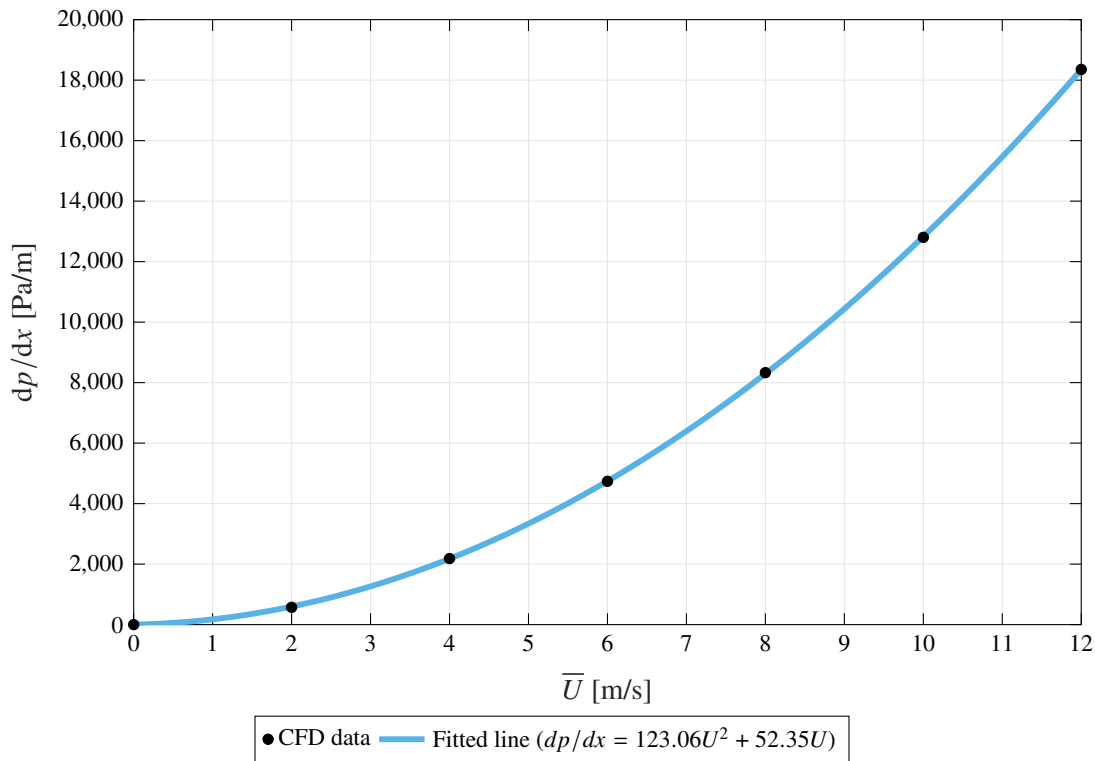


Figure 9.3: Pressure gradient as a function of velocity

Several studies ([163], [164], [165]) have demonstrated that, at large pore-scale Reynolds numbers, the Forchheimer contribution becomes dominant. Therefore, only the Forchheimer term was retained. This coefficient, however, is obtained under the assumption of uniform flow through the material, whereas the flow normal to the cylinder is inherently reduced in the vicinity of the inner cylinder surface. Based on inviscid, incompressible, irrotational potential flow around a cylinder, the velocity component normal to the wall is described by Equation 9.8, where  $a$  is the cylinder radius and  $x$  is the distance from the cylinder centre:

$$U_N = U_\infty \left(1 - \frac{a^2}{x^2}\right) \quad (9.8)$$

Equation 9.8 was then used to estimate the mean velocity normal to the wall within the coating,  $U_{internal}$ , relative to the normal velocity component entering the coating,  $U_{in}$ . For the thickness ratio considered here,  $U_{internal}/U_{in}$  was found to be 0.598. As the Forchheimer term varies with  $U^2$ ,  $F$  was multiplied by 0.358, giving a revised value of  $F = 73$  [1/m].



## 9.2 Darcy–Forchheimer Approximated SPCC in Uniform Flow

### 9.2.1 Methodology

#### 9.2.1.1 Computational Domain

Since the structural implications of the coating were outside the scope of this research, the coatings were treated as an addition to a bare cylinder. Therefore, the uncoated (bare) cylinder had a diameter of  $d = 0.032$  m. To create the Darcy–Forchheimer region, a volume zone with a thickness of  $0.008$  m was placed around the bare cylinder, resulting in an outer diameter of  $D = 0.048$  m for the coated case. However, for the purposes of validation against the experiments, in which both the bare and coated cylinders had an outer diameter of  $0.048$  m, an additional bare-cylinder case was run with a diameter of  $0.048$  m. The bare cylinder with a diameter of  $0.032$  m is referred to as the “small” case, while the bare cylinder with a diameter of  $0.048$  m is referred to as the “large” case.

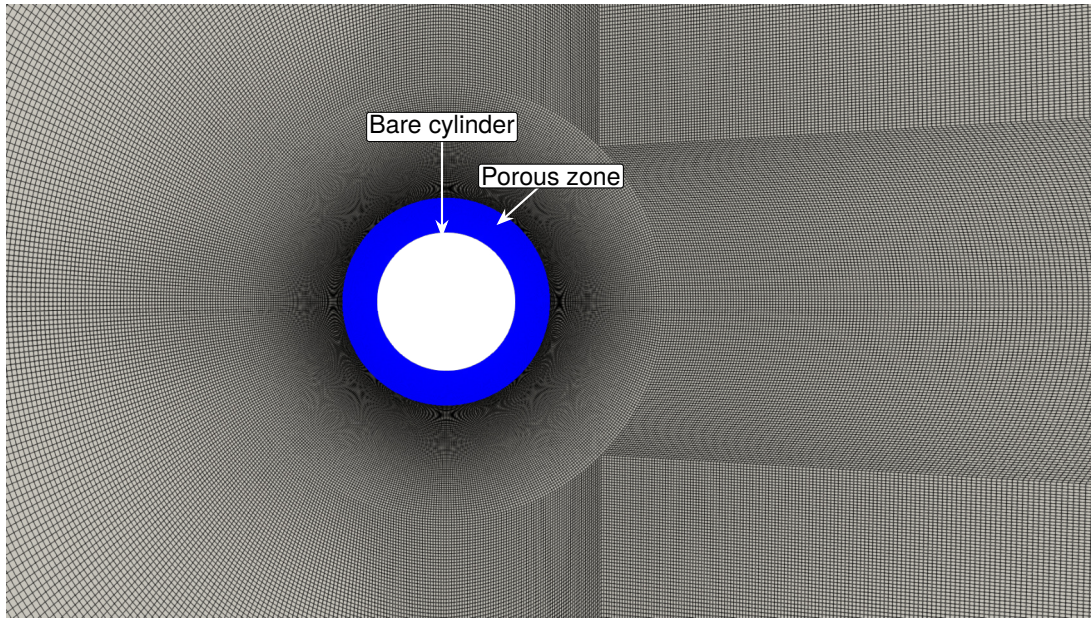


Figure 9.4: Computational domain around the cylinder showing the porous zone

The computational domain (shown in Figure 9.5) contained approximately 127M hexahedral cells, and extended  $10D$  above and below the cylinder centre and  $50D$  in the wake. The span of the domain was  $3.66D$ .

#### 9.2.1.2 Numerical Methods

The simulations were conducted in OpenFOAM v2212 using the rhoPimpleFoam solver. For this study,  $y_{\text{RANS}}^+$  was set to 30 and  $y_{\text{LES}}^+$  was set to 60. In all cases, temporal discretisation was performed using the second-order backward difference scheme. For spatial discretisation, the LUST scheme was employed. The simulations were run for  $193.75T^*$  with temporal statistics conducted over  $129.17T^*$ , where  $T^* = D/U_\infty$ . The transient timestep was  $4.04 \times 10^{-4}T^*$  for the small bare cylinder,  $6.46 \times 10^{-4}T^*$  for the large bare cylinder and  $1.29 \times 10^{-3}T^*$  for both the coated cylinders.

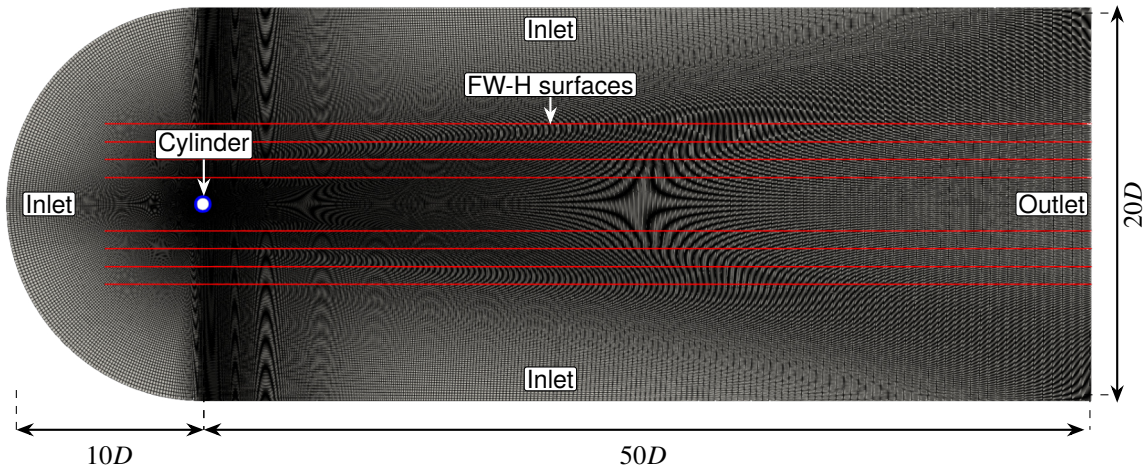


Figure 9.5: Computational domain

### 9.2.1.3 Boundary Conditions

At the inlet, the velocity was set to  $U_\infty = 31$  m/s, corresponding to a diameter-based Reynolds number of  $Re_d = 67,000$  for the bare cylinder, and  $Re_D = 100,000$  for the coated cylinder, and a Mach number of  $M = 0.091$ , while a zero-gradient condition was set for pressure. At the outlet, a zero-gradient condition was applied for velocity, while the pressure was set to  $p = 101,325$  Pa. A no-slip condition was applied at the walls, and spanwise periodicity was used.

### 9.2.1.4 Acoustics

The observer locations used for both the experiment and CFD are shown in Figure 9.6. For the CFD, the array was positioned along the radius  $r = 20.8D$ , in  $30^\circ$  intervals between  $30^\circ$  and  $150^\circ$ . The experiments contained one microphone at the  $90^\circ$  polar angle and  $12.7D$  above the cylinder centre.

The permeable FW-H surfaces (shown in Figure 9.5) extended  $5.5D$  upstream of the cylinder centre and to the end of computational domain downstream. Four FW-H surfaces were used, with the closest being located at  $y/D = 1.5$  and the space between adjacent surfaces being  $D$ . The surfaces were chosen on a case-specific basis, i.e., where the difference in average OASPL between adjacent surfaces dropped to below 1 dB. For the uncoated CFD it was placed at a height  $y/D = 2.5$  above and below the cylinder centre. For both the coated cases, it was placed at  $y/D = 3.5$  above and below the cylinder centre. All surfaces had a uniform resolution of  $0.067D$ . The Farassat 1A formulation [30] of the FW-H equation [29] was used.

The CFD acoustic data was recorded over  $0.2$  s, or  $129T^*$ , where  $T^* = D/U$ , with a sampling frequency of  $50$  kHz. Welch's method was then applied to the raw pressure data using a Hann window. The analysis was performed with a Fast Fourier Transform (FFT) length of  $4096$  and a  $50\%$  overlap between consecutive segments. The acoustic pressure power spectral density  $PSD_p(f)$  was first calculated in linear units of  $\text{Pa}^2/\text{Hz}$ . It was then expressed on a logarithmic decibel scale as  $\phi_{pp}(f)$  [dB/Hz], using a reference acoustic pressure of  $P_{\text{ref}} = 2 \times 10^{-5}$  Pa according to Equation 9.9:



$$\phi_{pp}(f) = 10 \log_{10} \left( \frac{\text{PSD}_p(f)}{P_{\text{ref}}^2} \right), \quad P_{\text{ref}} = 2 \times 10^{-5} \text{ Pa.} \quad (9.9)$$

The experimental data was recorded for  $10,000T^*$  at a sampling frequency of 51.2 kHz, and Welch's method was also applied using a Hann window and an FFT length of 51,200.

To accommodate for the difference in both span and microphone location between the CFD and experimental data, the CFD data was adjusted according to Equation 9.10:

$$\phi_{pp,corr}(f) = \phi_{pp}(f) + 10 \log_{10} \left( \frac{L_{exp}}{L_{CFD}} \right) + 20 \log_{10} \left( \frac{r_{CFD}}{r_{exp}} \right) \quad (9.10)$$

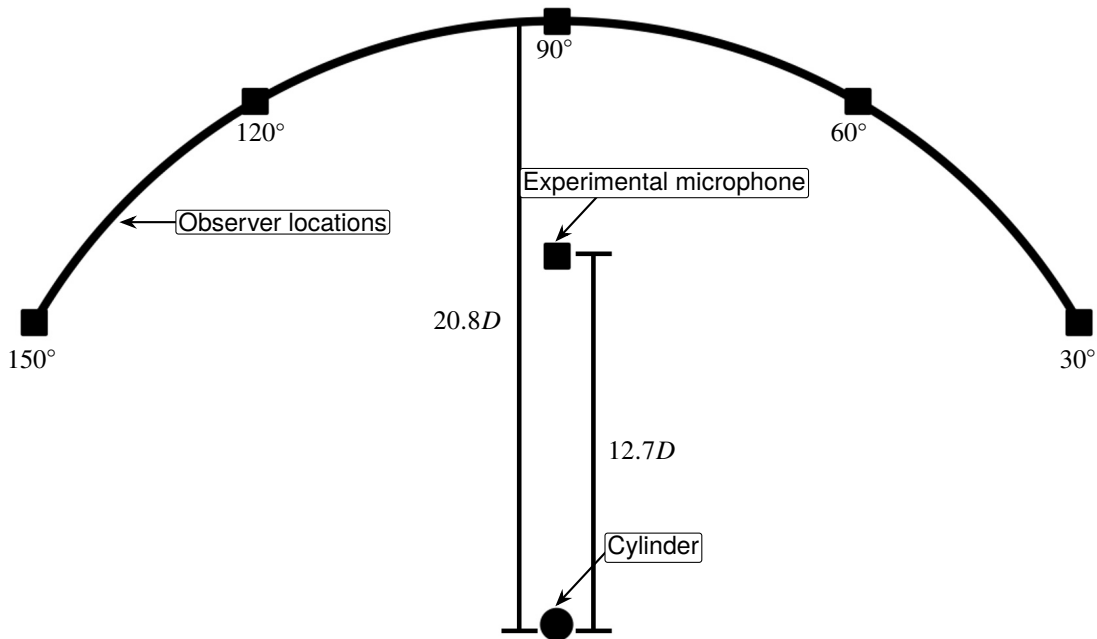


Figure 9.6: Observer locations

## 9.2.2 Results

### 9.2.2.1 Mean Velocity Wake Profiles

For comparison with the experiments, the instantaneous streamwise velocities were first converted to absolute values prior to computing the time-averaged mean. Figure 9.7 presents the normalised streamwise mean velocity profiles.

At  $x/D = 0.75$  (Figure 9.7a), both the  $F = 203$  and  $F = 73$  cases show good agreement with the experimental data. The  $F = 203$  case captures the wake deficit width well but slightly underpredicts the peak velocity. In contrast,  $F = 73$  captures the peak accurately but exhibits a slightly narrower deficit near  $y = 0$ .

At  $x/D = 1.5$  (Figure 9.7b),  $F = 203$  predicts a significantly wider and deeper wake deficit compared to the experiment. The  $F = 73$  case captures the deficit width more accurately but underpredicts the peak

velocity. A similar trend is observed at  $x/D = 3.0$  (Figure 9.7c), where  $F = 203$  again exhibits an overly wide and deep deficit, while  $F = 73$  reproduces the width well but begins to slightly overpredict the peak.

Further downstream at  $x/D = 7$  (Figure 9.7d), very good agreement is observed between  $F = 203$  and the experimental data. The  $F = 73$  case, however, slightly overpredicts both the wake width and the peak velocity.

At all streamwise locations, both the experimental and CFD bare (small) cylinders exhibit shallower wake deficits compared to their coated counterparts.

The CFD bare (large) cylinder shows an overprediction in the wake width and an underprediction in the peak velocity compared to the experimental bare cylinder at all wake locations.

Figure 9.8 shows contours of the normalised mean streamwise velocity ( $\bar{U}/U_\infty$ ) for the bare cylinder (Figure 9.8a) and the coated cylinder with  $F = 73$  (Figure 9.8b). For the bare cylinder, there is a pronounced velocity deficit in the near-wake region, forming a relatively short and narrow low-momentum core. The velocity gradients along the shear layers are steep, indicating a strong momentum exchange with the surrounding flow. The coated case on the other hand exhibits a broader wake with the low-velocity region extending further downstream. The shear layers appear to develop more gradually, and the wake recovery occurs over a longer streamwise distance, suggesting a redistribution of mean momentum compared to the bare cylinder.

### 9.2.2.2 Turbulence Intensity Wake Profiles

Similar to the mean data, the unsteady streamwise velocity fluctuations were converted to absolute values prior to evaluating the TI.

Figure 9.9a shows good agreement between the experimental data and the  $F = 73$  case, with  $F = 73$  accurately capturing the peak magnitudes. In contrast,  $F = 203$  significantly underpredicts the peaks. A similar trend is observed in Figure 9.9b, where  $F = 73$  again captures the peaks well, although a slight underprediction of the local minimum between the peaks is evident. The  $F = 203$  case continues to substantially underpredict the peak TI levels. In both near-wake locations, the coated cylinders exhibit lower TI compared to the bare cylinders.

Further downstream, Figure 9.9c shows that both  $F = 73$  and  $F = 203$  agree well with the experimental data at the local minimum; however, both cases overpredict the peak values. The bare cylinder CFD (small) case exhibits significantly lower TI than the coated configurations, which is consistent with the lower Reynolds number of this case.

At  $x/D = 7$  (Figure 9.9d),  $F = 203$  matches the experimental peak values well but underpredicts the local minimum. The  $F = 73$  case overpredicts the peaks while also underpredicting the local minimum. Similar to the behaviour at  $x/D = 3$ , the bare cylinder CFD (small) case exhibits considerably lower TI at this downstream location.

Up to  $x/D = 3$ , the bare cylinder (large), shows relatively good agreement with the experimental bare cylinder, with a slight overprediction in the TI at these locations. At  $x/D = 7$ , the peaks of the bare cylinder CFD (large) agree well with the experimental bare cylinder, although the local minimum between

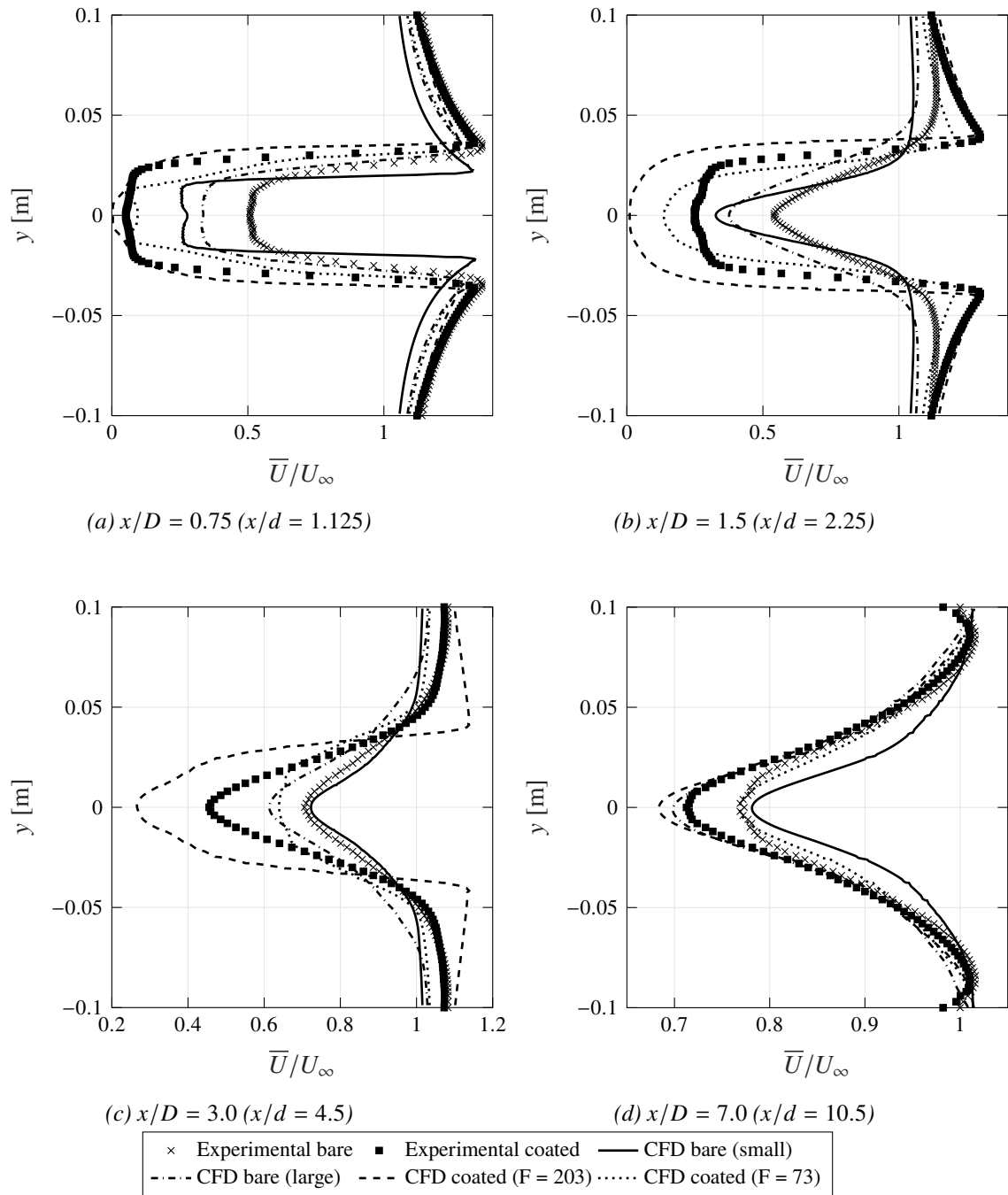


Figure 9.7: Comparison of normalised streamwise mean velocity between the bare and coated cylinders

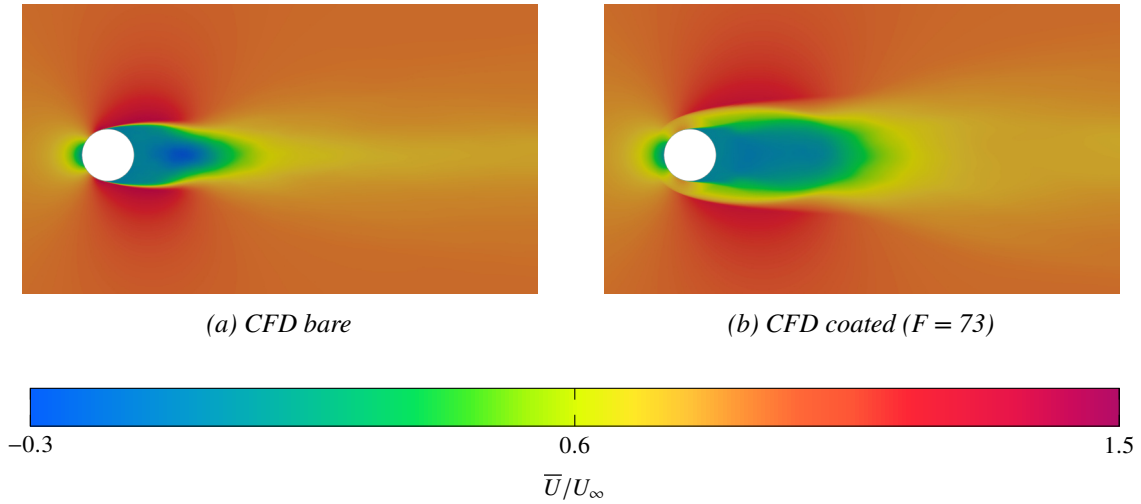


Figure 9.8: Contours of normalised streamwise mean velocity comparing the bare and coated cylinders

peaks is underpredicted.

Figure 9.10 shows contours of TI for the bare cylinder (Figure 9.10a) and the coated cylinder with  $F = 73$  (Figure 9.10b). For the bare cylinder, elevated regions of TI occur immediately downstream of the cylinder, forming two symmetric high-energy regions associated with shear layer roll-up. For the coated cylinder, the regions of elevated TI are shifted further downstream. Although these high-energy regions are broader than that of the bare cylinder, their peak values are reduced, suggesting a delayed growth of velocity fluctuations in the near-wake.

### 9.2.2.3 Acoustic Power Spectral Density

For the CFD bare cylinder, Figures 9.11a, 9.11b and 9.11c all show distinct peaks at approximately 208 Hz, corresponding to a Strouhal number of  $St = 0.21$ , where  $St = \frac{fd}{U}$  for the bare cylinder CFD and  $St = \frac{fD}{U}$  for all other cases. This peak is consistent across all observer angles for the bare cylinder configuration.

At the upstream polar angle ( $\theta = 150^\circ$ ), Figure 9.11a shows similar levels of  $\phi_{pp}$  for the CFD bare cylinder and the coated cylinder with  $F = 73$ . The  $F = 203$  cylinder exhibits generally higher  $\phi_{pp}$  levels than the bare cylinder up to approximately 1.2 kHz.

In Figure 9.11b, the experimental coated cylinder shows two distinct peaks at  $St = 0.2$  and  $St = 0.4$ , which were identified by the authors in [119] as  $f_1$  and  $f_2$  tones, corresponding to vortex shedding from the outer diameter of the coating and drag fluctuations from the bare cylinder respectively. For the CFD, the relatively short simulation time meant that these peaks were not clearly captured in the acoustic PSD. In the work done by Geyer et al. [126], the peaks not being captured was also attributed to the low frequency resolution. The peaks were, however, captured in the velocity PSD and are discussed in Section 9.2.2.4. Nevertheless, Figure 9.11b shows good agreement between the experimental coated cylinder and the CFD coated cylinder with  $F = 73$  in the mid-frequency range, with the two spectra deviating beyond approximately 1 kHz. The CFD coated cylinder with  $F = 203$  appears to overpredict  $\phi_{pp}$  in this mid-frequency range compared to the experiment and similarly deviates beyond 1 kHz.

Figure 9.11c shows that the  $F = 203$  configuration exhibits a similar trend to that observed at  $\theta = 150^\circ$ ,

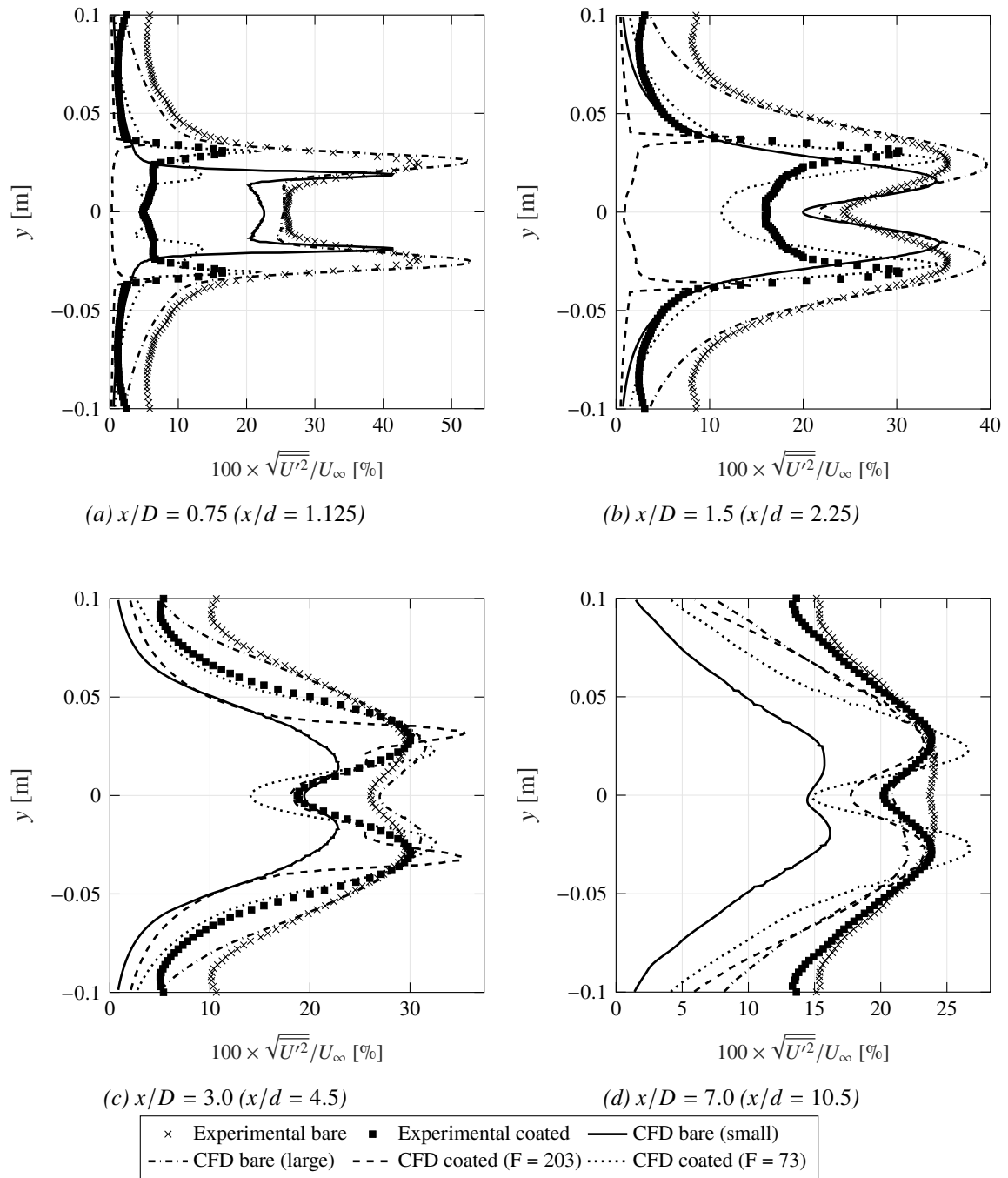


Figure 9.9: Comparison of turbulence intensity between the bare and coated cylinders

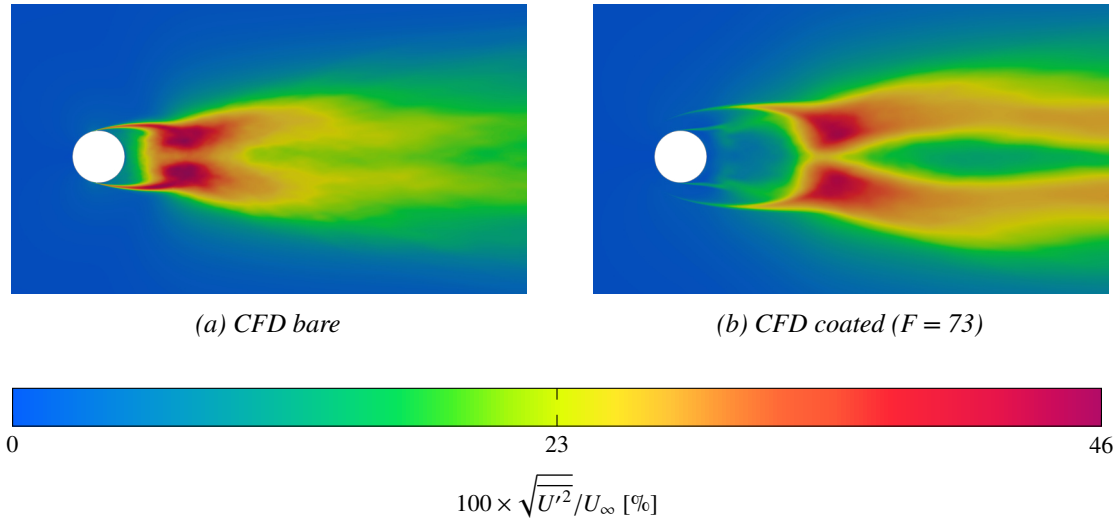


Figure 9.10: Contours of turbulence intensity comparing the bare and coated cylinders

with generally higher  $\phi_{pp}$  levels compared to the bare cylinder up to approximately 1 kHz. In contrast, the  $F = 73$  configuration exhibits generally lower  $\phi_{pp}$  levels in the mid- to high-frequency range.

The Overall Sound Pressure Level (OASPL) in Figure 9.11d shows a 2.85 dB reduction for  $F = 73$  compared to the CFD bare cylinder (small) at  $\theta = 90^\circ$ , and a 3.12 dB reduction at  $\theta = 60^\circ$ . When compared with the large bare cylinder, the noise reduction becomes more pronounced with a 14.67 dB and 15.39 dB reduction at the  $\theta = 90^\circ$  and  $\theta = 60^\circ$  polar angles respectively.

#### 9.2.2.4 Velocity Power Spectral Density

Figure 9.12 shows the probe locations used to determine the velocity power spectral density,  $\phi_{uu}$ . The points were selected downstream of the cylinders to align with the top of the bare cylinder (points 1 and 5), and the coated cylinder (points 2 and 6). Additional points were positioned further above the cylinders (points 3, 4, 7, and 8). These locations were chosen to assess changes in the shear-layer characteristics downstream of each cylinder, as well as the flow behaviour further from the cylinder surfaces.

The velocity power spectral density  $\text{PSD}_u(f)$  was first calculated in linear units of  $(\text{m/s})^2/\text{Hz}$ . It was then expressed on a logarithmic decibel scale as  $\phi_{uu}(f)$  [dB/Hz], using a reference velocity of 1 m/s according to Equation 9.11:

$$\phi_{uu}(f) = 10 \log_{10} \left( \frac{\text{PSD}_u(f)}{U_{\text{ref}}^2} \right), \quad U_{\text{ref}} = 1 \text{ m/s}. \quad (9.11)$$

Figure 9.13 shows the velocity PSD at the various locations shown in Figure 9.12.

The coated cylinder with  $F = 203$  shows a general reduction in the spectral energy of the velocity fluctuations across the frequency range, relative to the bare cylinder, at all probe locations.

The coated cylinder with  $F = 73$  similarly shows a general reduction in high-frequency content compared with the bare cylinder at all probed locations. The attenuation at low frequencies, however, appears to

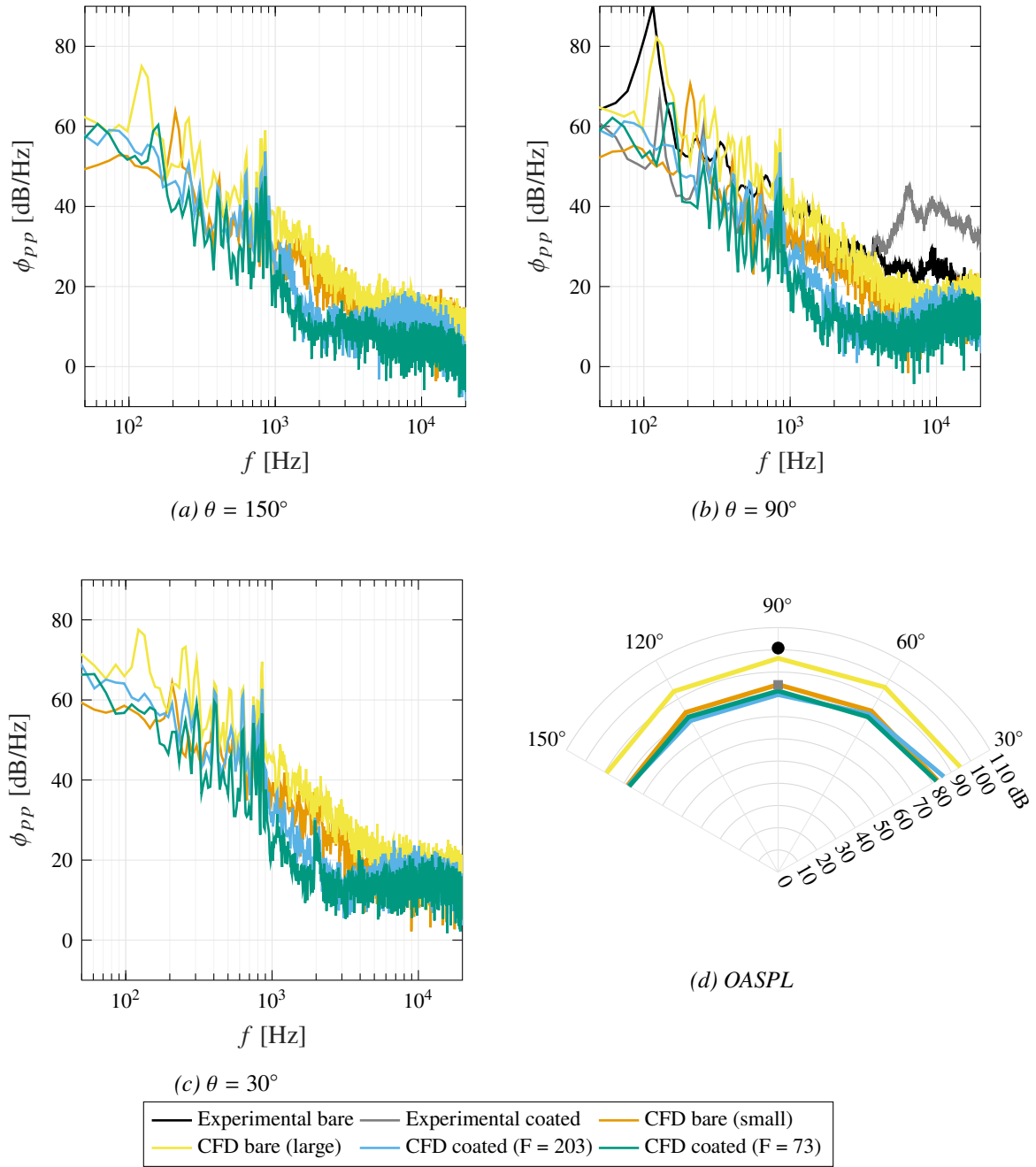
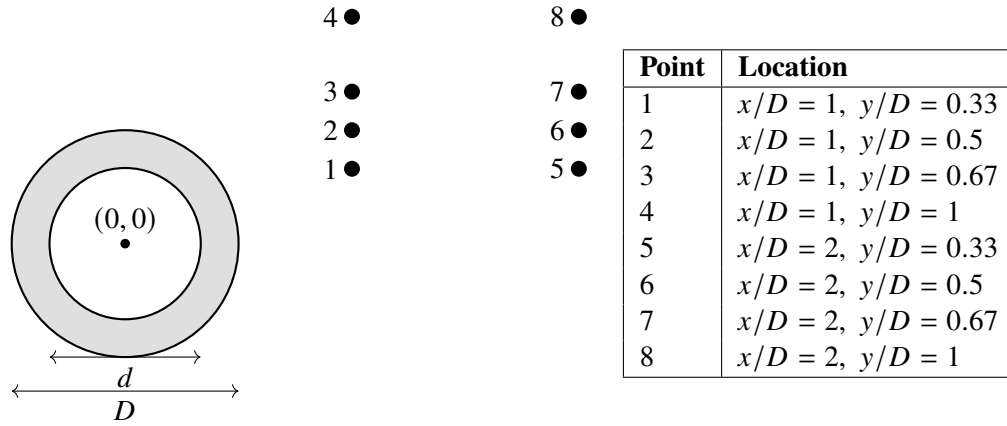


Figure 9.11: Acoustic PSD ( $\phi_{pp}$ ) [dB/Hz] as a function of frequency  $f$  [Hz]

Figure 9.12:  $\phi_{uu}$  probe locations.

be more dependent on probe location. The primary tonal peak is larger for the coated cylinder than for the bare cylinder at all probed locations except at  $x/D = 1, y/D = 0.33$  (9.13a). The secondary tone, which was not clearly captured in the acoustic PSD, is also visible at all probed locations for the coated cylinder with  $F = 73$ , suggesting that the short recording duration is the most likely reason it was not resolved in the acoustic spectra. At  $x/D = 1, y/D = 0.67$  (9.13c), two additional distinct tertiary tones are also observed. One possible explanation for the tertiary tones is an interaction between the shear layer associated with the cylinder and that developing from the outer surface of the coating.

Figure 9.14 shows the instantaneous spanwise vorticity ( $\omega_z$ ) for the bare cylinder (Figure 9.14a) and the coated cylinder with  $F = 73$  (Figure 9.14b). For the bare cylinder, the shear layers separate sharply from the cylinder surface, resulting in regions of high vorticity near the separation points. In the near-wake, a strong shear layer roll-up followed by alternating vortical structures characteristic of a von Kármán vortex street can be noticed. For the coated cylinder, a noticeably different wake can be seen. The vorticity field near the surface appears more smoothly distributed, and the shear layers are more coherent and elongated before rolling up. The delay of the shear-layer roll-up leads to a more organized wake structure, suggesting that the coating has modified the instability mechanisms responsible for vortex shedding.

### 9.2.2.5 Drag Coefficient Comparison

The drag coefficient was determined using a wake survey method, similar to the work of Son and Cetiner [166], where the drag coefficient,  $C_D$ , is given by:

$$C_d = 2 \int_{-\infty}^{+\infty} \frac{U}{U_\infty} \left( 1 - \frac{U}{U_\infty} \right) d\eta + 2 \int_{-\infty}^{+\infty} \frac{\overline{v'^2} - \overline{u'^2}}{U_\infty^2} d\eta \quad (9.12)$$

Table 9.1 summarises the drag coefficients for each case. From Table 9.1, it can be seen that when compared to the large bare cylinder, the  $F = 73$  coated cylinder exhibits only a 0.8% increase in  $C_D$ . In the experimental work done by Arcondoulis et al. [119], for cylinders at  $Re_D = 1.66 \times 10^5$ , the coated cylinder showed a 1.63% increase in  $C_D$ , albeit for a different Reynolds number and using the resolved geometry. In the experiments, the  $C_D$  for the bare and coated cylinders was 1.23 and 1.25 respectively. Comparing the experimental data to the bare (large) and coated ( $F = 73$ ) CFD cases, the CFD overpredicts



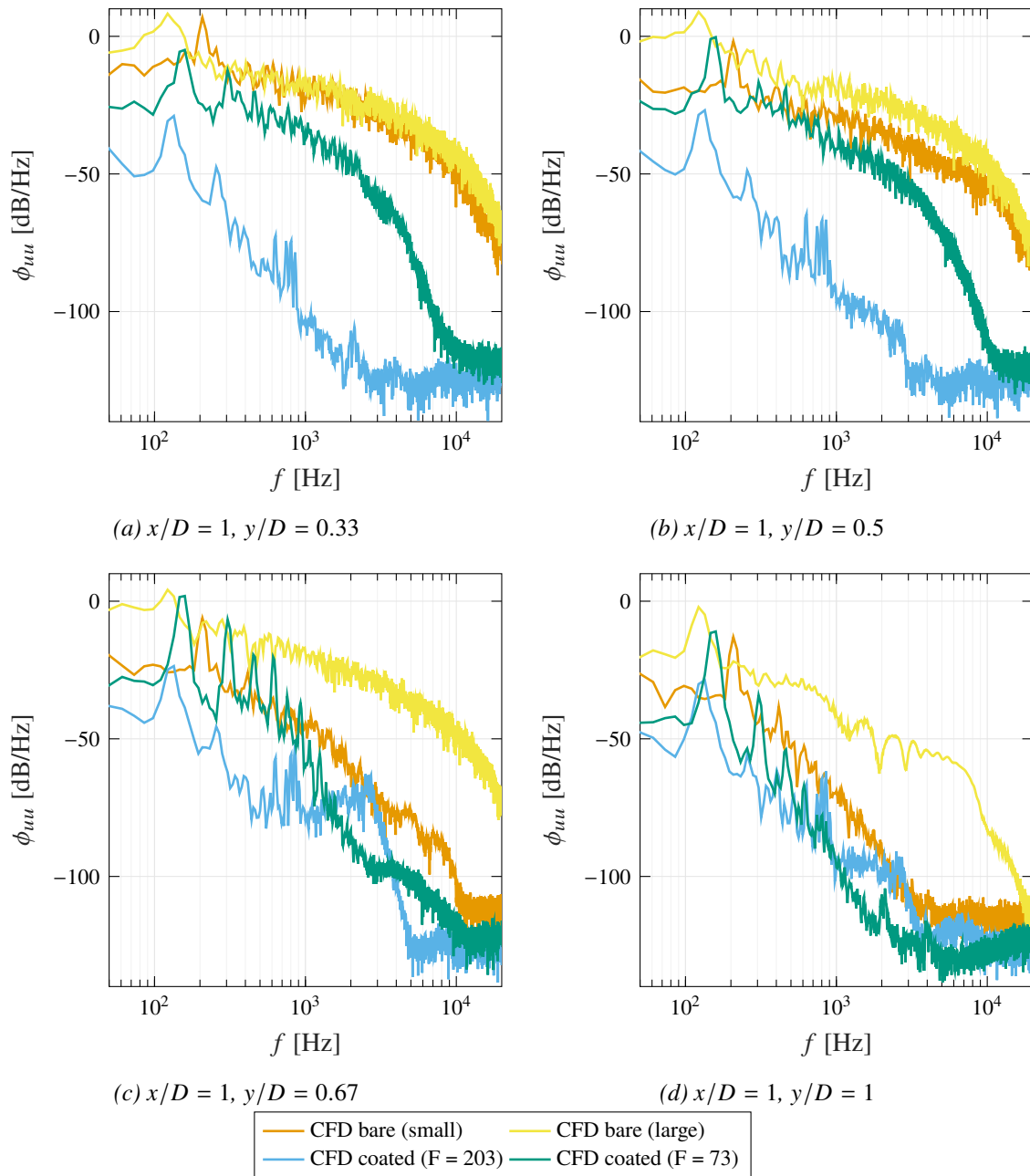


Figure 9.13: Velocity PSD ( $\phi_{uu}$ ) [dB/Hz] as a function of frequency  $f$  [Hz] (continued on next page).

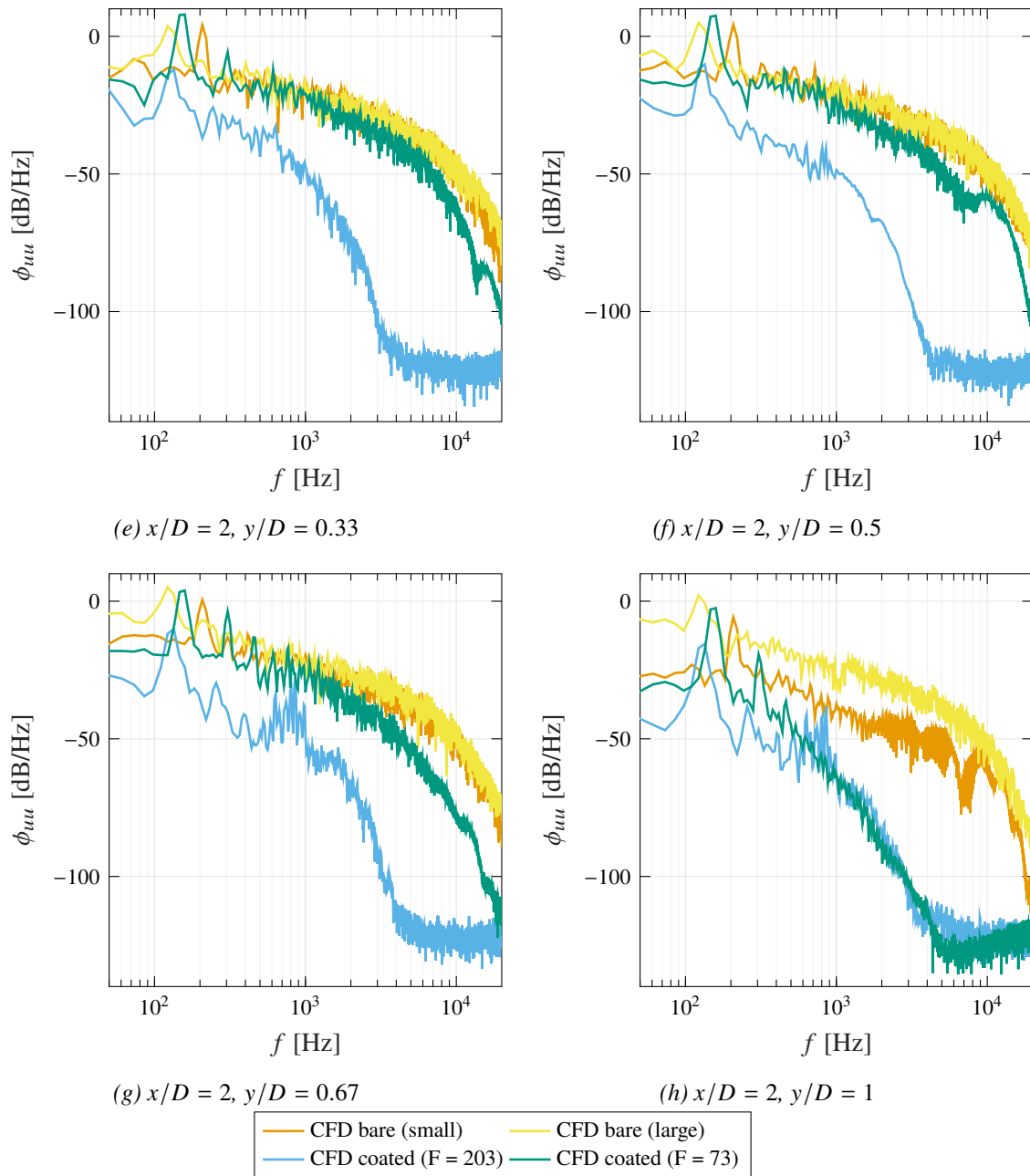


Figure 9.13: Velocity PSD ( $\phi_{uu}$ ) [dB/Hz] as a function of frequency  $f$  [Hz] (continued).

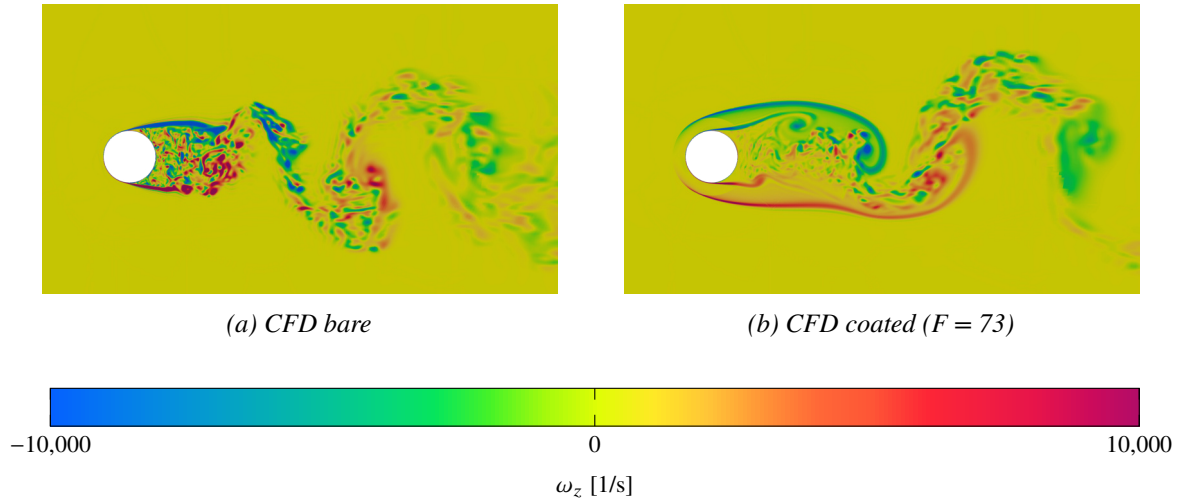


Figure 9.14: Instantaneous spanwise vorticity comparing the bare and coated cylinders.

the drag coefficient by 1.63% and 0.8% respectively, showing excellent agreement with the experimental data. However, when compared to the small bare cylinder, the  $F = 73$  coated cylinder exhibits a 7.7% higher  $C_D$ . Although the structural properties of an SPCC are beyond the scope of this research, the results demonstrate that when compared with a bare cylinder with the same outer diameter, the drag penalty is minimal, whilst when the coating is used as an addition onto the smaller bare cylinder, the drag penalty becomes more significant.

Table 9.1: Drag coefficients for each of the CFD cases.

| Bare (small) | Bare (large) | Coated ( $F = 203$ ) | Coated ( $F = 73$ ) |
|--------------|--------------|----------------------|---------------------|
| 1.17         | 1.25         | 1.27                 | 1.26                |

#### 9.2.2.6 Confirmation of Short Span Causing Increased TI in the Wake

In Chapter 8, it was conjectured that the short span of the resolved SPCC LES caused the increased TI in the wake compared to the experimental data. Figure 9.15 confirms this conjecture, where, especially in Figure 9.15b, it can be seen that the short-span Darcy–Forchheimer coated cylinder also exhibits higher TI in the wake, agreeing well with the short-span resolved LES, while the longer-span Darcy–Forchheimer coated cylinder agrees well with the experimental data.

### 9.2.3 Summary

In this chapter, a Darcy–Forchheimer approximation was developed and assessed for modelling the flow and acoustic behaviour of a structured porous coated cylinder. The model was introduced through an additional momentum sink term, with the resistance coefficients obtained from the pressure drop across an idealised porous unit cell. Two Forchheimer coefficients were assessed,  $F = 203$ , obtained from the unit-cell pressure-gradient fit, and  $F = 73$ , a calibrated value based on the slowing down of flow near the inner cylinder wall.

The mean wake profiles showed that both Darcy–Forchheimer coated-cylinder cases reproduced the main wake-broadening effect observed experimentally. Both porous cases produced deeper and wider wake deficits than the bare cylinder, indicating that the bulk influence of the coating was captured. The  $F = 203$

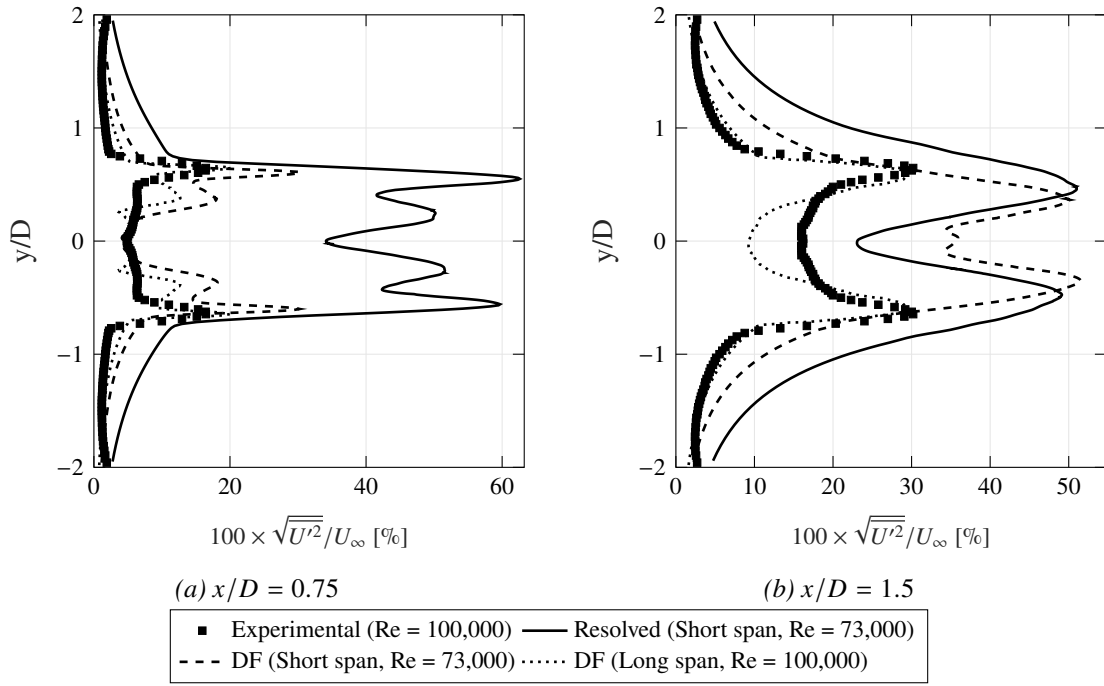


Figure 9.15: Turbulence intensity in the wake confirming that the short span caused the increased TI

case generally produced a wider and deeper wake, giving good agreement in the far wake at  $x/D = 7.0$ , but overpredicting the wake deficit at some intermediate locations. The  $F = 73$  case provided better agreement in wake width and peak magnitude at several locations, particularly in the near wake, although it slightly underpredicted the far-wake deficit.

The turbulent kinetic energy results were more sensitive to the selected Forchheimer coefficient. In the near wake, the  $F = 73$  case captured the experimental peak magnitudes more accurately, while the  $F = 203$  case significantly underpredicted the TKE peaks. Further downstream, both coated cases showed reasonable agreement in some regions but overpredicted the peak values at others. Overall, the porous coating shifted the high-TKE regions further downstream compared with the bare (small) cylinder, producing broader regions but with lower TKE peaks. This indicates that the Darcy–Forchheimer coating modified the separated shear-layer development and delayed the growth of velocity fluctuations in the near wake.

The acoustic predictions were obtained using permeable FW-H surfaces with the Farassat 1A formulation. The bare-cylinder case produced a clear tonal peak at approximately 208 Hz, corresponding to  $St = 0.21$ . For the coated cylinder, the experimental spectra contained two tones associated with shedding from the outer coating diameter and drag fluctuations from the inner bare cylinder. These tones were not clearly captured in the acoustic PSD, most likely due to the relatively short CFD recording duration and limited frequency resolution, although they were visible in the velocity PSD.

Despite this limitation, the  $F = 73$  case showed better agreement with the experimental coated-cylinder acoustic spectrum, particularly in the mid-frequency range at the  $90^\circ$  observer location. It also produced an OASPL reduction relative to the CFD bare cylinder, with reductions of 2.84 dB at  $90^\circ$  and 3.12 dB at  $60^\circ$ . In contrast, the  $F = 203$  case generally overpredicted the acoustic PSD in the mid-frequency range and produced higher levels than the bare cylinder at the  $30^\circ$  polar angle. This suggests that the lower

Forchheimer coefficient provided a more representative acoustic response for the coated configuration when compared to the experimental data.

The velocity PSD and instantaneous vorticity fields provided further insight into the wake dynamics. Both coated cases reduced the high-frequency velocity fluctuation content relative to the bare cylinder, while the  $F = 73$  case retained stronger low-frequency tonal behaviour. The secondary tone was visible at all probe locations, supporting the conjecture that the missing acoustic tones were mainly caused by insufficient acoustic recording duration. The vorticity fields also showed that the Darcy–Forchheimer coating produced a more distributed wake structure, with elongated shear layers and delayed vortex roll-up compared with the sharp separation and clear von Kármán vortex street of the bare cylinder.

The Darcy–Forchheimer approximation captured the principal wake-broadening and noise-modifying effects of the structured porous coating, while avoiding the computational cost of fully resolving the internal porous geometry. The  $F = 73$  case generally provided better agreement with the experimental turbulent kinetic energy and acoustic data, while the unit-cell-derived  $F = 203$  case gave better agreement in selected far-wake mean velocity profiles but tended to over-dampen near-wake turbulence and overpredict parts of the acoustic spectrum. Therefore, the Darcy–Forchheimer model provides a practical approach for representing the bulk effect of the structured porous coating and enabling larger-span simulations, although its accuracy remains sensitive to the selected resistance coefficient and it cannot fully reproduce all of the detailed flow mechanisms present in the resolved geometry.

# 10

## VORTEX IMPINGEMENT ON A CYLINDER AND AN AIRFOIL LEADING EDGE

The coated cylinders have so far been investigated in uniform flow conditions. However, in tilt-wing and related UAMV configurations, the rotors are positioned close to the wing and its supporting structures. Coherent vortical structures generated near the rotor-blade tips convect towards the pylon (idealised as a cylinder) during hover/take-off (Figure 10.1) and towards the wing during forward-flight (Figure 10.2), inducing unsteady surface-pressure fluctuations and disturbing the downstream flow, thus generating rotor-airframe installation noise. The impingement of an idealised modelled tip-vortex on a cylinder and an airfoil, with and without porous coatings, is investigated in this chapter.

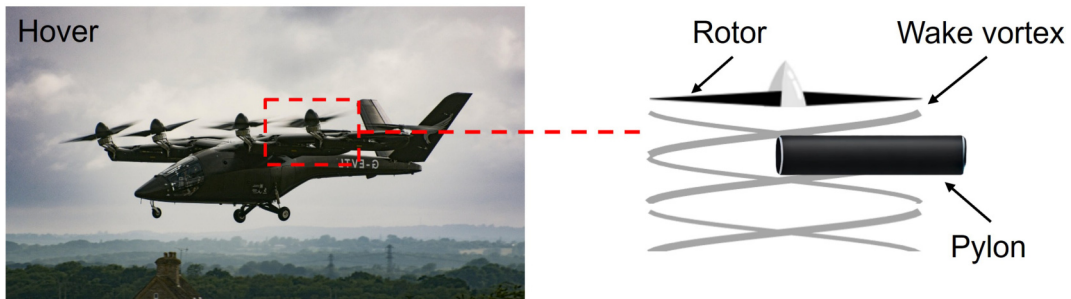


Figure 10.1: Visualisation of vortex impingement during hover/take-off (adapted from Vertical Magazine [7])

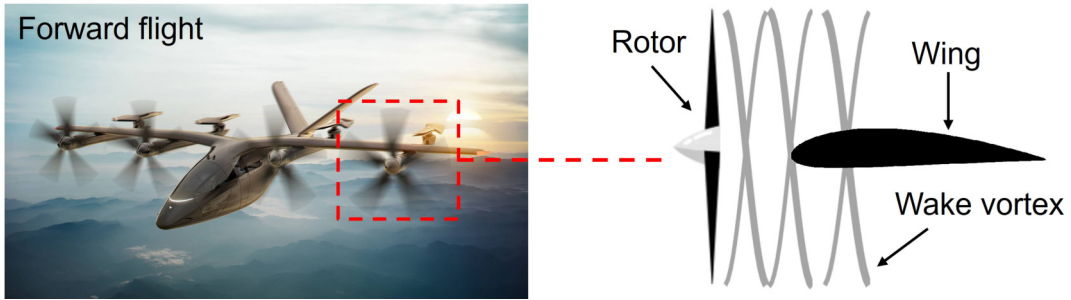


Figure 10.2: Visualisation of vortex impingement during forward-flight (adapted from Crumley [8])

### 10.1 Modelled Tip Vortex

To reduce computational cost and isolate the effects of tip-vortex interaction, the inflow tip vortex was modelled based on the coefficient of thrust ( $C_T$ ) of the APC  $12 \times 4.7$ SF (shown in Figure 10.3) propeller at 3000 RPM, which is  $C_T = 0.1185$ .

Two vortex models were compared in order to assist in the modelling of an appropriate tip vortex, the first being the Rankine Vortex (Equation 10.1) and the second being the Kaufmann vortex (Equation 10.2):



Figure 10.3: APC 12 × 4.7SF Propeller

$$V_{\theta}(r) = \begin{cases} \frac{\Gamma}{2\pi} \frac{r}{r_c^2}, & r \leq r_c \\ \frac{\Gamma}{2\pi} \frac{1}{r}, & r > r_c \end{cases} \quad (10.1)$$

$$V_{\theta}(r) = \frac{\Gamma}{2\pi} \frac{r}{r_c^2 + r^2} \quad (10.2)$$

The model chosen for the tip vortex was the Kaufmann vortex [167]. Figure 10.4 shows a comparison between the Rankine and Kaufmann vortices. A known limitation of the Rankine vortex is the sharp transition in  $V_{\theta}$  beyond the core radius  $r_c$ , hence the choice to use the Kaufmann vortex.

With  $C_T$  known and assuming ideal induced power ( $k_i = 1$ ), the rotor inflow ratio,  $\lambda$ , can be determined by Equation 10.3:

$$\lambda = k_i \sqrt{\frac{C_T}{2}} \quad (10.3)$$

The rotor tip speed,  $V_T$ , can be determined by Equation 10.4:

$$V_T = R \times \Omega \text{ m/s} \quad (10.4)$$

Where  $R$  is the radius of the blade in m and  $\Omega$  is the rotational speed of the propeller in rad/s.

Assuming hover, i.e., no freestream velocity, the induced velocity,  $v$ , can be found by Equation 10.5:

$$v = \sqrt{\frac{C_T}{2} \times k_i^2 \times V_T^2} \text{ [m/s]} \quad (10.5)$$

The vortex circulation strength,  $\Gamma$  can then be found by Equation 10.6

$$\Gamma = \frac{4\pi \times R \times v^2}{V_T} \text{ [m}^2\text{/s]} \quad (10.6)$$

The core radius can be found using Equation 10.7:

$$\ln(r_c/R) = \ln(8) - 0.25 \left( 9 - 2\ln(2) + \frac{C_T}{\lambda^3} \right) \quad (10.7)$$

Since the domains to be used for the inflow vortex contained periodic boundary conditions, the radius of the vortex was limited using a cosine smoothing function shown in Equation 10.8. The taper was intended to prevent the interaction of the vortex with the periodic boundaries, as well as prevent the interaction of the vortices with each other, allowing coherent vortex structures to interact with the porous coating.

$$f(r) = \frac{1}{2} \left[ 1 + \cos \left( \pi \frac{r - r_c}{r_{\text{cutoff}} - r_c} \right) \right] \quad (10.8)$$

With the cosine taper applied, the modified Kaufmann vortex equation is shown in Equation 10.9:

$$V_\theta(r) = \begin{cases} \frac{\Gamma}{2\pi} \frac{r}{r_c^2 + r^2}, & r \leq r_c \\ \frac{\Gamma}{2\pi} \frac{r}{r_c^2 + r^2} \left( \frac{1 + \cos \left( \pi \frac{r - r_c}{r_{\text{cutoff}} - r_c} \right)}{2} \right), & r_c < r \leq r_{\text{cutoff}} \\ 0, & r > r_{\text{cutoff}} \end{cases} \quad (10.9)$$

A similar cosine taper was also used to bring the vortex rotationally to rest at a desired height to prevent the interaction of the vortices with the permeable FW-H surfaces, thus isolating the captured noise to the interaction effect.

Figure 10.5 shows a comparison between the Rankine vortex (Figure 10.5a), the Kaufmann vortex (Figure 10.5b) and the modified vortices with  $r_{\text{cutoff}} = 2r_c$  (Figure 10.5c) and  $r_{\text{cutoff}} = 3r_c$  (Figure 10.5d). From Figure 10.5, it can be seen that both the Rankine and original Kaufmann vortices have significant  $V_\theta$  away from the vortex core, whilst the modified vortices smoothly bring the vortex to rest at the chosen distance from the core.



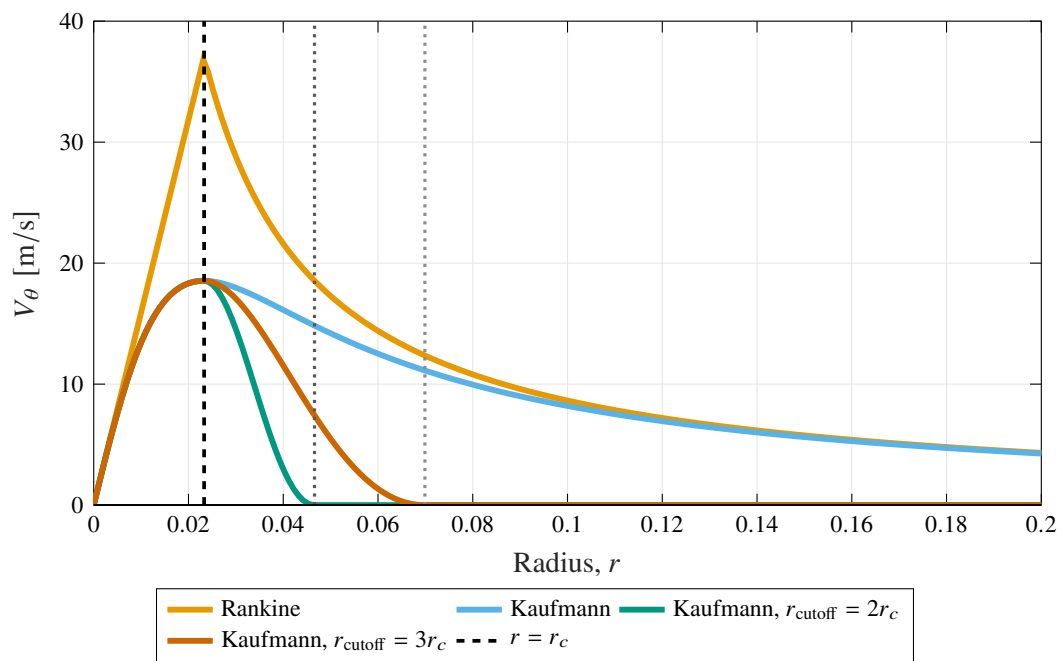


Figure 10.4: Vortex comparison

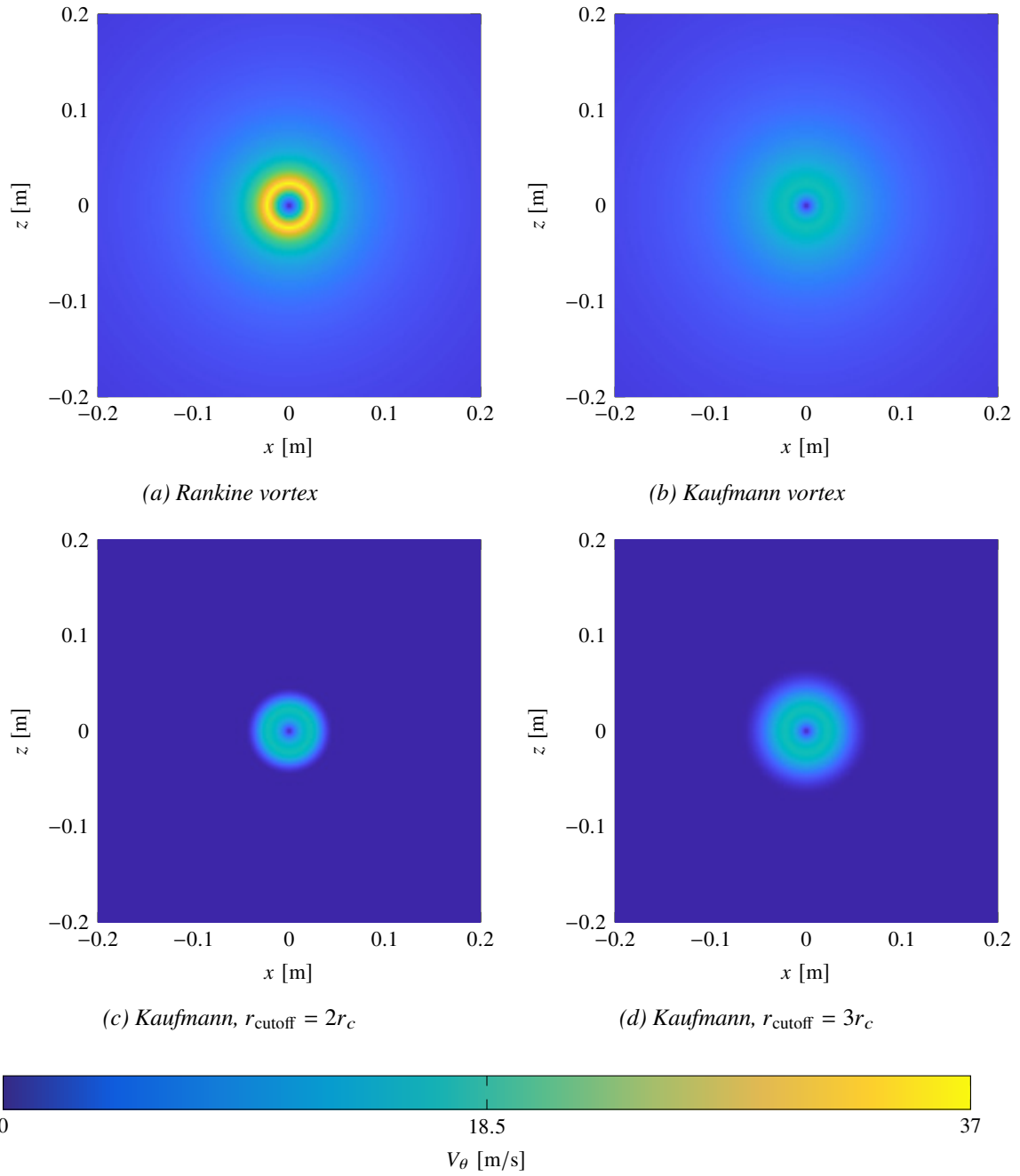


Figure 10.5: Comparison of vortex models: Rankine, Kaufmann, and modified Kaufmann vortices.

## 10.2 Vortex Impingement on Cylinder

For certain EUAMVs operating in the hover and take-off flight phases, vortices shed from the propeller tips can interact with the pylon, producing installation noise. A schematic of this interaction is shown in Figure 10.6a. To assess the extent of this noise, as well as the effect of applying a porous coating to the pylon, a series of tube vortices were introduced into the computational domain and allowed to impinge on the pylon, which was modelled as a cylinder, as shown in Figure 10.6b. The cylinder span is oriented in the  $z$ -direction, with the inflow in the  $x$ -direction. The vortical motion occurs in the  $xz$ -plane, is convected downstream in  $x$ , and extends vertically in the  $y$ -direction. The resulting flow and acoustic characteristics are investigated in this section.

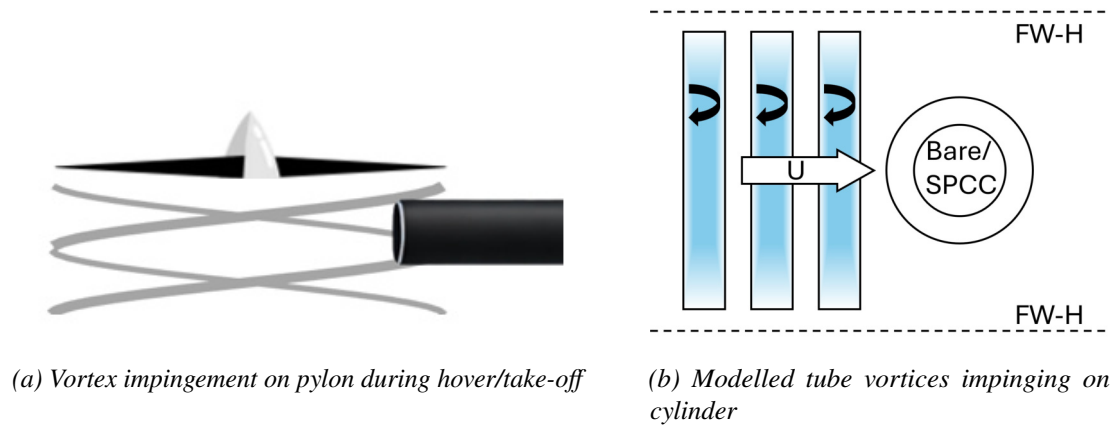


Figure 10.6: Visualisation of vortex impingement on a pylon represented by a cylinder.

### 10.2.1 Methodology

#### 10.2.1.1 Computational Domain

The uncoated (bare) cylinder had a diameter of  $d = 0.032$  m. To create the Darcy–Forchheimer region, a volume zone with a thickness of 0.008 m was placed around the bare cylinder, resulting in an outer diameter of  $D = 0.048$  m for the coated case. The cylinder with the coated porous zone is shown in Figure 10.7.

The computational domain (shown in Figure 10.8) contained approximately 46.5M hexahedral cells, and extended  $10D$  above and below the cylinder centre and  $50D$  in the wake. The span of the domain was  $3.66D$ .

#### 10.2.1.2 Boundary Conditions

At the inlet, the velocity was set to  $U_\infty = 11.65$  m/s, corresponding to the induced velocity of the propeller calculated in Section 10.1. This corresponds to a diameter-based Reynolds number of  $Re_d = 25,000$  for the bare cylinder, and  $Re_D = 37,500$  for the coated cylinder, and a Mach number of  $M = 0.034$ , while a zero-gradient condition was set for pressure. At the outlet, a zero-gradient condition was applied for velocity, while the pressure was set to  $p = 101,325$  Pa. A no-slip condition was applied at the walls, and spanwise periodicity was used. The vortices were injected into the domain every 0.01 s (based on a propeller rotating at 3,000 RPM), and were injected one core radius ( $R$ ) upstream of the obstruction. For the bare case, this was at  $x/d = -1.23$  and for the coated case, this was  $x/D = -0.99$ . This allowed the

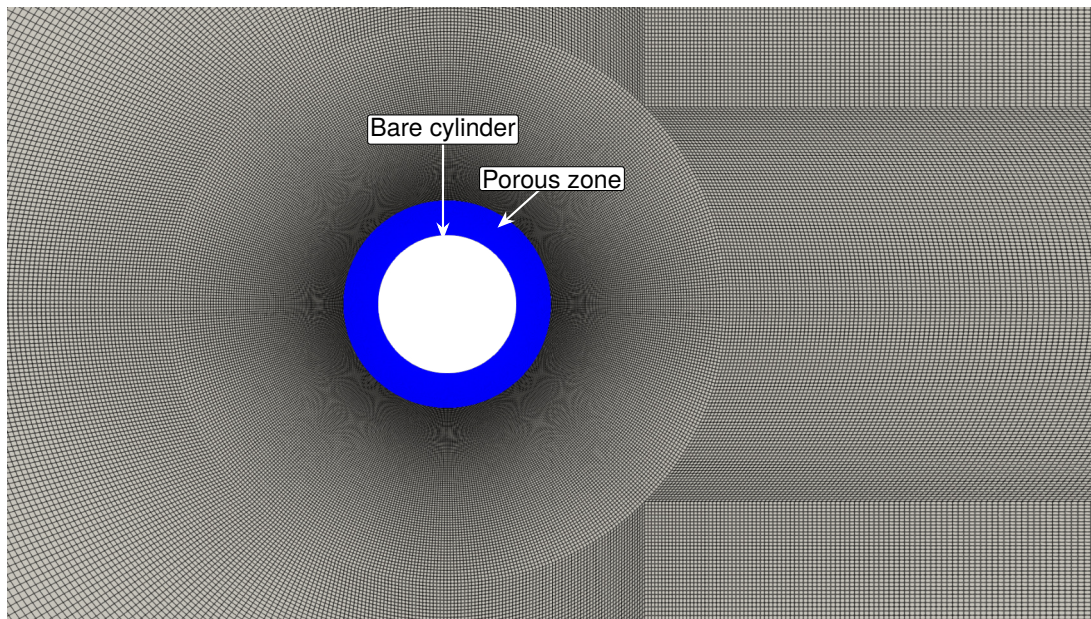


Figure 10.7: Computational domain around the cylinder showing the porous zone

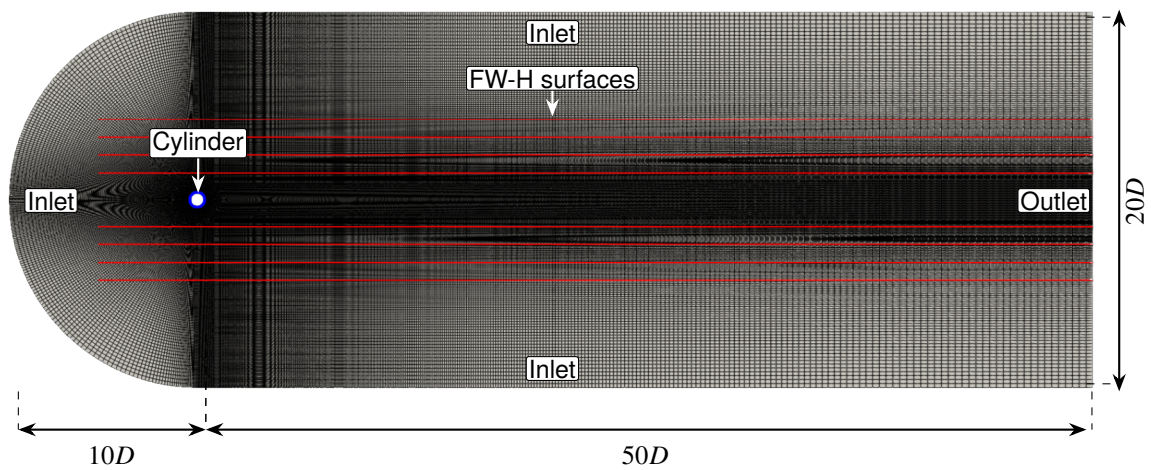


Figure 10.8: Computational domain

vortices to naturally convect and interact with the cylinders before significantly breaking down.

### 10.2.1.3 Acoustics

The acoustics setup was left largely unchanged from that discussed in Section 9.2.1.4, with the only key difference being that no spanwise acoustic corrections were made to the data, since it was not being compared to any experimental data.

### 10.2.1.4 Numerical Methods

The simulations were conducted in OpenFOAM v2212 using the rhoPimpleFoam solver. The turbulence model used was the Zonal LES model implemented in Section 3.3.3. For this study,  $y_{\text{RANS}}^+$  was set to 30 and  $y_{\text{LES}}^+$  was set to 60. In all cases, temporal discretisation was performed using the second-order backward difference scheme. For spatial discretisation, the LUST scheme was employed. The cases were run for  $218T^*$  and  $146T^*$  for the bare and coated cases respectively, where  $T^* = d/U_\infty$  and  $T^* = D/U_\infty$ . Vortices were injected into the domain from  $109T^*$  and  $73T^*$  for the bare and coated cases respectively, with temporal statistics and acoustic data being recorded from  $146T^*$  and  $97T^*$  for the bare and coated cylinders. This meant a total of 30 vortices were injected into the domain over the simulation time. The transient timestep was  $7.28 \times 10^{-4}T^*$  and  $4.85 \times 10^{-4}T^*$  for the bare and coated cases respectively.

## 10.2.2 Results

Figure 10.9 shows normalised Q-criterion iso-surfaces coloured by mean velocity magnitude, where  $Q_{\text{norm}} = Q \times D^2/U_\infty^2$ . The value of  $Q_{\text{norm}}$  was chosen to be 5, but to be able to visualise the inflow vortices, the value of  $Q_{\text{norm}}$  was set to 0.1 for the vortices. Figure 10.9a shows that the vortex appears to wrap around the bare cylinder, whereas in Figure 10.9b, the vortex appears to pass over the porous coating. There is also a considerably longer region of low velocity magnitude in the wake of the coated cylinder compared to the bare cylinder.

Figure 10.10 shows contours of  $x - z$  vorticity. From the figure, the vortex interacting with the bare (Figure 10.10a) and coated (Figure 10.10b) cylinders can be seen. The wake of the bare cylinder exhibits a stronger  $\omega_z$  in the wake, indicating stronger rotation from the vortex persisting downstream, whilst this vorticity is noticeably diffused in the wake of the coated case.

### 10.2.2.1 Mean Velocity Wake Profiles

Figure 10.11 shows the contours of normalised streamwise mean velocity, comparing the bare (Figure 10.11a) and coated (Figure 10.11b) cylinders in uniform flow. Although at a much lower Reynolds number, the cylinders exhibit very similar characteristics to those discussed in Section 9.2.2.1. For the bare cylinder, there is a pronounced velocity deficit in the near-wake region, forming a relatively short and narrow low-momentum core with evidence of mean reversed flow. The velocity gradients along the shear layers are steep, indicating a strong momentum exchange with the surrounding flow. The coated case, on the other hand, exhibits a broader wake with the low-velocity region extending further downstream. The shear layers appear to develop more gradually, and the wake recovery occurs over a longer streamwise distance, suggesting a redistribution of mean momentum compared to the bare cylinder.

Figure 10.12 shows the contours of normalised streamwise mean velocity, comparing the bare (Figure 10.12a) and coated (Figure 10.12b) cylinders with the impinging vortices. The cylinders with the inflow



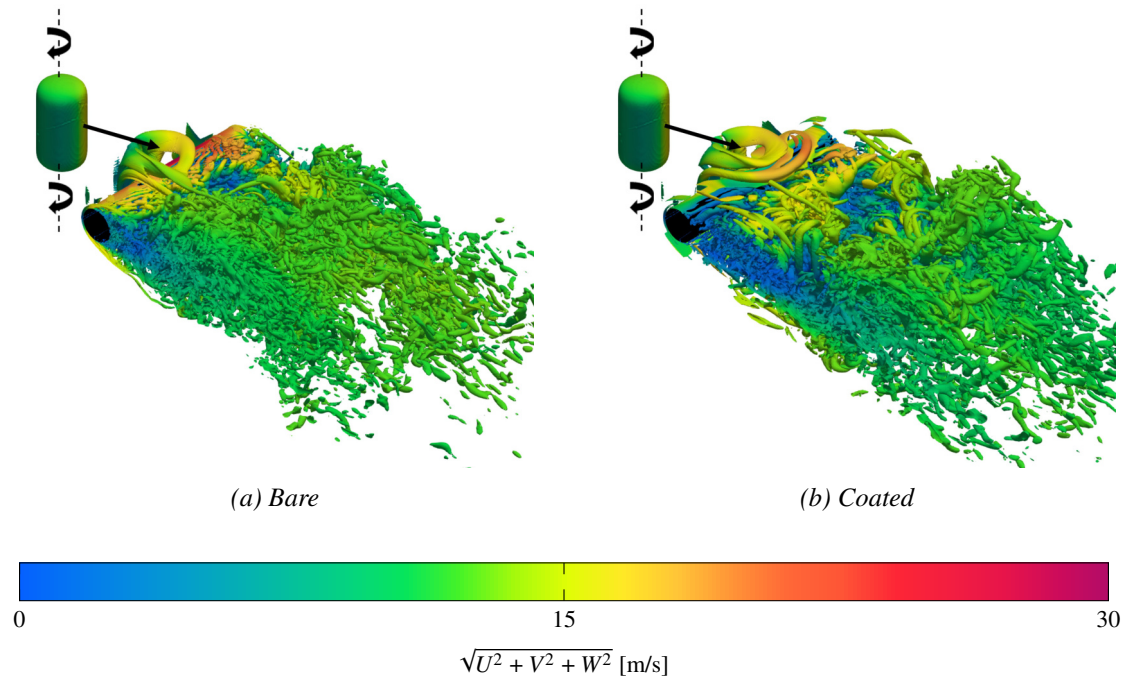


Figure 10.9: Normalised  $Q$ -criterion iso-surfaces coloured by mean velocity magnitude,  $Q_{norm} = 5$  (Inflow vortices shown at  $Q_{norm} = 0.1$ ) comparing the bare and coated cylinders.

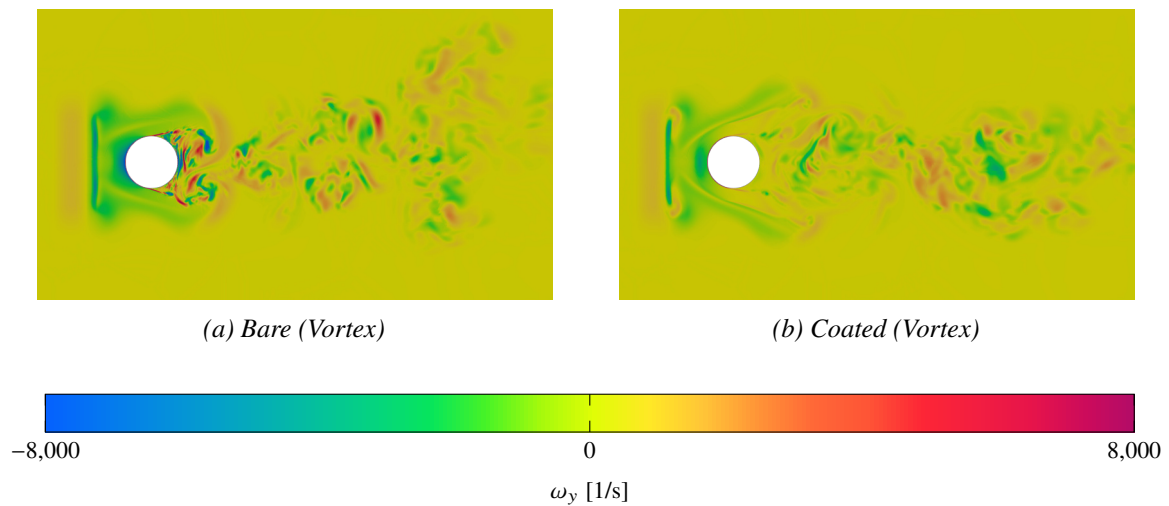


Figure 10.10: Instantaneous wall-normal vorticity comparing the bare and coated cylinders with impinging vortex.

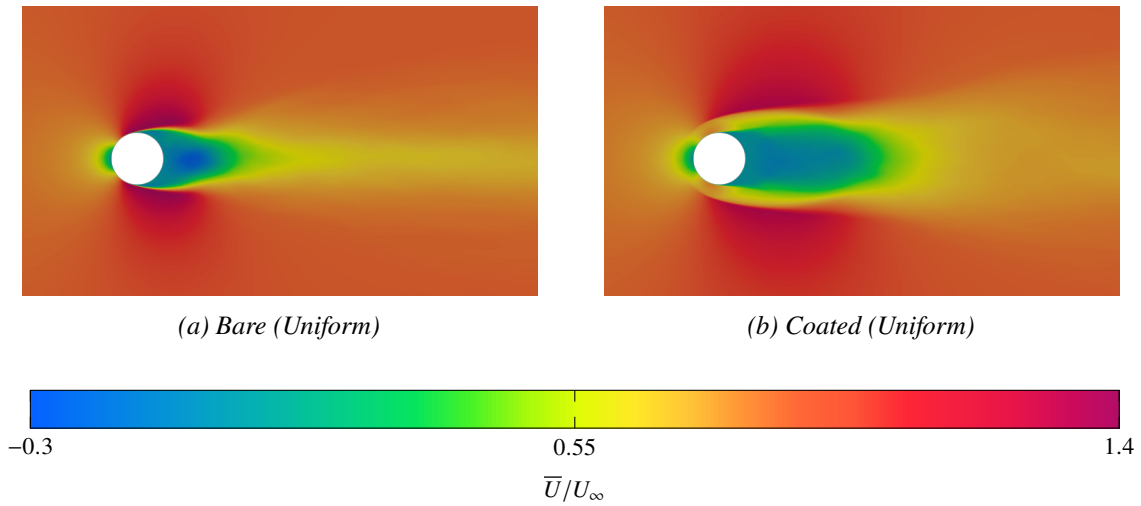


Figure 10.11: Comparison of normalised streamwise mean velocity between the bare and coated cylinders in uniform flow

vortices show qualitatively similar characteristics to the cylinders in uniform flow (Figure 10.11). For the bare cylinder, there is a pronounced velocity deficit in the near-wake region, but it appears to be narrower than the bare cylinder in uniform flow. There is also evidence of mean reversed flow. The velocity gradients along the shear layers are steeper than those from the outer diameter of the coated cylinder. The coated case also exhibits a broader and longer low-velocity wake than the bare cylinder with inflow vortices, although narrower than its uniform flow counterpart.

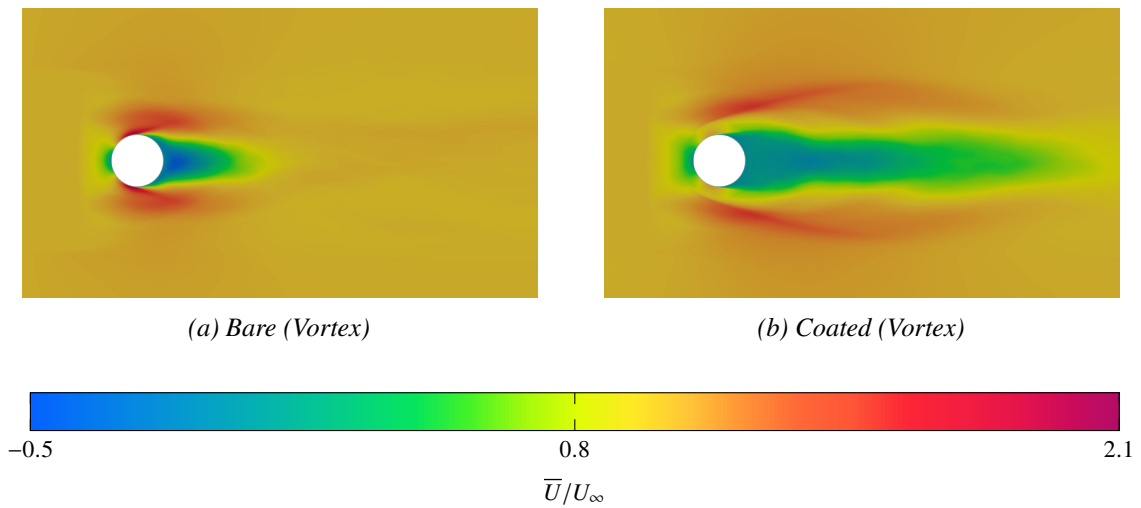


Figure 10.12: Comparison of normalised streamwise mean velocity between the bare and coated cylinders with impinging vortex

Figure 10.13a shows that at  $x/D = 0.75$ , the bare cylinders both in uniform flow and with inflow vortices exhibit a deeper wake deficit than their coated counterparts, although the deficit is slightly wider for the coated cases.

At both  $x/D = 1.5$  (Figure 10.13b) and  $x/D = 3.0$  (Figure 10.13c), the coated cylinder shows a deeper and wider wake deficit both in uniform flow and with inflow vortices compared to their bare counterparts.

Further downstream at  $x/D = 7.0$  (Figure 10.13d), the coated cylinder still exhibits a deeper wake deficit than the bare cylinder in uniform flow, although the difference is not as significant as the previous two upstream locations. The coated cylinder also exhibits a deeper and wider wake deficit than the bare cylinder for the cases with inflow vortices.

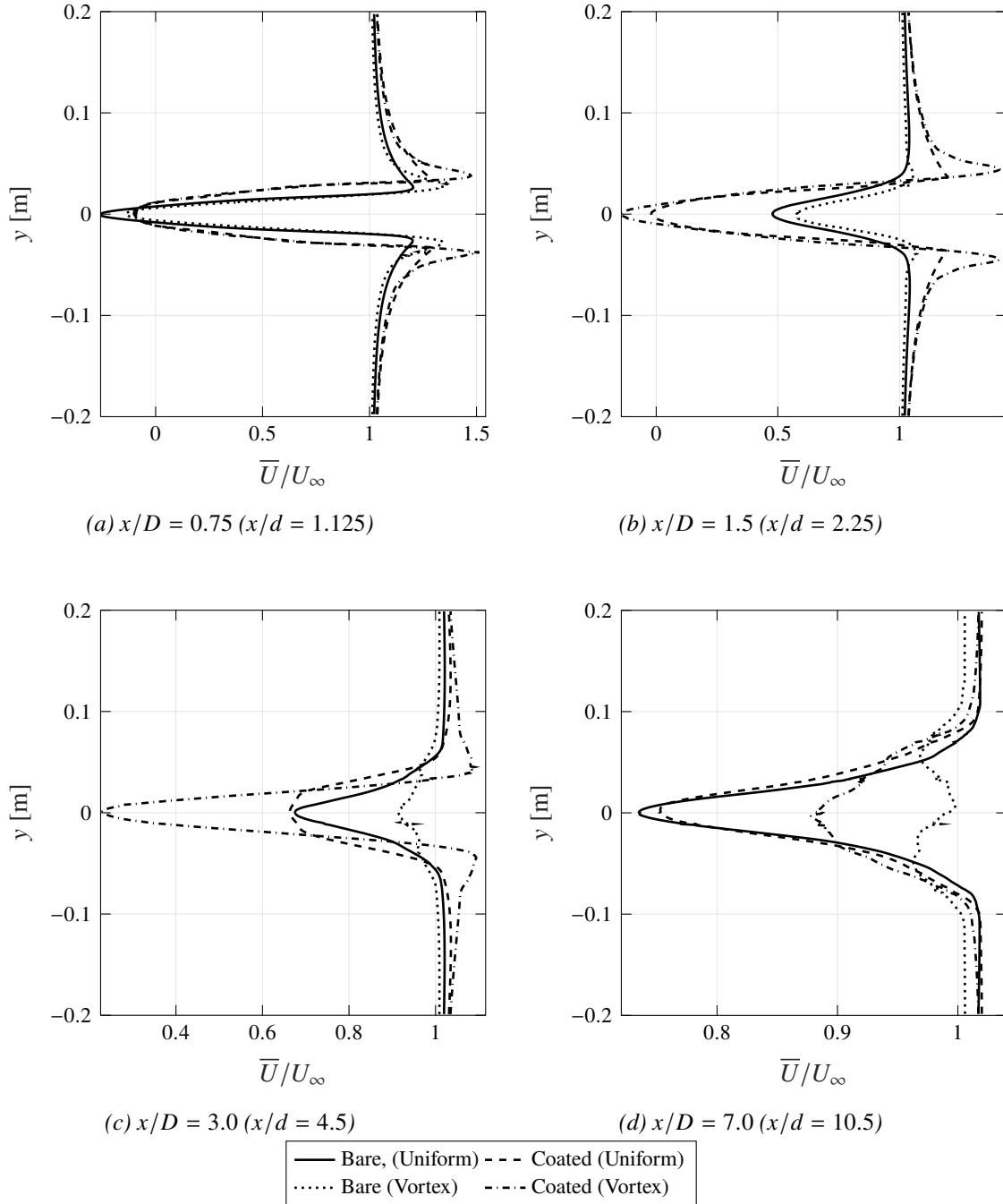


Figure 10.13: Normalised streamwise mean velocity.

### 10.2.2.2 Turbulence Intensity Wake Profiles

Figure 10.14 shows contours of turbulence intensity for the bare cylinder (Figure 10.14a) and the coated cylinder (Figure 10.14b) in uniform flow. For the bare cylinder, elevated regions of turbulence occur immediately downstream of the cylinder, forming two symmetric high-energy regions associated with shear



layer roll-up. For the coated cylinder, the regions of elevated turbulence are shifted further downstream. Although these high-energy regions are broader than that of the bare cylinder, their peak values are reduced, suggesting a delayed growth of velocity fluctuations in the near-wake.

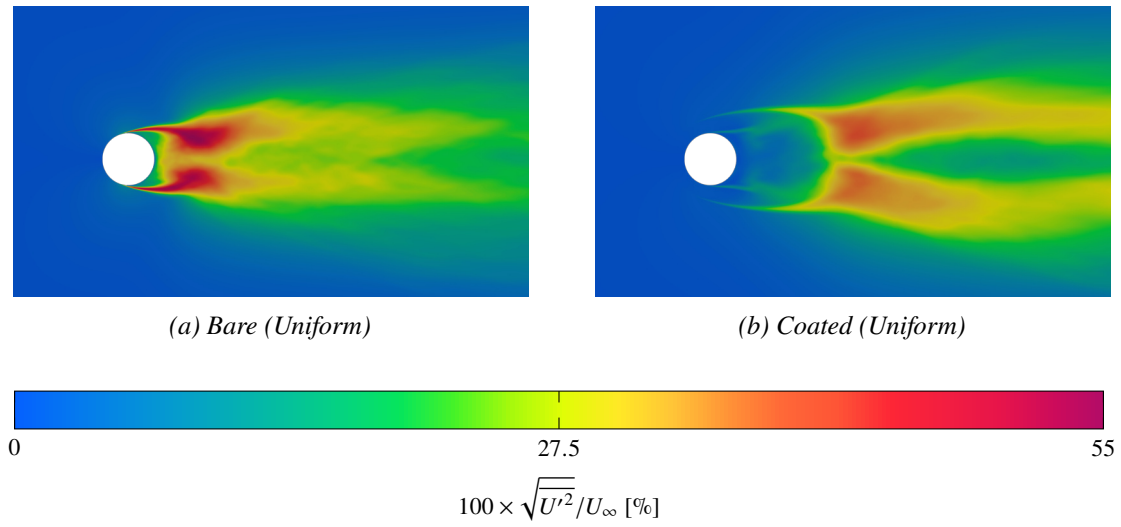


Figure 10.14: Comparison of turbulence intensity between the bare and coated cylinders in uniform flow

Figure 10.15 shows contours of turbulence intensity for the bare cylinder (Figure 10.15a) and the coated cylinder (Figure 10.15b) with inflow vortices. For the bare cylinder, elevated regions of high turbulence intensity appear to be clustered in the near-wake region of the cylinder, with two strong peaks noticeable over the cylinder surface. For the coated cylinder, the turbulence intensity appears to be much more diffused. There is also evidence of a shear layer emanating from the outer diameter of the coating.

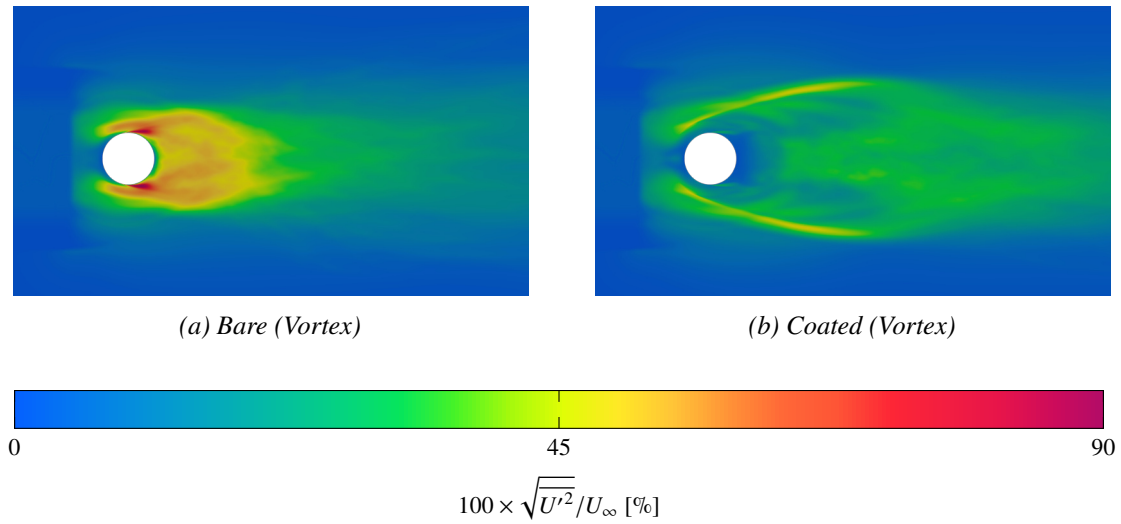


Figure 10.15: Comparison of turbulence intensity between the bare and coated cylinders with impinging vortex

At  $x/D = 0.75$ , Figure 10.16a shows the coated cylinder significantly decreasing the turbulence intensity peaks for both the uniform flow and inflow vortices cases.

At  $x/D = 1.5$ , Figure 10.16b shows that in uniform flow, the coated cylinder once again significantly

reduces the turbulence intensity compared to the bare cylinder. For the cases with inflow vortices, the coated cylinder exhibits a wider peak than the bare cylinder, although at a lower magnitude.

At the two locations further downstream ( $x/D = 3.0$  and  $x/D = 7.0$ ), Figures 10.16c and 10.16d show that the coated cylinder in both uniform flow and with inflow vortices exhibits a stronger and wider turbulence intensity profile compared to the bare cylinder.

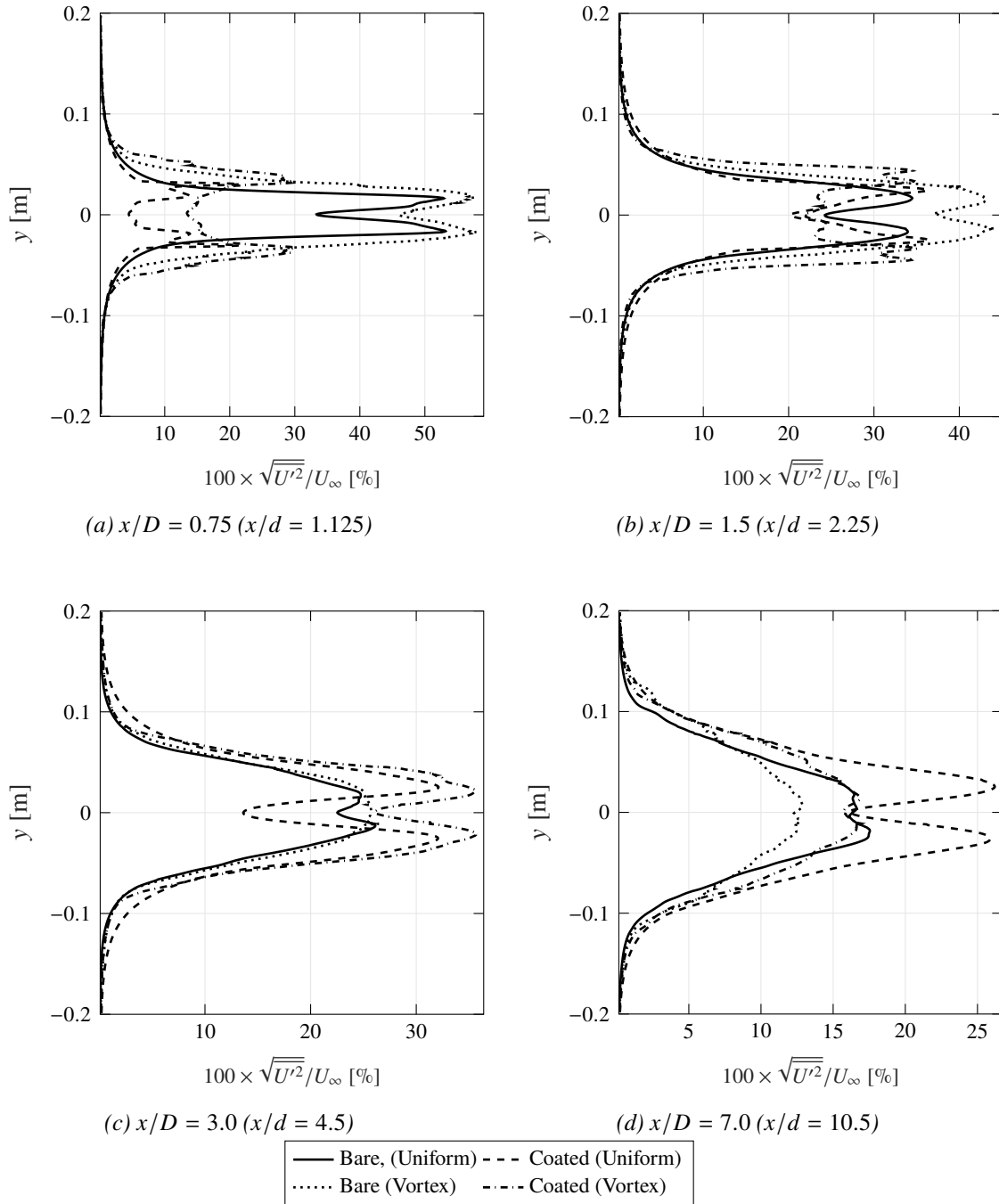


Figure 10.16: Turbulence Intensity

### 10.2.2.3 Acoustic Power Spectral Density

Figure 10.17 shows the acoustic power spectral density ( $\phi_{pp}$ ) at different polar angles and the OASPL.

For the uniform flow cases, at  $\theta = 150^\circ$ ,  $\theta = 90^\circ$  and  $\theta = 30^\circ$ , Figures 10.17a, 10.17b and 10.17c all show that the coated cylinder significantly reduces the  $\phi_{pp}$  between approximately 200 Hz and 700 Hz compared to the bare cylinder, and then shows a slight increase in  $\phi_{pp}$  between approximately 1,000 Hz and 2,000 Hz. The OASPL shown in Figure 10.17d shows that the coated cylinder exhibits a noise reduction at all polar angles with a maximum reduction of 10 dB achieved at the  $\theta = 90^\circ$  polar angle.

For the cases with inflow vortices, all polar angles show distinct tones every 100 Hz between 200 Hz and 600 Hz. These tones appear to be associated with the frequency of vortex injection which was also 100 Hz. It appears that the coated case exhibits a higher  $\phi_{pp}$  than the bare case for the first visible tone at  $\theta = 150^\circ$  and  $\theta = 30^\circ$ , but reduces the  $\phi_{pp}$  of the tones at the higher frequencies. Beyond 200 Hz, the coated case  $\phi_{pp}$  spectrum remains below that of the bare case for all polar angles. The coated case exhibits a higher OASPL at the  $\theta = 30^\circ$  polar angle than the bare case, showing an approximate 3.1 dB increase. A maximum reduction of approximately 5.8 dB was achieved by the coated case at the  $\theta = 90^\circ$  polar angle.

#### 10.2.2.4 Velocity Power Spectral Density

The probe locations used for the velocity power spectral density ( $\phi_{uu}$ ) are the same as those shown in Figure 9.12.

With respect to the uniform-flow cases, the coated cylinder exhibits a noticeable reduction in the spectral energy of the velocity fluctuations across the frequency range at all probe locations (Figure 10.18). For the cases exposed to the impinging vortex, however, the spectral energy shows a stronger dependence on probe location, particularly on the  $y/D$  position. At both  $x/D = 1$  and  $x/D = 2$ , the coated cylinder generally exhibits reduced spectral energy across the frequency range compared with the bare cylinder at  $y/D = 0.33$  (Figures 10.18a and 10.18e) and  $y/D = 0.5$  (Figures 10.18b and 10.18f). At  $y/D = 0.67$  (Figures 10.18c and 10.18g), the coated cylinder exhibits increased low-frequency content relative to the bare cylinder, while still showing reduced high-frequency content. At  $y/D = 1$  (Figures 10.18d and 10.18h), the coated cylinder exhibits increased spectral energy over most of the frequency range, relative to the bare cylinder, up to approximately 10,000 Hz.

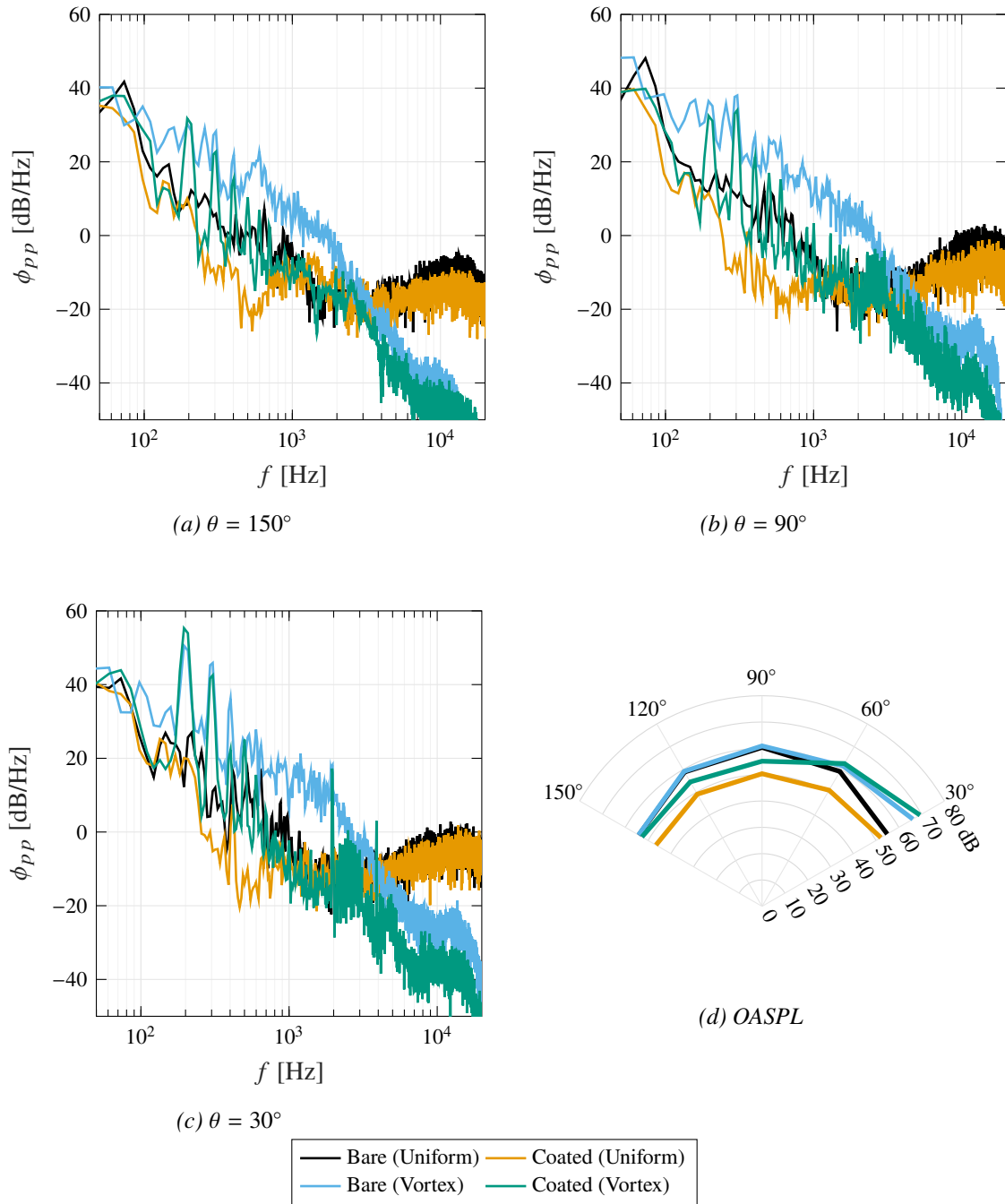


Figure 10.17: Acoustic PSD ( $\phi_{pp}$ ) [dB/Hz] as a function of frequency  $f$  [Hz]

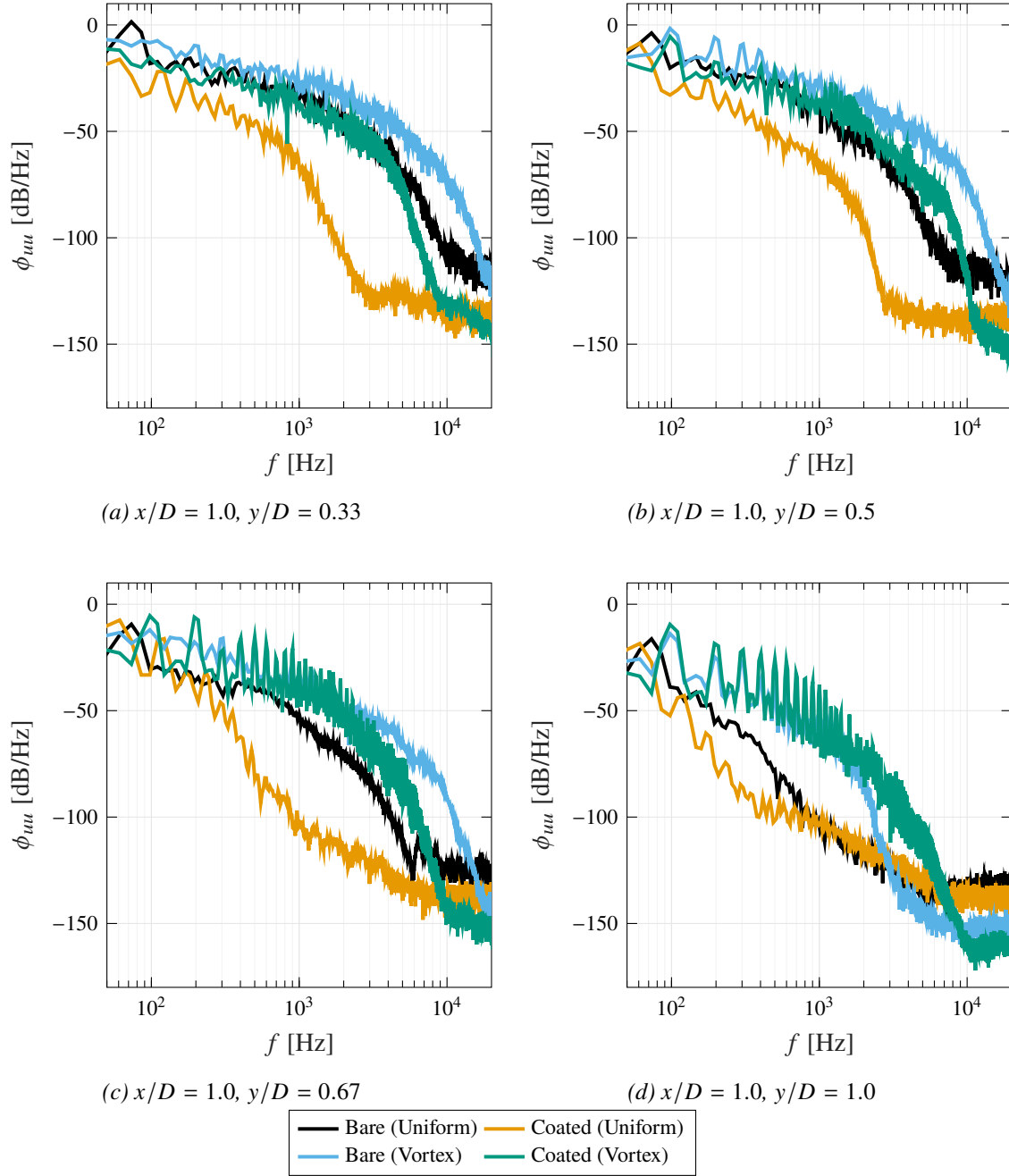


Figure 10.18: Velocity PSD ( $\phi_{uu}$ ) [dB/Hz] as a function of frequency  $f$  [Hz] (continued on next page).

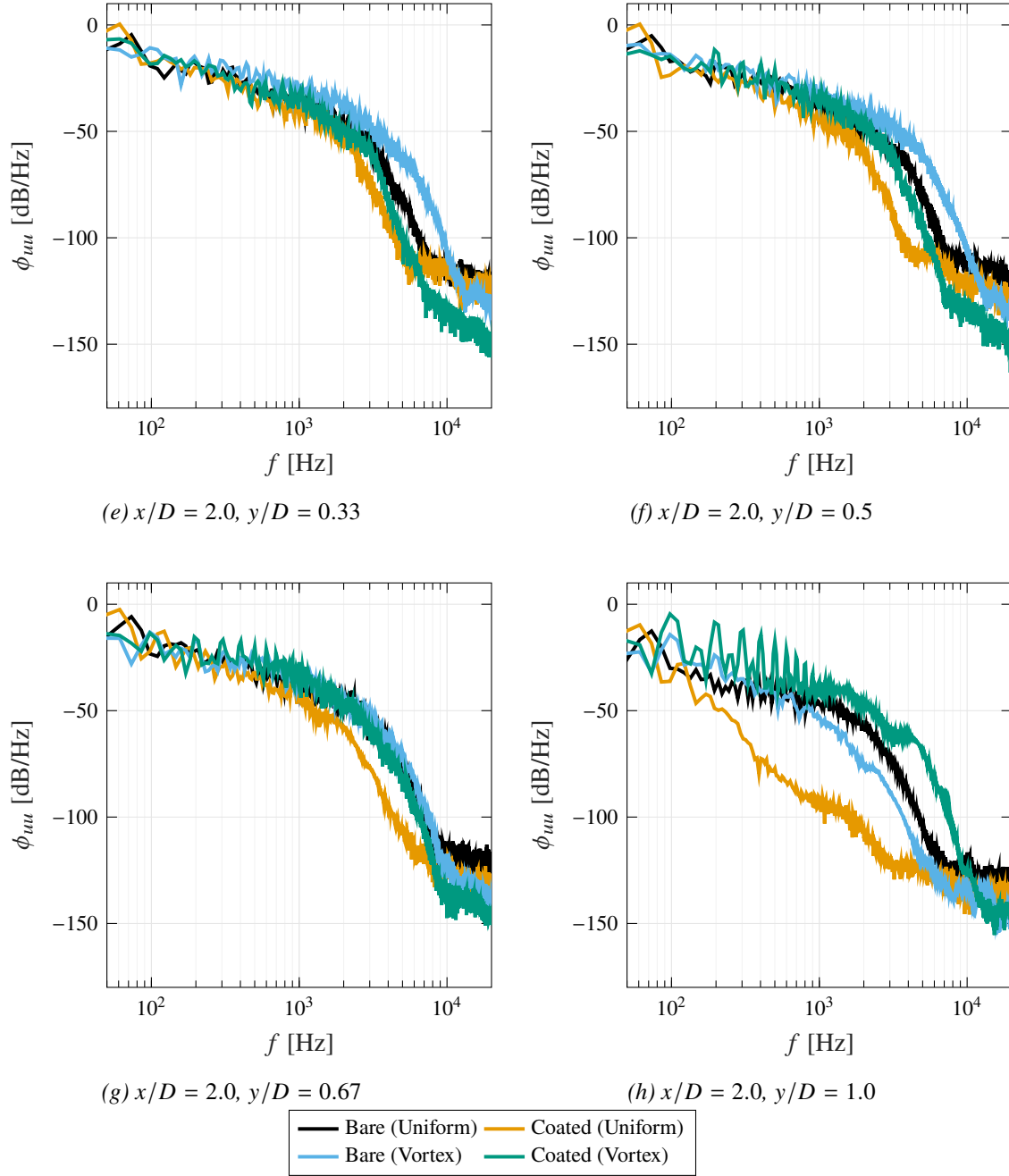


Figure 10.18: Velocity PSD ( $\phi_{uu}$ ) [dB/Hz] as a function of frequency  $f$  [Hz] (continued).

### 10.3 Vortex Impingement on Airfoil Leading Edge

For EUAMVs operating in the forward flight phases, vortices shed from the propeller tips can interact with the wing leading edge, producing installation noise. A schematic of this interaction is shown in Figure 10.19a. To assess the extent of this noise, as well as the effect of applying a porous coating to the leading edge of the wing, a series of tube vortices were introduced into the computational domain and allowed to impinge on the wing leading edge, which was modelled as an airfoil, as shown in Figure 10.19b. The airfoil span is oriented in the  $z$ -direction, with the inflow in the  $x$ -direction. The vortical motion occurs in the  $xz$ -plane, is convected downstream in  $x$ , and extends vertically in the  $y$ -direction.

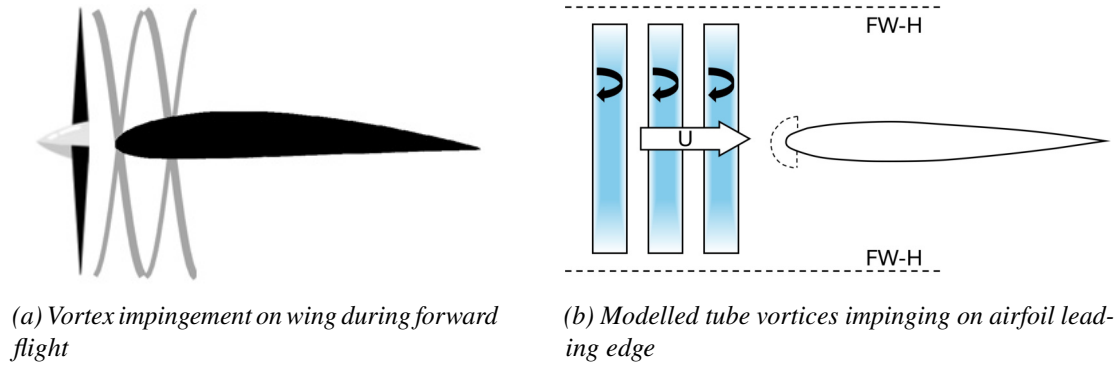


Figure 10.19: Visualisation of vortex impingement on a wing represented by an airfoil.

Although the porous treatment is applied over the full span in the current study, this is primarily a consequence of the short-span periodic domain used in the simulations. In practice, the porous region is proposed to be localised to the vicinity of the tip-vortex interaction rather than applied across the entire wing. This localisation would minimise, or potentially eliminate, any aerodynamic penalty associated with porosity. The porous treatment could also be implemented using an adaptive-control approach, allowing it to be selectively activated only when required for vortex-interaction mitigation. The resulting flow and acoustic characteristics are investigated in this chapter.

#### 10.3.1 Methodology

##### 10.3.1.1 Computational Domain

The computational domain around the airfoil is shown in Figure 10.20. Based on the equation of a sharp trailing edge NACA 0012 airfoil shown in Equation 10.10, the point along the chord ( $y = 0$ ) where the distance to the upper and lower airfoil surfaces was equal to the distance to the leading was found to be  $x/C \approx 0.02715$ . The porous zone was therefore wrapped around the leading edge from this  $x/C$  location, with a thickness of 0.008 m to match the thickness of the coatings investigated in Chapters 9.2 and 10.2. The coated region is also shown in Figure 10.20.

$$y_t = 0.6c \left( 0.2969 \sqrt{\frac{x}{c}} - 0.1260 \frac{x}{c} - 0.3516 \left( \frac{x}{c} \right)^2 + 0.2843 \left( \frac{x}{c} \right)^3 - 0.1036 \left( \frac{x}{c} \right)^4 \right) \quad (10.10)$$

The computational domain (shown in Figure 10.21) contained approximately 119M hexahedral cells, and extended  $10C$  above, below and upstream of the airfoil and  $20C$  in the wake. The span of the domain was



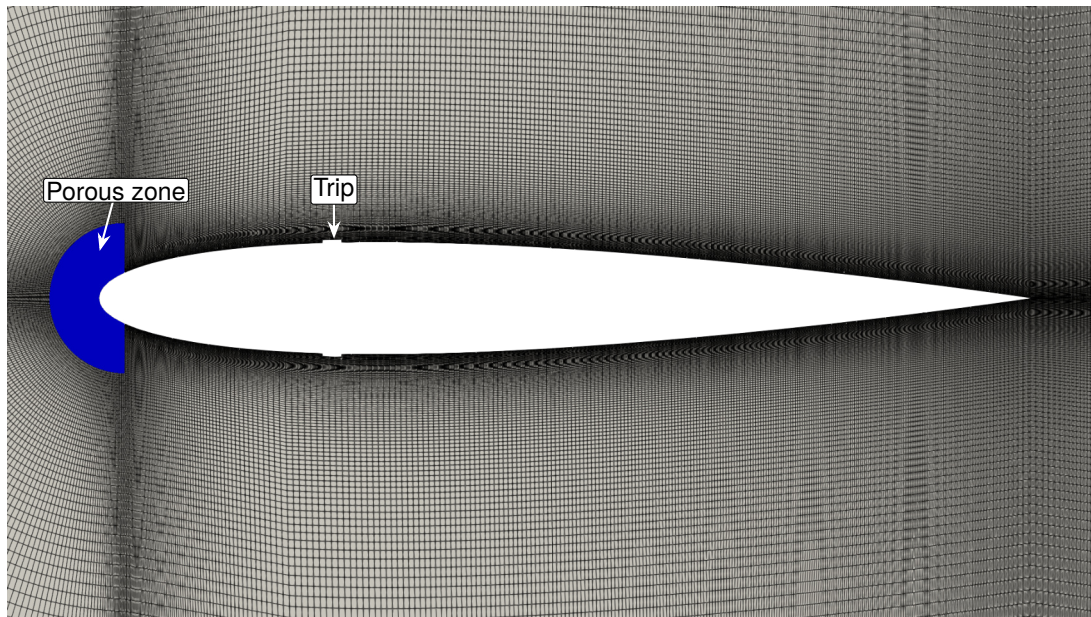


Figure 10.20: Computational domain around the airfoil showing the porous zone

1.15C.

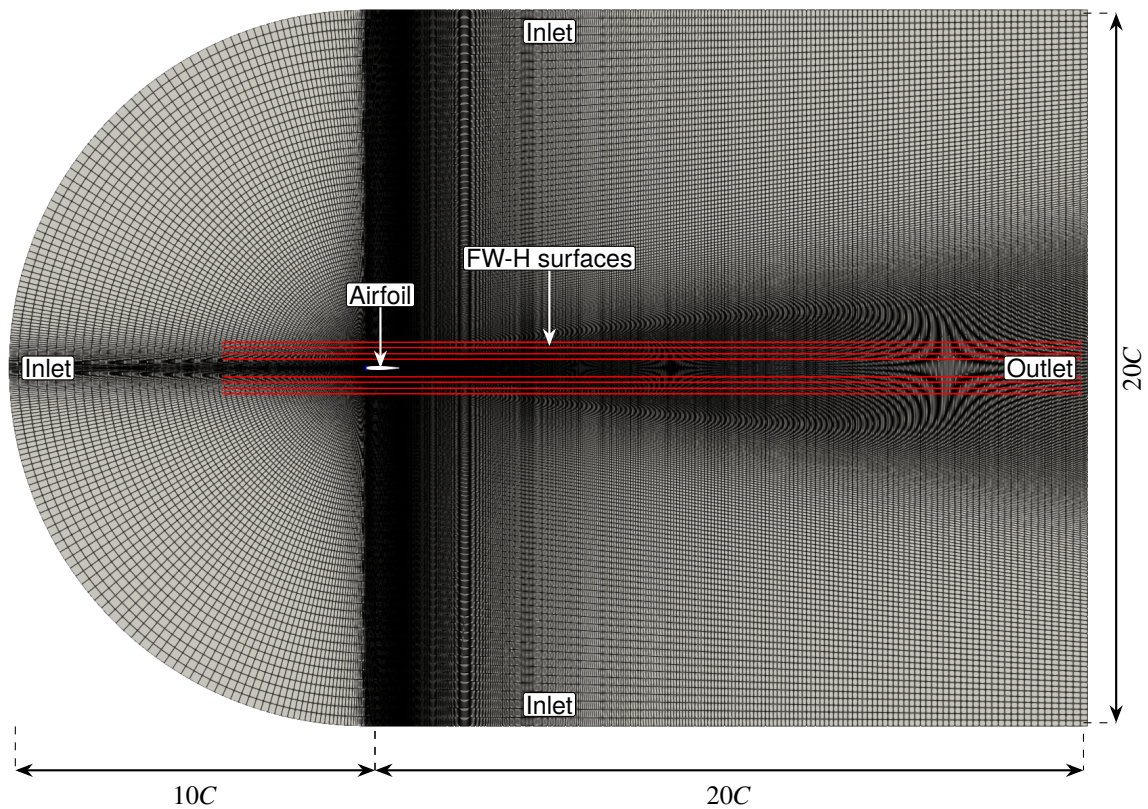


Figure 10.21: Computational domain

### 10.3.1.2 Boundary Conditions

At the inlet, the velocity was set to  $U_{\infty} = 11.65$  m/s, corresponding to the induced velocity of the propeller calculated in Section 10.1. This corresponds to a chord-based Reynolds number of  $Re_C = 117,000$ , and



a Mach number of  $M = 0.034$ , while a zero-gradient condition was set for pressure. At the outlet, a zero-gradient condition was applied for velocity, while the pressure was set to  $p = 101,325$  Pa. A no-slip condition was applied at the walls, and spanwise periodicity was used. The vortices were injected into the domain every 0.01 s (based on a propeller rotating at 3,000 RPM), and were injected one core radius ( $R$ ) upstream of the obstruction. For the bare airfoil, this was at  $x/C = -0.15$  and for the coated case, this was  $x/C = -0.21$ . This allowed the vortices to naturally convect and interact with the airfoil leading edge before significantly breaking down.

### 10.3.1.3 Acoustics

The permeable FW-H surfaces had a uniform resolution of  $0.027C$ , and extended  $4C$  upstream of the leading edge and to the end of the domain downstream. The closest FW-H surface was placed at  $y/C = 0.24$  with every surface separated by a distance of  $y/C = 0.16$ .

The microphone array was an arc with radius  $6.67C$ , with microphones arranged at  $30^\circ$  intervals between  $\theta = 30^\circ$  and  $\theta = 150^\circ$ .

### 10.3.1.4 Numerical Methods

The simulations were conducted in OpenFOAM v2212 using the rhoPimpleFoam solver. The turbulence model used was the Zonal LES model implemented in Section 3.3.3. For this study,  $y_{\text{RANS}}^+$  was set to 30 and  $y_{\text{LES}}^+$  was set to 60. In all cases, temporal discretisation was performed using the second-order backward difference scheme. For spatial discretisation, the LUST scheme was employed. The cases were run for  $46.6T^*$ , where  $T^* = C/U_\infty$ . Vortices were injected into the domain from  $23.3T^*$ , with temporal statistics and acoustic data being recorded from  $31T^*$ . This meant a total of 30 vortices were injected into the domain over the simulation time. The transient timestep for both cases was  $1.55 \times 10^{-4}T^*$ .

## 10.3.2 Results

Figure 10.22 shows normalised Q-criterion iso-surfaces coloured by mean velocity magnitude, where  $Q_{\text{norm}} = Q \times C^2/U_\infty^2$  comparing the bare airfoil leading edge (Figure 10.22a) with the coated airfoil leading edge (Figure 10.22b). The value of  $Q_{\text{norm}}$  was chosen to be 20. From the figure, persistent vortical structures can be seen over the airfoil surfaces in both cases. However, these structures appear closer to the airfoil surface for the bare airfoil, and further above the airfoil surface for the coated airfoil (associated with the vortex passing over the porous coating). As the vortex convects over the airfoil surface, there appears to be horseshoe-like structures forming around the convected tip-vortex. There is also a noticeably lower velocity magnitude in the wake of the coated airfoil compared to the bare airfoil.

Figure 10.23 shows contours of instantaneous  $\omega_y$ . The figure shows that the incoming vortex leaves a clear downstream imprint on the flow aft of the trailing edge for both the bare (Figure 10.23a) and coated (Figure 10.23b) airfoils. For the bare airfoil case, the vortex remains relatively coherent after leading-edge interaction, and the resulting disturbances convect downstream with less attenuation compared to the coated airfoil. The coated airfoil appears to diffuse the vortex, resulting in the imprint at the trailing edge being weaker.

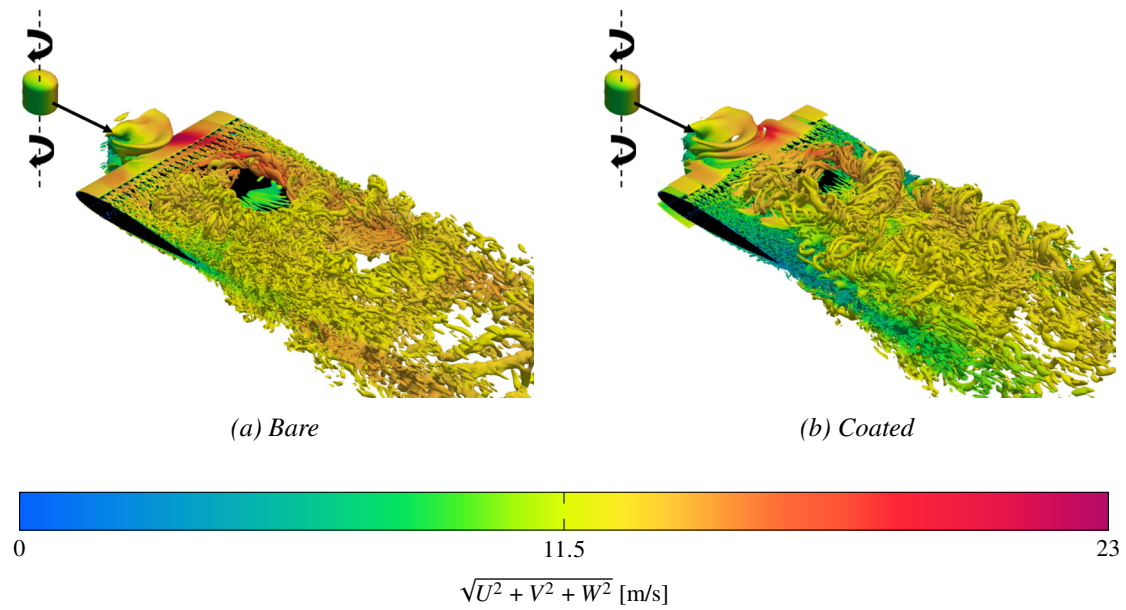


Figure 10.22:  $Q$ -criterion iso-surfaces coloured by mean velocity magnitude,  $Q_{norm} = 20$ , for the bare and coated airfoil leading edges.

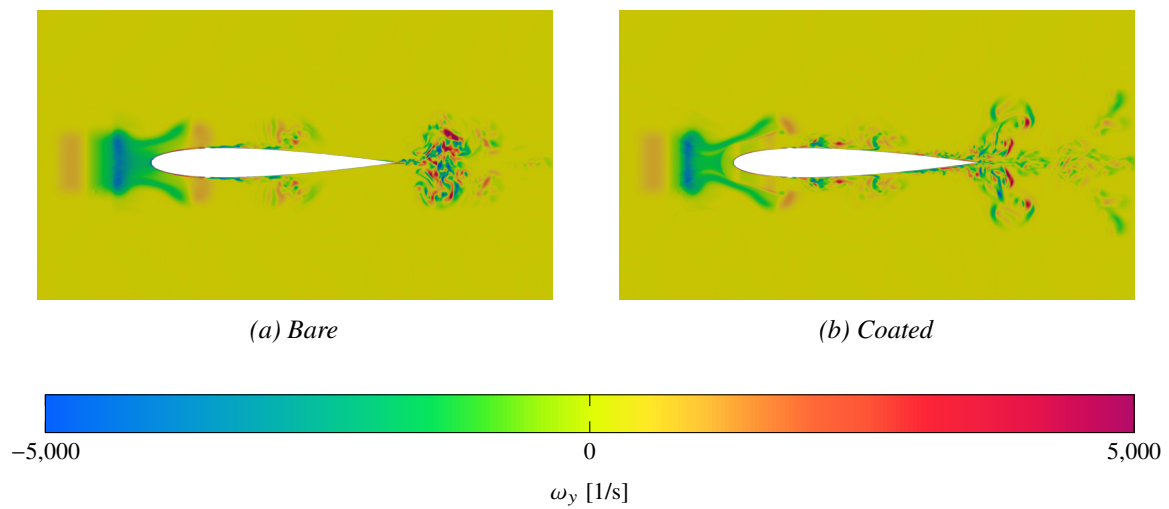


Figure 10.23: Instantaneous wall-normal vorticity comparing the bare and coated airfoil leading edges with impinging vortex.

### 10.3.2.1 Mean Velocity Profiles

Figure 10.24 shows the contours of normalised streamwise mean velocity for the bare (Figure 10.24a) and coated (Figure 10.24b) airfoils. For the bare airfoil, the wake remains relatively narrow compared to the coated airfoil, with a pronounced velocity deficit confined close to the centreline. The coated airfoil produces a noticeable shear layer emanating from the outer diameter of the coating. This shear layer appears to create a visibly thicker wake with a more distributed velocity deficit.

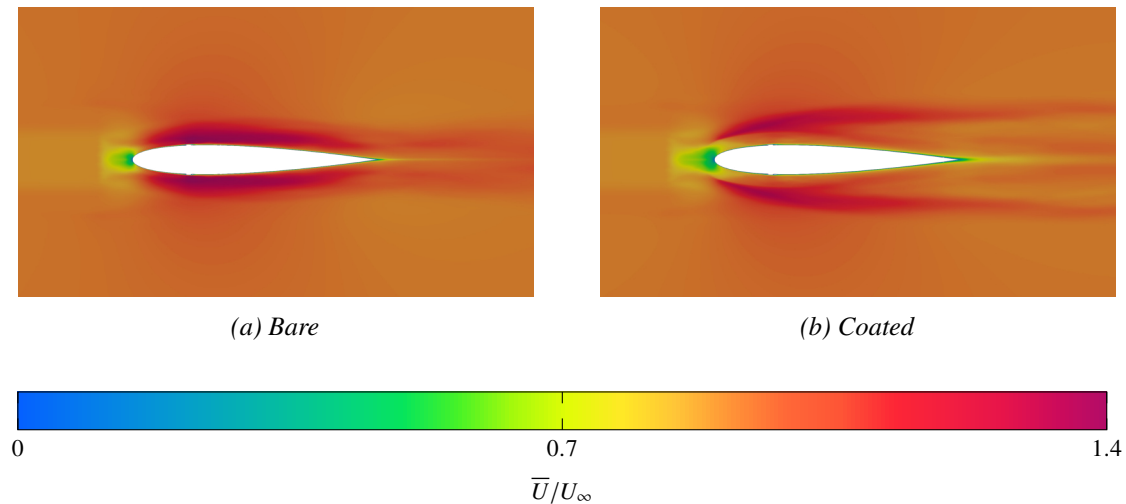


Figure 10.24: Comparison of normalised streamwise mean velocity contours between the bare and coated airfoils with impinging vortex.

Figure 10.25 shows the normalised streamwise mean velocity profiles at various locations.

At the boundary layer locations (Figures 10.25a–10.25d), a general trend can be observed where the coated airfoil exhibits a sharp increase in velocity, with the peak velocity occurring further away from the airfoil surface compared to the uncoated airfoil. On the other hand, the uncoated airfoil exhibits a much flatter boundary layer profile at all boundary layer locations, although it reaches a higher peak velocity compared to the coated airfoil. The peaks occurring at higher  $y/C$  locations are associated with flow over the coating, whilst the lower peak values indicate that the coating slows down the flow close to the airfoil surface.

At both wake locations ( $x/C = 1.1$  and  $x/C = 1.2$ ), Figures 10.25e and 10.25f show that the coated airfoil exhibits a significantly wider and deeper wake deficit compared to the bare airfoil. This agrees well with the outcomes of the cylinder investigated in Section 10.2. In the case of the cylinders, the noise reduction was attributed to the momentum deficit in the wake. Hence, it is expected that a similar mechanism is responsible for the airfoil case.

### 10.3.2.2 Turbulence Intensity Profiles

The contours of turbulence intensity are shown in Figure 10.26. For the uncoated airfoil (Figure 10.26a), there is a high turbulence intensity over the airfoil surface aft of the leading edge. This strong turbulence intensity appears to be diffused by the porous coating (Figure 10.26b), which shows a much lower turbulence intensity in the same vicinity. However, the shear layer generated by the porous coating is clearly visible.

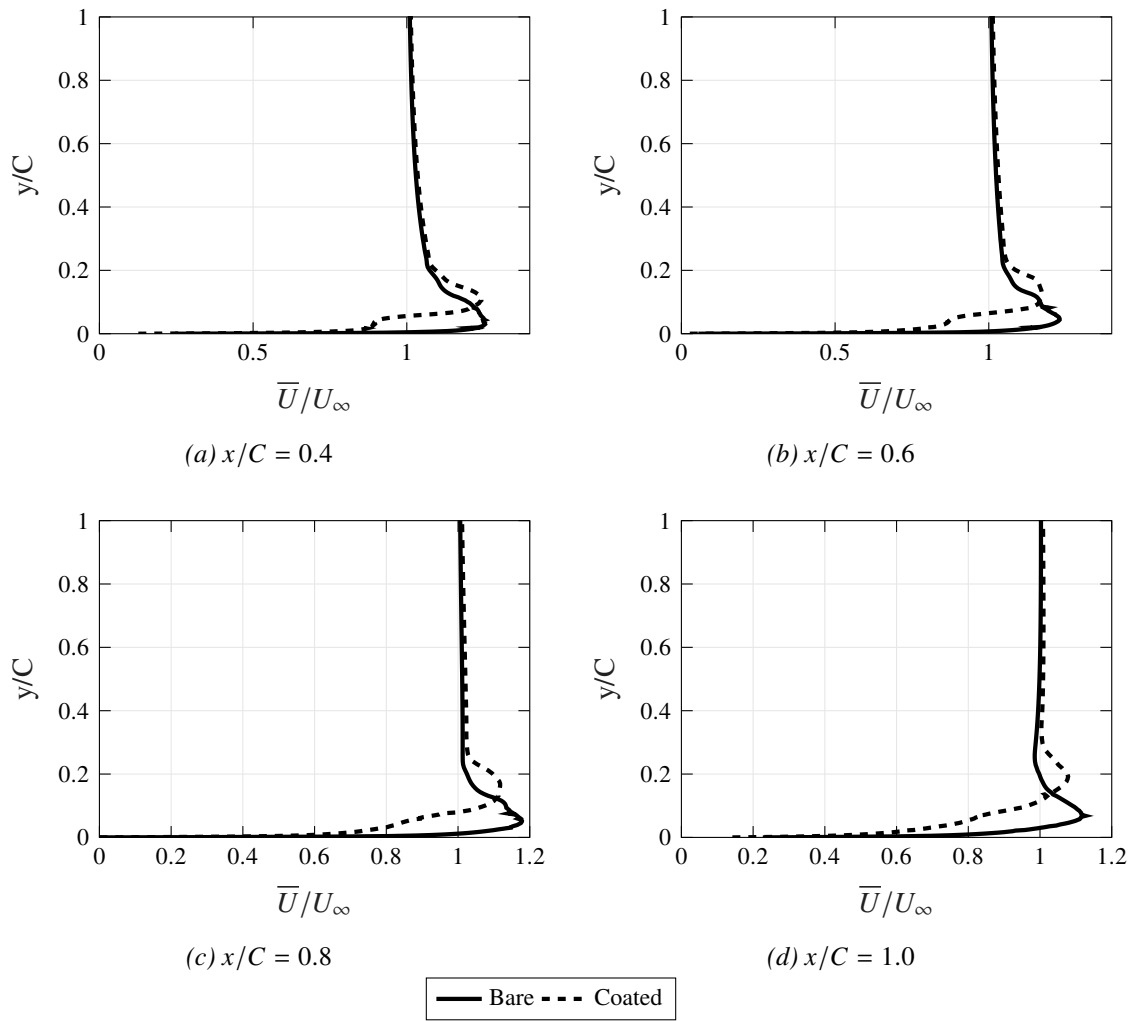


Figure 10.25: Comparison of normalised streamwise mean velocity profiles between the bare and coated airfoils with impinging vortex (continued on next page).

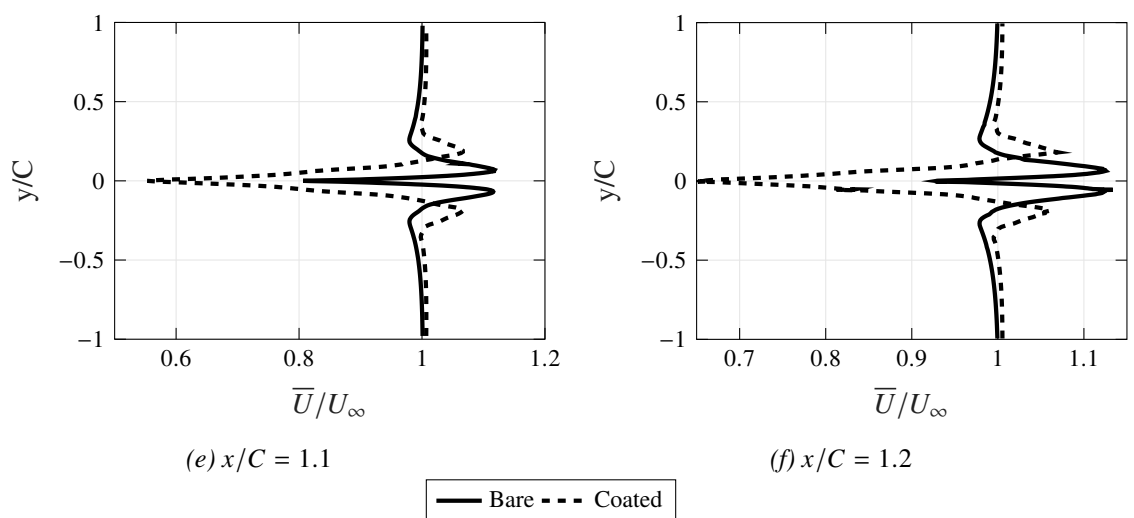


Figure 10.25: Comparison of normalised streamwise mean velocity profiles between the bare and coated airfoils with impinging vortex (continued).

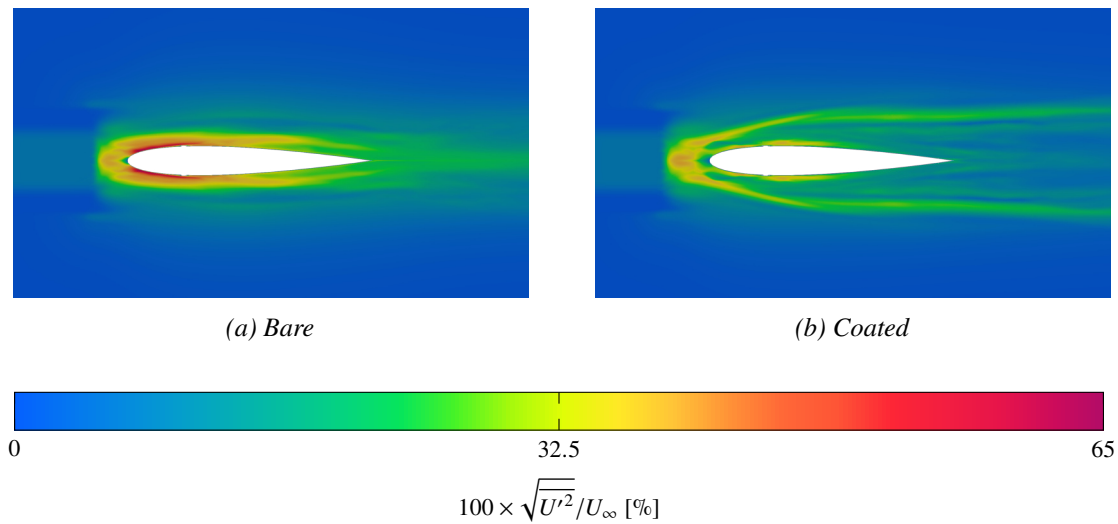


Figure 10.26: Comparison of turbulence intensity contours between the bare and coated airfoils with impinging vortex

Figure 10.27 shows the turbulence intensity profiles at various locations. For the boundary layer locations (Figures 10.27a–10.27d), a general trend is observed where the coated case exhibits a slightly wider peak but produces a significant reduction in the magnitude of the peak turbulence intensity compared to the bare case.

A similar trend is noticed in the wake (Figures 10.27e, 10.27f), where the coated airfoil exhibits a slightly wider but significantly reduced magnitude peak of turbulence intensity compared to the bare airfoil.

### 10.3.2.3 Acoustic Power Spectral Density

Figure 10.28 shows the acoustic power spectral density ( $\phi_{pp}$ ) at different polar angles and the OASPL. At  $\theta = 150^\circ$  and  $\theta = 30^\circ$ , Figures 10.28a and 10.28c show distinct tones at 100, 200, 300 and 400 Hz. At  $\theta = 90^\circ$ , Figure 10.28b only clearly shows the 100 Hz and 200 Hz tones. These tones coincide with the vortex injection frequency of 100 Hz. The ability of the coating to reduce the  $\phi_{pp}$  of these tones varies with polar angle, although the 100 Hz tone appears to be reduced at all polar angles.

Beyond 400 Hz, the coated airfoil shows a generally lower  $\phi_{pp}$  compared to the bare airfoil. This leads to a significant reduction in OASPL across all polar angles as shown in Figure 10.28d. A maximum decrease in OASPL of 8.1 dB was achieved at the  $\theta = 60^\circ$  polar angle.

Figure 10.29 shows the surface pressure RMS for the bare (Figure 10.29a) and coated (Figure 10.29b) airfoils. The reduction in far-field noise for the coated airfoil is consistent with the lower surface pressure RMS observed on the airfoil surface, particularly near the leading edge. This suggests that the coating reduces the strength of the unsteady pressure fluctuations that contribute to noise generation, providing a likely explanation for the reduction in OASPL.

### 10.3.2.4 Velocity Power Spectral Density

Figure 10.30 shows the probe locations used for the velocity power spectral density. Points 1-4 were placed above the airfoil at two streamwise locations,  $x/C = 0.5$  and  $x/C = 1.0$ , with vertical offsets of  $y/C = 0.1$  and  $y/C = 0.2$ . These locations were selected to examine the development of unsteady velocity

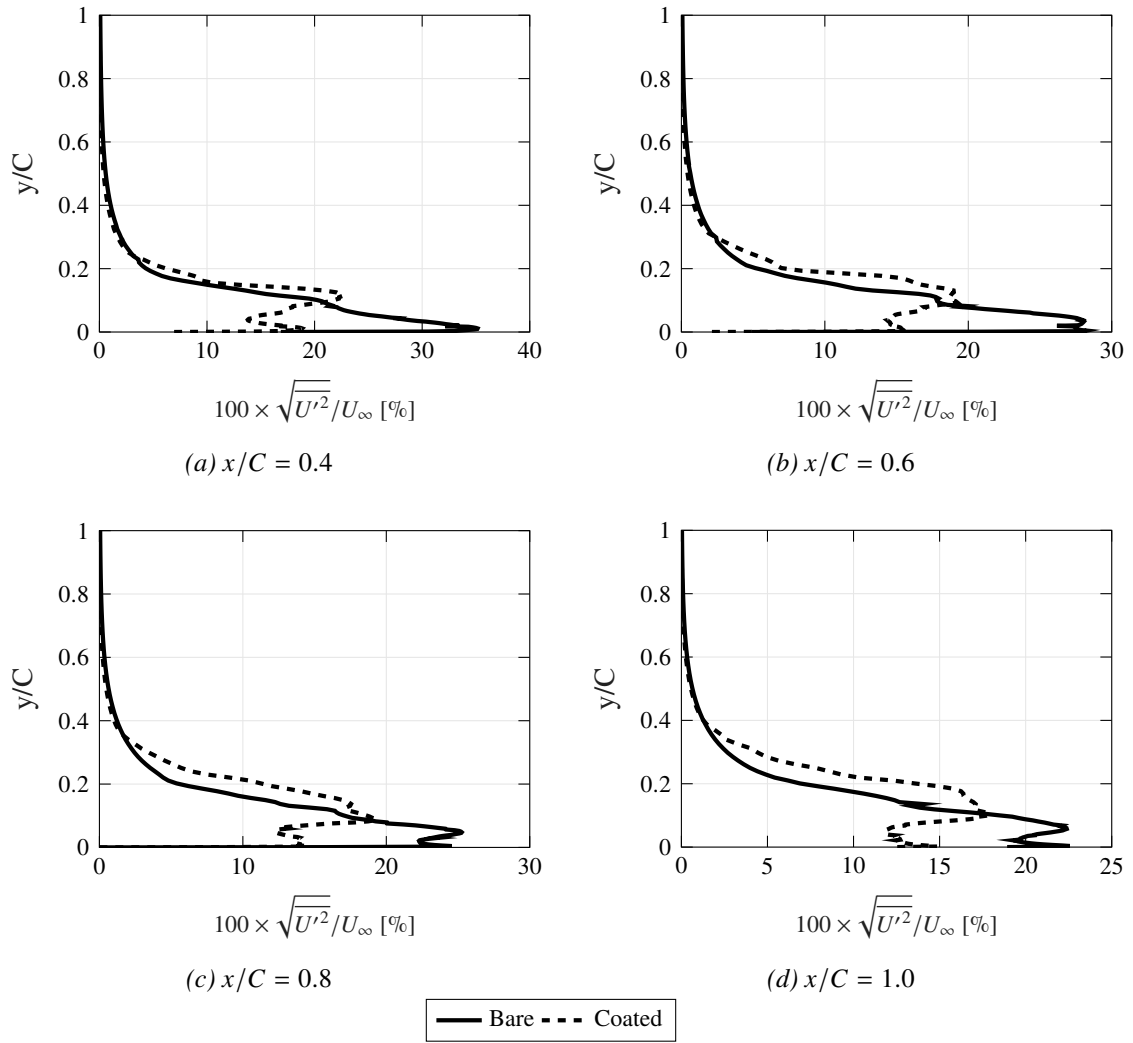


Figure 10.27: Comparison of turbulence intensity profiles between the bare and coated airfoils with impinging vortex (continued on next page).

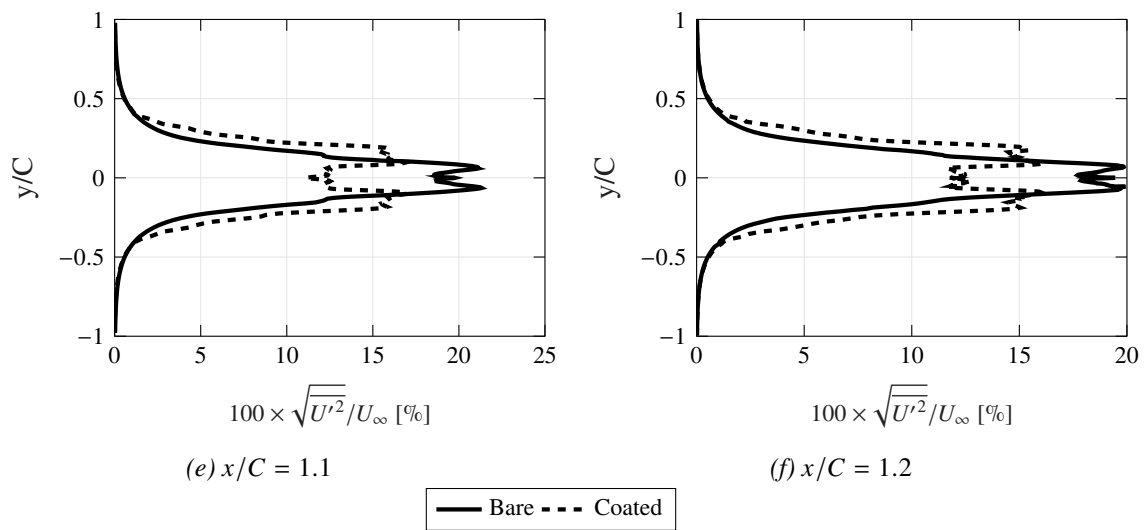


Figure 10.27: Comparison of turbulence intensity profiles between the bare and coated airfoils with impinging vortex (continued).

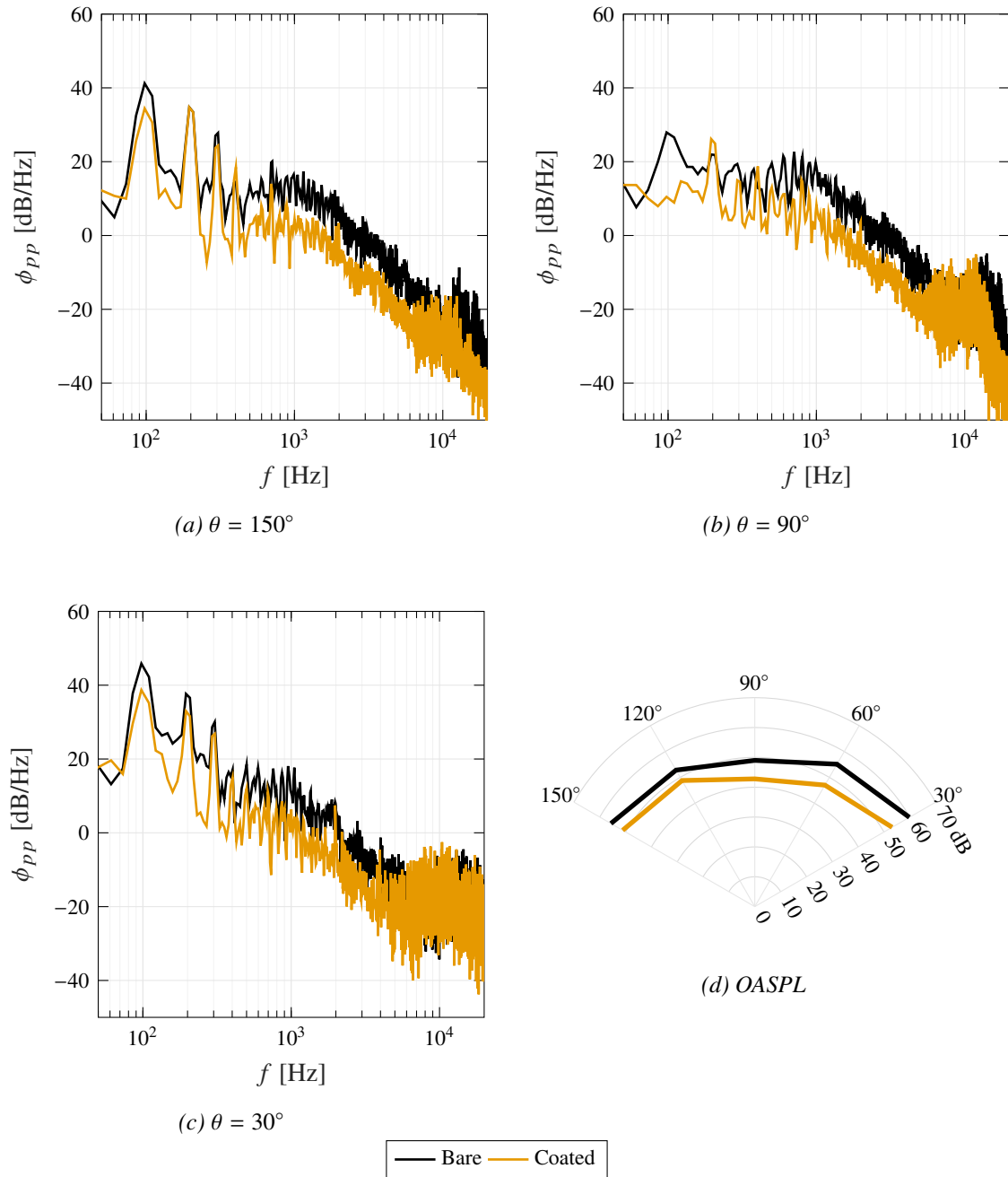


Figure 10.28: Acoustic PSD ( $\phi_{pp}$ ) [dB/Hz] as a function of frequency  $f$  [Hz]

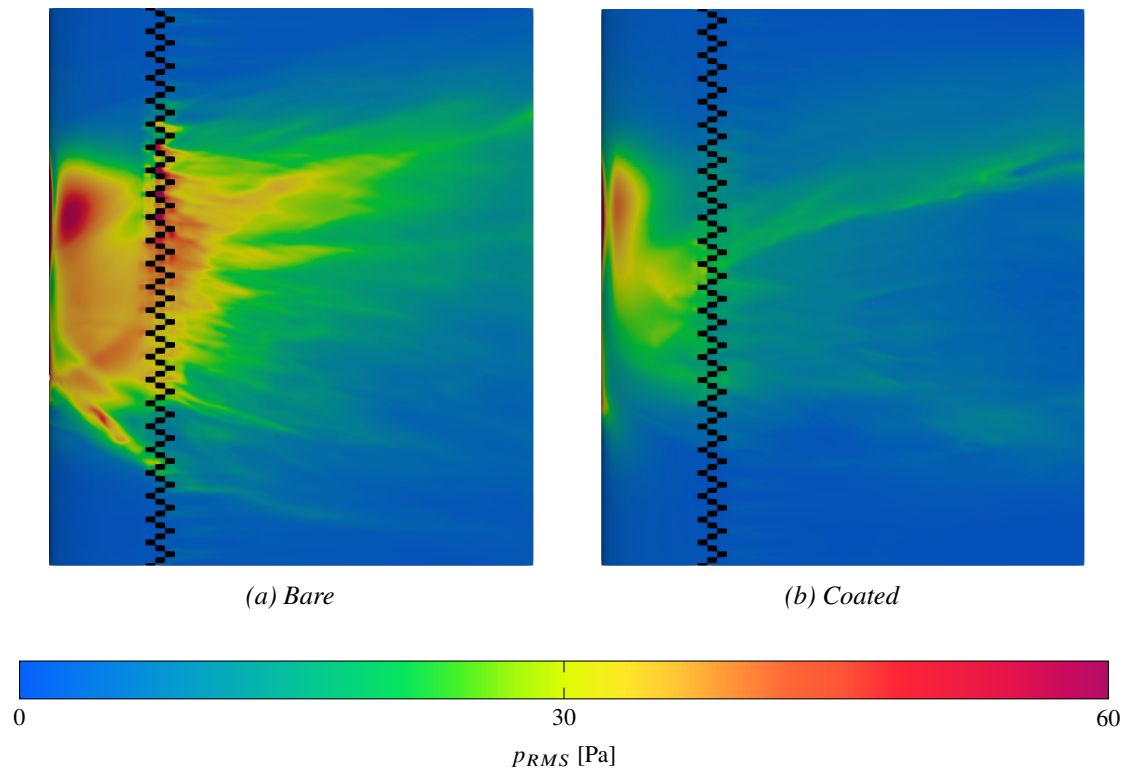


Figure 10.29: Comparison of surface  $p_{RMS}$  between the bare and coated airfoils.

fluctuations over the airfoil surface. Points 5 and 6 were positioned slightly downstream of the trailing edge, at  $x/C = 1.1$ , to assess the wake dynamics immediately downstream of the airfoil.

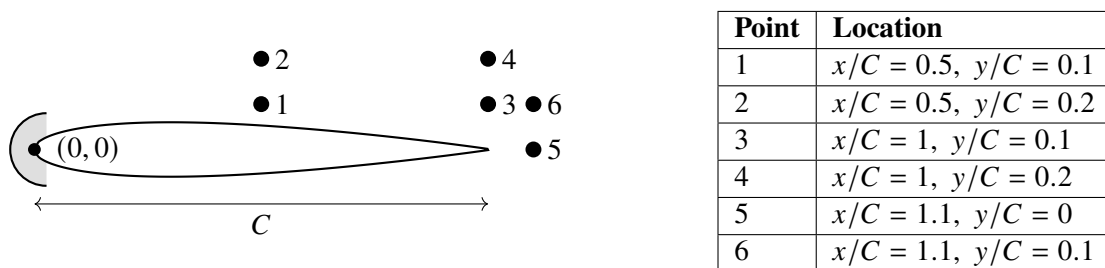


Figure 10.30:  $\phi_{uu}$  probe locations

Figure 10.31 shows the velocity PSD at the various locations shown in Figure 10.30. For the boundary-layer locations (Figures 10.31a–10.31d), the two points closest to the airfoil surface (Figures 10.31a and 10.31c) show a reduction in both low-frequency tonal content and high-frequency broadband content for the coated airfoil relative to the bare airfoil. However, at the two points farther from the airfoil surface (Figures 10.31b and 10.31d), the coated airfoil shows a slight increase in low-frequency tonal content and a more pronounced increase in high-frequency broadband content compared with the bare airfoil. This behaviour is associated with the shear layer emanating from the outer surface of the coating. In the wake, at both probe locations (Figures 10.31e and 10.31f), the coated airfoil shows a reduction in both low-frequency tonal content and high-frequency broadband content relative to the bare airfoil, indicating an overall attenuation of wake unsteadiness across the frequency spectrum.



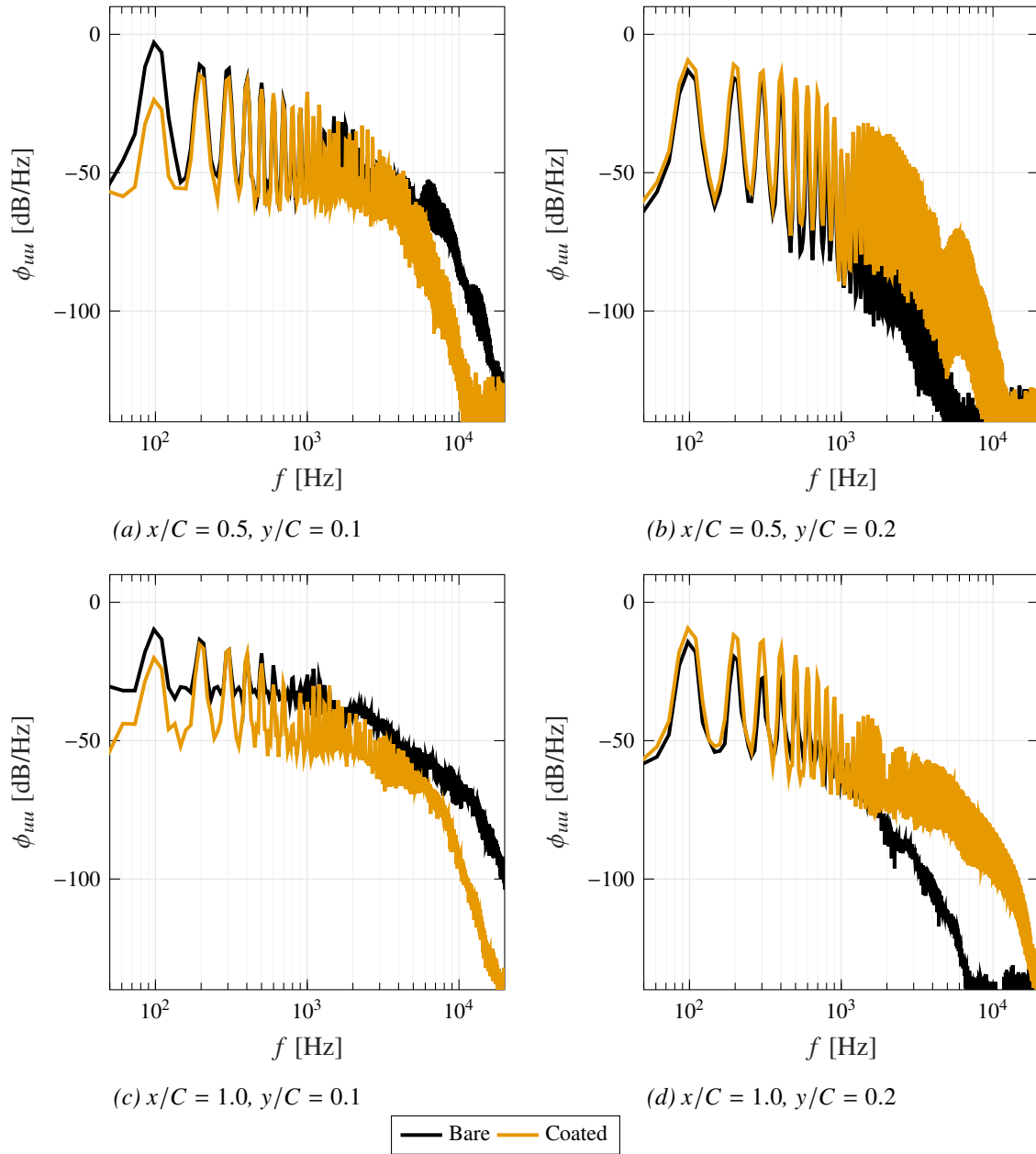


Figure 10.31: Velocity PSD ( $\phi_{uu}$ ) [dB/Hz] as a function of frequency  $f$  [Hz] (continued on next page).

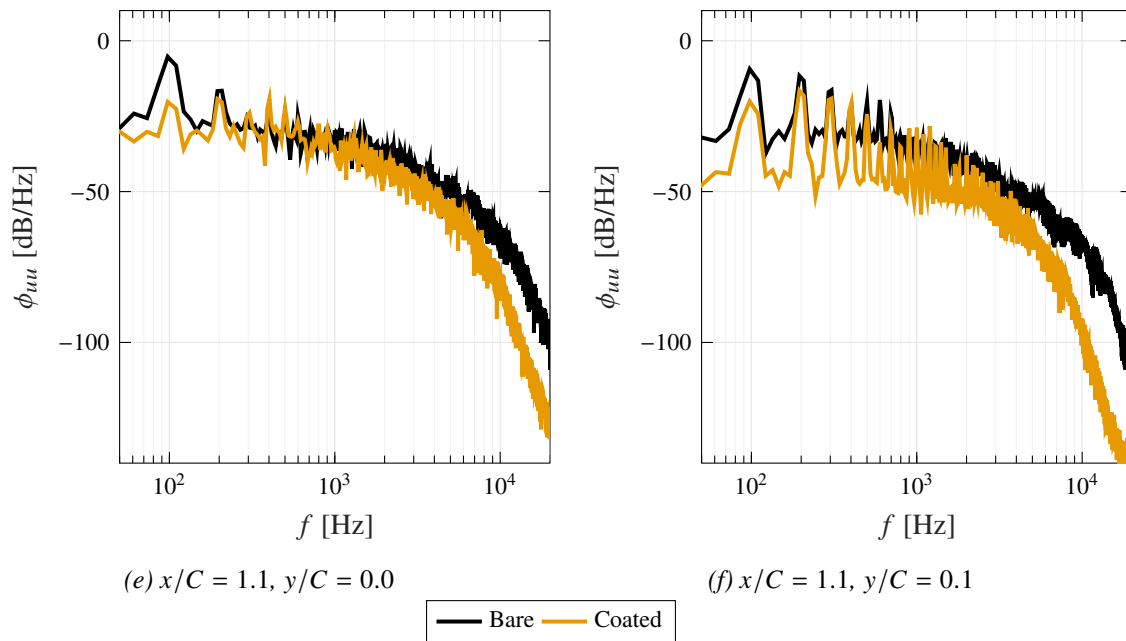


Figure 10.31: Velocity PSD ( $\phi_{uu}$ ) [dB/Hz] as a function of frequency  $f$  [Hz] (continued).

### 10.3.3 Summary

In this chapter, the effect of porous coatings on vortex-body interaction noise was investigated for configurations representative of EUAMV installation-noise mechanisms. A modelled inflow vortex, based on the APC  $12 \times 4.7$ SF propeller operating at 3,000 RPM, was first developed and then applied to both cylinder and airfoil cases. The aim was to assess whether a porous coating could weaken the interaction between an incoming vortex and a nearby pylon (modelled as a cylinder), or a wing (modelled as an airfoil).

For the cylinder case, the incoming vortex wrapped around the bare cylinder, while it appeared to pass over the porous coating in the coated case. The coated cylinder also produced a more diffused wake vorticity field, indicating that the coating reduced the coherence of the vortex after impingement. The mean velocity results showed that the coated cylinder produced a broader and longer low-velocity region in the wake than the bare cylinder, both in uniform flow and with inflow vortices. This behaviour was consistent with the previous Darcy–Forchheimer coated-cylinder results and confirms that the coating redistributed momentum and delayed wake recovery.

The turbulence intensity results showed that the porous coating reduced the near-wake turbulence intensity peaks for both uniform-flow and vortex-impingement conditions. In the vortex case, the bare cylinder produced strong turbulence close to the surface, whereas the coated cylinder produced a more diffused turbulence field. Although the coated cylinder reduced the near-body unsteadiness, it also shifted part of the turbulence farther downstream, where wider and stronger turbulence intensity profiles were observed compared with the bare cylinder.

The cylinder acoustic results showed that the porous coating reduced the acoustic PSD over a broad frequency range. In uniform flow, the largest OASPL reduction was approximately 10 dB at the  $90^\circ$  observer location. Under vortex impingement, tones appeared at 100 Hz intervals, corresponding to

the vortex injection frequency. The coated cylinder increased the first visible tone, but reduced the higher-frequency tones and remained below the bare-cylinder spectrum beyond 200 Hz. A maximum OASPL reduction of approximately 5.8 dB was obtained at the 90° polar angle, although an increase of approximately 3.1 dB was observed at 30°. This shows that the porous coating provided a noise reduction, although this does depend on the observer location.

The airfoil leading-edge case showed similar vortex-diffusion behaviour. The incoming vortex remained closer to the surface for the bare airfoil, while it passed farther above the surface in the coated case due to the porous leading-edge region. The bare airfoil preserved a more coherent downstream vortex imprint, whereas the porous coating diffused the incoming vortex and weakened the imprint near the trailing edge. The mean velocity profiles showed that the coating altered the boundary-layer and wake development, producing a wider and deeper wake deficit than the bare airfoil. The coating was also able to reduce surface pressure fluctuations on the airfoil surface when compared with the bare airfoil.

The turbulence intensity results for the airfoil showed that the porous leading-edge coating reduced the unsteadiness generated near the surface. The bare airfoil produced high turbulence intensity aft of the leading edge, while the coated airfoil diffused this region and reduced the peak magnitude. The coated case generally produced slightly wider but significantly lower turbulence intensity peaks across both the boundary-layer and wake locations, indicating that the coating reduced the strength of the vortex-induced unsteadiness.

The airfoil acoustic results showed distinct tones at 100 Hz intervals, coinciding with the vortex injection frequency. The porous coating reduced the 100 Hz tone at all observer locations and generally produced lower acoustic PSD than the bare airfoil above 400 Hz. This resulted in an OASPL reduction across all polar angles, with a maximum decrease of 8.1 dB at the 60° polar angle. The velocity PSD supported this trend, showing reductions in both tonal and broadband content near the airfoil surface and in the wake, although some low-frequency increases occurred farther from the surface due to the shear layer generated by the outer coating.

The results show that porous coatings can reduce vortex-interaction noise for both pylon-like cylinder geometries and wing-like airfoil leading edges. For the cylinder, the coating diffused the impinging vortex, broadened the wake, reduced near-wake turbulence intensity and achieved a maximum OASPL reduction of approximately 5.8 dB under vortex impingement. For the airfoil, the porous leading-edge coating weakened the downstream vortex imprint, reduced turbulence intensity over the airfoil and in the wake, and reduced OASPL across all polar angles by up to 8.1 dB. However, the coating also introduced shear-layer structures from its outer surface, which locally increased spectral energy at some probe locations. Clearly, the aerodynamic effects need to be investigated, especially at non-zero angle of attack. Therefore, while porous coatings show potential for reducing vortex-impingement noise in EUAMV installation scenarios, their effectiveness depends on the observer direction, geometry and inflow conditions.

# 11 CONCLUSIONS

## 11.1 Conclusion

This thesis investigated the use of computational fluid dynamics for predicting and reducing aerodynamic noise associated with Urban Air Mobility Vehicles. Particular attention was given to airfoil self-noise, bare and porous-coated cylinder noise, and vortex-body interaction noise, since these mechanisms are directly relevant to rotor-pylon and rotor-wing installation effects. The numerical framework was first validated using established 2-D RANS and 3-D LES reference cases, providing confidence in the OpenFOAM solver setup, turbulence modelling choices, discretisation schemes and acoustic post-processing methods used in the later investigations.

The main contributions of this work were as follows:

- The development and use of a consistent OpenFOAM framework for aerodynamic and aeroacoustic prediction of UAMV-relevant flow mechanisms, including airfoil self-noise, bare and coated cylinder noise and vortex-body interaction noise.
- The combined aerodynamic and aeroacoustic assessment of a tripped NACA 0012 airfoil using LES, ZLES and IDDES, including the development and assessment of a new ZLES implementation in OpenFOAM. The results showed that LES and ZLES were able to capture the relevant boundary-layer, wake and acoustic behaviour, while IDDES was unable to generate sufficient resolved turbulence for accurate acoustic prediction in this configuration.
- The comparison of Curle, FW-H 1A and GT acoustic analogies, together with the investigation of FW-H surface and closure-disc placement. Convergent behaviour in OASPL was found with FW-H surface distance from turbulent flow. The use of the pressure RMS field to inform FW-H surface placement was also proposed, providing a physically motivated approach for reducing hydrodynamic contamination of the acoustics.
- The investigation of structured porous coated cylinders using both fully resolved LES and Darcy–Forchheimer modelling. The resolved simulations provided insight into the internal flow mechanisms of the coating, including separation around the structural members, flow through the circumferential pores, boundary-layer development on the inner cylinder and shear-layer diffusion from the outer coating. The Darcy–Forchheimer approximation provided a reduced-cost approach for modelling the same class of structured porous coating over larger spanwise domains.
- The comparison between the fully resolved and modelled porous-cylinder approaches, which highlighted both the usefulness and limitations of porous-medium approximations. In particular, the results showed the sensitivity to the chosen coefficients, the difficulty of Darcy–Forchheimer calibration for non-channel-like structured topologies, and the inability of the model to reproduce all internal flow physics. A calibration correction was introduced and found to give the best results.

- The extension of porous coating concepts to vortex-body interaction noise mechanisms relevant to UAMV installation effects. A modified Kaufmann vortex with cosine tapering was implemented to provide a controlled inflow vortex that could interact with the bodies without contaminating the periodic boundaries or acoustic surfaces.
- The assessment of porous coatings under modelled propeller-tip-vortex impingement on both a cylinder and an airfoil leading edge. The results showed that porous coatings can diffuse incoming vortices, reduce the surface pressure fluctuations, and reduce momentum in the near-wake, resulting in a generally lower Overall Sound Pressure Level.
- The demonstration that porous coatings can reduce vortex-body interaction noise in both configurations considered. For the cylinder, the porous coating produced a maximum OASPL reduction of approximately 5.8 dB under vortex impingement, while the coated airfoil leading edge produced reductions across all observer angles, with a maximum reduction of 8.1 dB.

Overall, the results demonstrate that porous coatings have clear potential as passive noise-reduction treatments for UAMV-relevant flow configurations. Their effectiveness is linked to their ability to redistribute momentum, diffuse coherent vortical structures and reduce surface pressure fluctuations responsible for sound generation, with some caveats. The porous coating can introduce additional shear-layer unsteadiness from its outer surface, and the resulting acoustic response depends on the geometry, observer direction, porous resistance and inflow condition.

The novelty of this thesis therefore lies not only in applying porous coatings to aeroacoustic noise reduction, but in linking the resolved flow mechanisms within structured porous coatings to reduced-order porous modelling and vortex-body interaction noise. The work demonstrates how validated CFD, acoustic analogies and porous-medium modelling can be combined to assess both the mechanisms and practical limitations of passive noise-reduction concepts for future UAMV applications.

## 11.2 Limitations

Although the accuracy of numerical modelling generally depends on the choice of solvers, numerical schemes, and turbulence models, two key limitations identified during this research were the domain spanwise extent and the simulation time. While both of these factors are less critical for the airfoil cases, the spanwise extent became important for the nozzle inflow case, and both spanwise extent and simulation time were important for the bare and coated cylinders. In particular, the limited spanwise extent prevented the quasi-2-D turbulent structures in the wake from breaking down into fully 3-D turbulence for the resolved SPCC. Increasing the spanwise extent to address this issue, however, results in a significantly higher cell count and therefore requires greater computational resources. Similarly, accurately capturing the vortex shedding tones requires the simulation to run for a sufficiently long time. Unfortunately, this also increases the computational cost, and in some cases the HPC scheduling can be prohibitive.

## 11.3 Future Work

In the present work, the main advantage of ZLES was the significant reduction in computational cost whilst maintaining predictive accuracy comparable to that of LES. The implemented ZLES model and the WALE LES model should be evaluated on the same sequence of progressively coarser computational grids.

This would enable the assessment of how well the ZLES maintains predictive accuracy as the resolution is reduced and help to identify the grid resolution at which the advantages over conventional LES become apparent.

Whilst the Darcy–Forchheimer modelling showed good agreement with the resolved SPCC, a “sandwich case” was proposed in which the resolved SPCC geometry could be placed between two Darcy–Forchheimer regions. This would have allowed the use of a larger span while investigating the effect of the resolved SPCC under inflow turbulence in the form of impinging vortices. Although the computational domain for this case had been created, the simulation was not run due to time constraints.

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# A HIGH PERFORMANCE COMPUTING FOR CFD

Figure A.1 shows the performance development of High Performance Computing (HPC) measured in Floating Point Operations per Second (FLOPS). As of June 2022, *Frontier*, in the United States was the fastest supercomputer in the world with a maximum performance of 1,102 petaflops, while the UK's *Archer2* was ranked the 25th fastest supercomputer with a maximum performance of 19.54 petaflops [168].

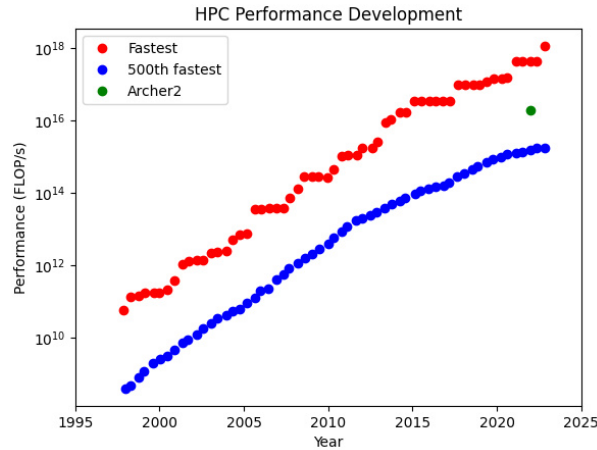


Figure A.1: HPC Performance Development  
(adapted from TOP500 [169])

With the increase in performance also came a decrease in the cost of supercomputing, from an estimated \$30,000/gigaflop in 1997 to \$0.03/gigaflop in 2017, roughly a million times decrease in cost [170]. This decrease in cost means that high performance computing is more accessible, and for CFD, it can be used for running many small simulations, single large simulations and optimisation.

## A.1 Scaling

Scaling in parallel computation refers to how the performance of a parallel application changes as the number of processors is increased. There are two types of scaling, as discussed below [171]:

- Weak scaling - increasing the size of the problem at the same rate as increasing the number of processors, for example, doubling the mesh size and doubling the number of processors used. Therefore, the amount of work done per processor remains constant.
- Strong scaling - keeping the total problem size the same but increasing the number of processors. Although this is more useful than weak scaling, it is also more difficult to achieve.

Although the speed of processors have reached a nearly constant level, computational power continues to increase with the use of parallel architectures. However, parallel scaling still remains a limitation [151]. According to Amdahl's Law, "the overall performance gained by optimizing a single part of a system is limited by the fraction of the time that the improved part is actually used" [172]. Given a fraction,  $\alpha$ , where  $\alpha = 1$  represents a completely serial section of the application, and assuming that the parallel

section is 100% efficient, the parallel runtime is given by Equation A.1:

$$T(N, P) = \alpha T(N, 1) + \frac{(1 - \alpha)T(N, 1)}{P} \quad (\text{A.1})$$

The parallel speedup is then given by Equation A.2:

$$S(N, P) = \frac{T(N, 1)}{T(N, P)} = \frac{P}{\alpha P + (1 - \alpha)} \quad (\text{A.2})$$

For Equations A.1 and A.2:

|          |                                       |     |                      |
|----------|---------------------------------------|-----|----------------------|
| $T$      | Execution time                        | $S$ | Speedup              |
| $N$      | Size of the problem                   | $P$ | Number of processors |
| $\alpha$ | Fraction of completely serial section |     |                      |

From Equations A.1 and A.2, it can be seen that the speedup is limited by the serial fraction  $\alpha$ . Therefore, when  $\alpha = 0$ , the speedup is equal to the number of processors and the efficiency of the operation is 100%. However, in all other cases, the speedup is limited by  $1/\alpha$ . Therefore, an application with a 10% serial section, will result in a maximum speedup of 10 times, regardless of the number of processors used.

## A.2 Scaling in OpenFOAM

The available matrix solvers in OpenFOAM and the claimed parallel scaling performance are shown in Table A.1.

*Table A.1: Claimed parallel scaling of OpenFOAM solvers*

[173]

| <b>Solver</b>                                               | <b>Parallel scaling</b> |
|-------------------------------------------------------------|-------------------------|
| Preconditioned Conjugate Gradient (PCG)                     | Good                    |
| Preconditioned Bi-conjugate Gradient (PBiCG)                | Good                    |
| Stabilized Preconditioned Bi-conjugate Gradient (PBiCGStab) | Good                    |
| Solver using a smoother (smoothSolver)                      | N/A                     |
| Generalised Geometric-algebraic Multi-grid (GAMG)           | Moderate                |

The preconditioned solvers require a preconditioner. The available preconditioners in OpenFOAM and the claimed parallel consistency are shown in Table A.2.

*Table A.2: Claimed parallel consistency of OpenFOAM preconditioners*

[173]

| <b>Preconditioner</b>                                         | <b>Parallel consistency</b> |
|---------------------------------------------------------------|-----------------------------|
| Diagonal Incomplete-Cholesky (DIC)                            | Inconsistent                |
| Faster Diagonal Incomplete-Cholesky (DIC with caching) (FDIC) | Inconsistent                |
| Diagonal Incomplete-LU (DILU)                                 | Inconsistent                |
| Diagonal                                                      | Consistent                  |
| GAMG                                                          | Inconsistent                |

With respect to scalability tests done on OpenFOAM, Pringle [174] found superlinear scalability of

OpenFOAM on the supercomputer, HECToR, up to 1,024 cores. Various OpenFOAM solvers (icoFoam, interFoam and pisoFoam) were tested for the respective tutorial cases. However, the authors concluded that for the different available solvers in OpenFOAM, there is no relation between the optimum number of cores used for a given number of cells.

Rivera and F rlinger [175] tested the scalability of the pisoFoam solver for LES of a backward facing step on the SGI Altix 4700 supercomputer. The mesh consisted of 2.15 million cells and tests were conducted from 16 to 256 cores. It was found that the solver scaled well up to 64 cores, and then slowed down for 128 and 256 cores. The authors suggested that less than 34,000 cells per core was insufficient, thus causing the drop in speedup above 64 cores.

Orgogozo et al. [140] developed a solver for Richards equation, named RichardsFoam. The scalability of the solver was tested on the CALMIP cluster (Hyperion), from 16 to 1,024 cores. The mesh consisted of approximately 36 million cells. With respect to the strong scaling tests, a 90% parallel efficiency was achieved for 1,024 cores without Input/Output (I/O), and an 80% parallel efficiency with I/O, corresponding to approximately 35,000 cells per core. For the weak scaling tests, 130,000 cells per core was used. Similar to the strong scaling test, a 90% and 80% efficiency was achieved without and with I/O respectively, up to 1,024 cores.

Duran et al. [176] tested the solver, icoFoam, on the TGCC Curie supercomputer for the purpose of blood flow simulations. The largest tested mesh consisted of 64 million cells. Speedup was observed up to 16,384 cores while linear speedup was achieved up to 4,096 cores, corresponding to 3,906 and 15,625 cells per core respectively.

 sidlof and  ridk y [177] tested the scalability of OpenFOAM for flow past a fluttering airfoil on the SGI Altix UV 100 supercomputer. The tests were conducted from 1 to 36 cores. It was concluded that as long as the number of cells per core does not drop below 100,000, the code scales well but not linearly.

Horgue et al. [178] developed a program named "porousMultiphaseFoam" to simulate multiphase flow in porous media. The scalability of the code was tested on a 1.6 million cell mesh, from 16 to 1,024 cores, on the Hyperion supercomputer. Superlinear speedup was achieved up to 256 cores, while linear scaling was achieved for 512 cores. The parallel efficiency for 1,024 cores was found to be only 0.59, and was attributed to the low cell count per core (1,560).

Guibert et al. [179] developed two solvers based on the SIMPLE algorithm to evaluate pore-scale flows. Scaling tests were conducted on the CALMIP cluster between 16 and 256 cores, and the tested mesh size consisted of 4,177,504 cells. Up to the maximum tested number of cores (256), superlinear speedup was achieved. However, for the same mesh size using periodic boundary conditions, superlinear speedup was only achieved up to 64 cores. This reduction in speedup was attributed to the increase in information exchange between processors when using periodic boundary conditions.

Axtmann and Rist [139] conducted both LES and Direct Numerical Simulation (DNS) for a backward facing step, using the Cray XC40 Hazel Hen supercomputer. The solver used was pisoFoam. Similar to Pringle [174], superlinear scalability was found up to 1,000 Message Passing Interface (MPI) tasks. However, in this case, the authors suggested an optimum number of cells per core of between 15,000 and



20,000.

Zheng et al. [180] tested the scaling of the solvers icoFoam and nonNewtonianIcoFoam for DNS of turbulent Newtonian and non-Newtonian flow. Simulations were conducted on the NCI-Raijin cluster, and scaling tests were carried out from 1 to 512 cores. For a mesh containing 4.1 million cells, the solvers were found to scale very well from 8 to 512 cores. However, it was noted that the intranode scalability between 1 and 8 cores was poor. Optimum parallel efficiency of 46.8% was achieved when using 128 cores, corresponding to 32,000 cells per core. It was also suggested that OpenFOAM scales better to multiple nodes due to the additional memory bandwidth available, and that saturation of the bandwidth to memory results in the intranode scalability being poor.

Elfverson and Lejon [138] tested the solver, simpleFoam, to simulate wind fields and pollution dispersion. Two meshes containing 10,839,462 and 2,292,898 cells were tested up to 80 cores. It was found that for both the mesh sizes, the solver exhibited excellent scaling up to 35 cores, corresponding to approximately 310,000 and 65,500 cells per core respectively. Based on the similar speedup achieved in both cases, it is implied that the speedup is independent of the number of cells per core, and limited only by the communication between processors beyond 35 cores.

From the above, it can be seen that while OpenFOAM exhibits adequate scalability, especially with respect to strong scaling, there is no clear “rule of thumb” to follow with respect to mesh decomposition across processors. The optimum number of cells to use per core very much depends on the available computing architecture, the solver being used, and the case setup with respect to the type of boundary conditions used, etc.

### A.3 Initial Strong Scaling Tests on Archer2

The same cases discussed in Section 4.1 were then run on the Tier 1 HPC machine, Archer2. The smaller mesh (10.4M cells) was run on 128 to 1,408 cores (1 to 11 nodes), while the larger mesh was run on 128 to 2,944 cores (1 to 23 nodes). The cells per core ranged from between 7,064 to 162,465. The results are tabulated in Table A.3 and Table A.4. The largest observed difference between clock time and CPU time was found to be 34.68% for the case run on 2,688 cores. While this difference is significant, the general trend shows a better correspondence between clock time and CPU time when compared to MMM Young, as can be seen in Figure A.2.

*Table A.3: CPU time and clock time for rhoPimpleFoam strong scaling (10.4M cell mesh)*

| No. of cores | Cells per core | Clock time (s) | CPU time (s) | % difference |
|--------------|----------------|----------------|--------------|--------------|
| 128          | 81,232         | 3,264          | 3,254        | 0.31         |
| 256          | 40,616         | 1,286          | 1,268        | 1.42         |
| 384          | 27,077         | 756            | 735          | 2.86         |
| 640          | 16,246         | 473            | 455          | 3.96         |
| 896          | 11,605         | 276            | 254          | 8.66         |
| 1,152        | 9,026          | 215            | 195          | 10.26        |
| 1,408        | 7,385          | 640            | 606          | 5.61         |

From Figure A.2, it can be seen that speedup is achieved up to 1,152 cores for the smaller mesh, corresponding to 9,062 cells per core, while for the larger mesh, speedup is achieved up to 2,688 cores,

Table A.4: CPU time and clock time for rhoPimpleFoam strong scaling (20.8M cell mesh)

| No. of cores | Cells per core | Clock time (s) | CPU time (s) | % difference |
|--------------|----------------|----------------|--------------|--------------|
| 128          | 162,465        | 7,679          | 7,638        | 0.54         |
| 256          | 81,232         | 3,322          | 3,308        | 0.42         |
| 384          | 54,155         | 1,971          | 1,956        | 0.77         |
| 640          | 32,493         | 993            | 976          | 1.74         |
| 896          | 23,209         | 649            | 633          | 2.53         |
| 1,152        | 18,052         | 547            | 516          | 6.01         |
| 1,408        | 14,770         | 478            | 441          | 8.39         |
| 1,664        | 12,497         | 331            | 289          | 14.53        |
| 1,920        | 10,831         | 299            | 255          | 17.25        |
| 2,176        | 9,557          | 284            | 224          | 26.79        |
| 2,432        | 8,551          | 246            | 197          | 24.87        |
| 2,688        | 7,736          | 233            | 173          | 34.68        |
| 2,944        | 7,064          | 293            | 228          | 28.51        |

corresponding to 7,736 cells per core.

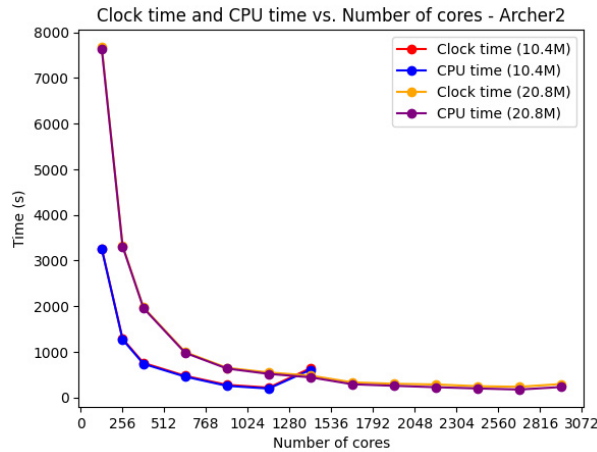


Figure A.2: Comparison between clock time and CPU time for a different number of cores

Similar to Section 4.1, Amdahl's Law was used to determine the parallel speedup and parallel efficiency for both the smaller and larger cases, using both a 128 and a 256 core reference. The results are shown in Table A.5 and Table A.6. When using a 128 core reference, it can be seen that for both the smaller and larger cases, a speedup of 2.57 and 2.31 respectively is achieved for 256 cores. This large speedup can be attributed to the poor intranode scalability of OpenFOAM, similar to that found by Zheng et al. [180], who suggested that OpenFOAM scales better to multiple nodes due to the additional memory bandwidth available.

Figures A.3a and A.3b show a log-log representation of the speedup versus the number of cores, using the 128 cores case and the 256 cores case as references respectively. For the smaller mesh, superlinear speedup is achieved up to 1,152 cores for both references. For the larger mesh, superlinear speedup is achieved for both references up to the maximum tested number of cores, i.e., 2,944 cores, although the simulation runs fastest on 2,688 cores.

Table A.5: Speedup and parallel efficiency of rhoPimpleFoam (10.4M cell mesh)

| No. of cores | 128 core reference |                     | 256 core reference |                     |
|--------------|--------------------|---------------------|--------------------|---------------------|
|              | Speedup            | Parallel efficiency | Speedup            | Parallel efficiency |
| 128          | 1.00               | 1.00                | N/A                | N/A                 |
| 256          | 2.57               | 1.28                | 1.00               | 1.00                |
| 384          | 4.43               | 1.48                | 1.73               | 1.15                |
| 640          | 7.15               | 1.43                | 2.79               | 1.11                |
| 896          | 12.81              | 1.83                | 4.99               | 1.43                |
| 1,152        | 16.69              | 1.85                | 6.50               | 1.45                |
| 1,408        | 5.37               | 0.49                | 2.09               | 0.38                |

Table A.6: Speedup and parallel efficiency of rhoPimpleFoam (20.8M cell mesh)

| No. of cores | 128 core reference |                     | 256 core reference |                     |
|--------------|--------------------|---------------------|--------------------|---------------------|
|              | Speedup            | Parallel efficiency | Speedup            | Parallel efficiency |
| 128          | 1.00               | 1.00                | N/A                | N/A                 |
| 256          | 2.31               | 1.15                | 1.00               | 1.00                |
| 384          | 3.90               | 1.30                | 1.69               | 1.13                |
| 640          | 7.83               | 1.57                | 3.39               | 1.36                |
| 896          | 12.07              | 1.72                | 5.23               | 1.49                |
| 1,152        | 14.80              | 1.64                | 6.41               | 1.42                |
| 1,408        | 17.32              | 1.57                | 7.50               | 1.36                |
| 1,664        | 26.43              | 2.03                | 11.45              | 1.76                |
| 1,920        | 29.95              | 2.00                | 12.97              | 1.73                |
| 2,176        | 34.10              | 2.01                | 14.77              | 1.74                |
| 2,432        | 38.77              | 2.04                | 16.79              | 1.77                |
| 2,688        | 44.15              | 2.10                | 19.12              | 1.82                |
| 2,944        | 33.50              | 1.46                | 14.51              | 1.26                |

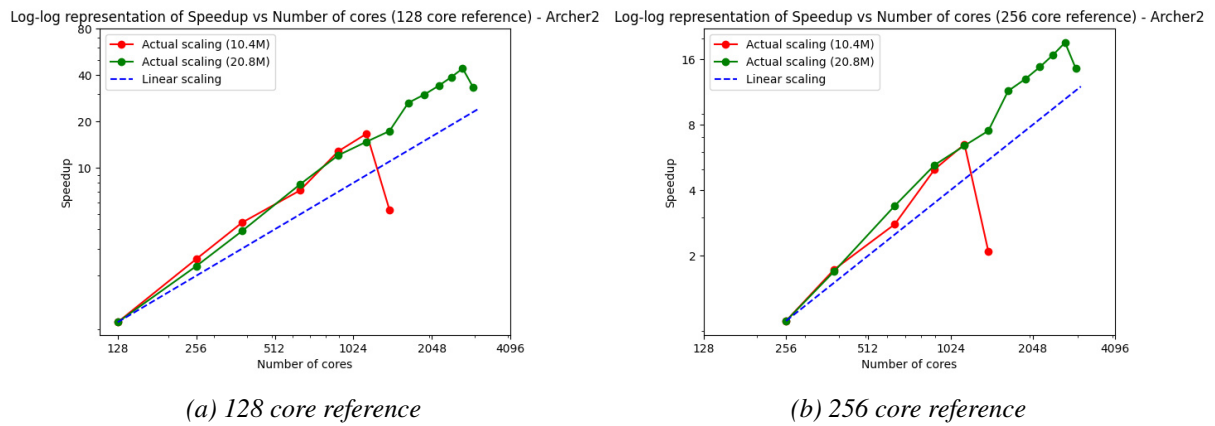


Figure A.3: Speedup versus number of cores using a 128 core and 256 core reference

## A.4 Requesting Resources on Archer2

Jobs on Archer2 are charged in Computing Resources (CUs), where 1 *CU* = 1 *Node hour* = 128 *Core hours*, since every node on Archer2 has 128 cores. Table A.7 shows the properties of the fastest tested case on Archer2 for the larger mesh.

Table A.7: Properties of fastest tested case on Archer2 (20.8M cell mesh)

|                     |            |
|---------------------|------------|
| Number of cells     | 20,795,500 |
| Number of timesteps | 1,000      |
| Number of cores     | 2,688      |
| Cells per core      | 7,736      |
| Clock time (s)      | 233        |
| CPU time (s)        | 173        |

Therefore, the number of CUs to request for a desired mesh size and number of timesteps can be found as follows:

- The core hours used is given by Equation A.3:

$$\text{Core hours} = \frac{(2688 \times 173)}{3600} = 129 \quad (\text{A.3})$$

- The core hours per time step is then given by Equation A.4:

$$\text{Core hours per timestep} = \frac{129}{1000} = 0.129 \quad (\text{A.4})$$

- The core hours per time step per cell is given by Equation A.5:

$$\text{Core hours per timestep per cell} = \frac{0.129}{20795500} = 6.21 \times 10^{-9} \quad (\text{A.5})$$

- The required CUs per time step per cell is given by Equation A.6:

$$\text{CUs per timestep per cell} = \frac{6.21 \times 10^{-9}}{128} = 4.85 \times 10^{-11} \quad (\text{A.6})$$

Therefore, by multiplying the value of  $4.85 \times 10^{-11}$  by a desired mesh size (number of cells), and desired number of timesteps, the required number of CUs can be found.

# B LES VALIDATION – 0.2C AND 0.6C

## B.1 Span = 0.2C

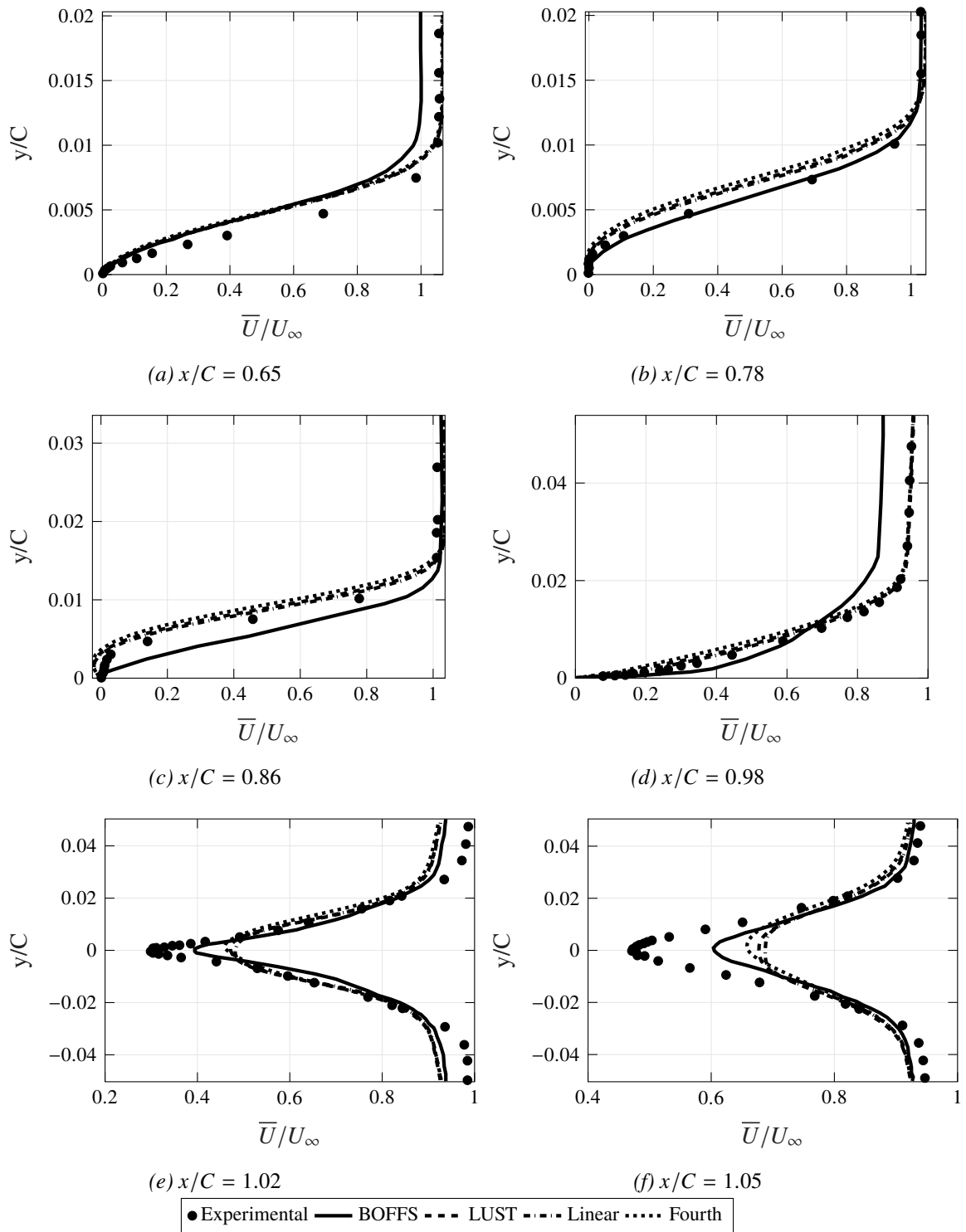
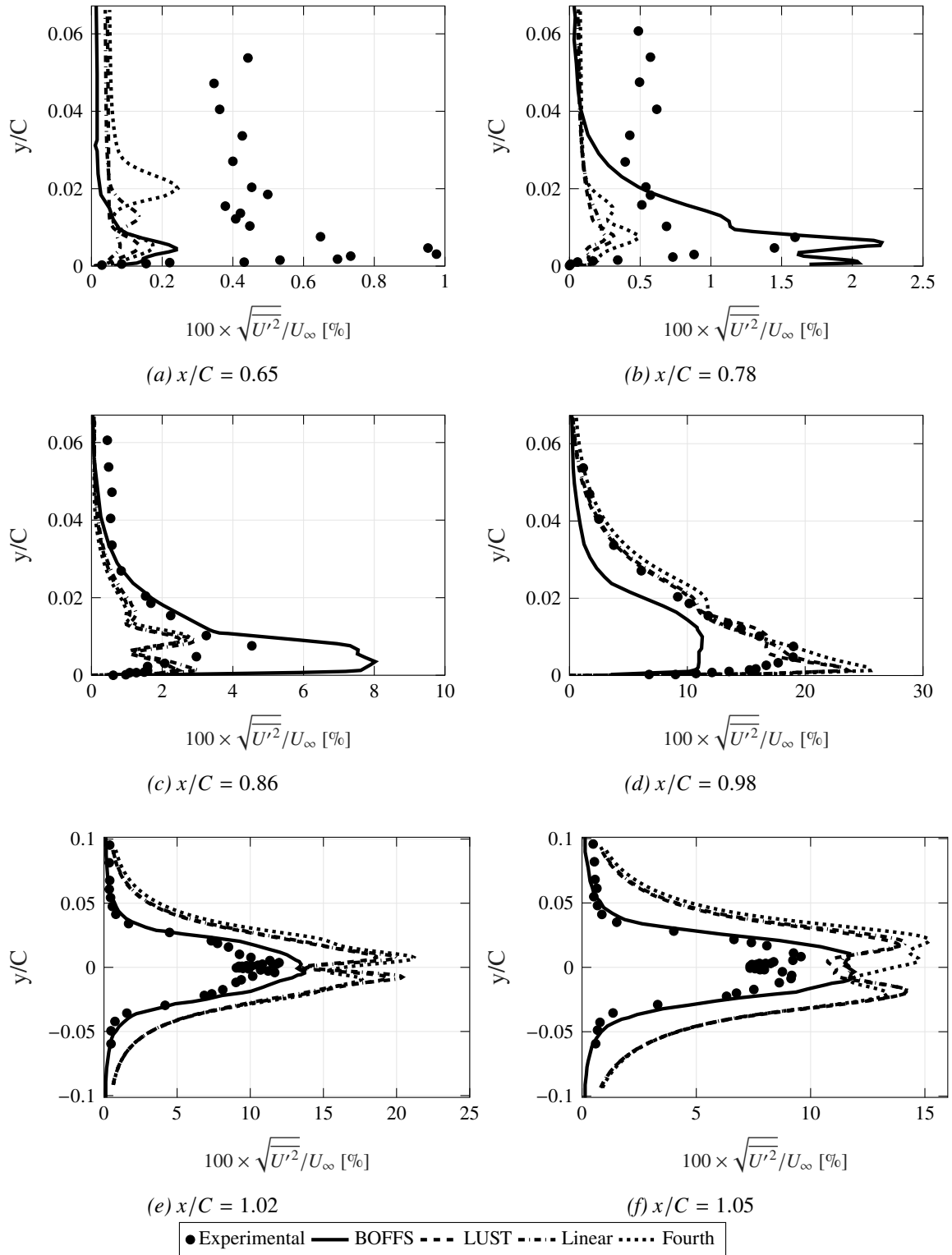


Figure B.1: Normalised streamwise mean velocity profiles for  $\text{Span} = 0.2C$

Figure B.2: Turbulence intensity profiles for  $Span = 0.2C$

## B.2 Span = 0.6C

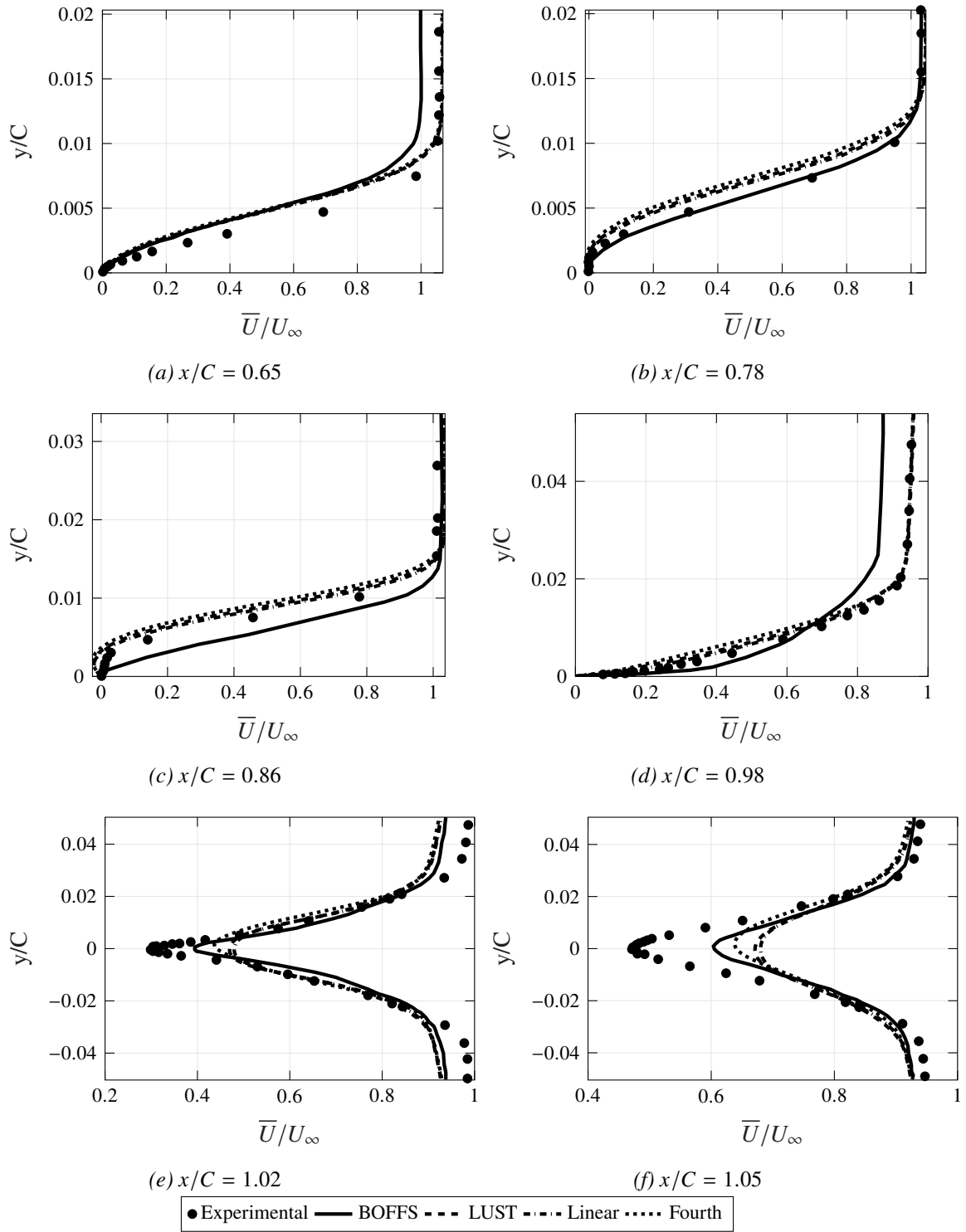
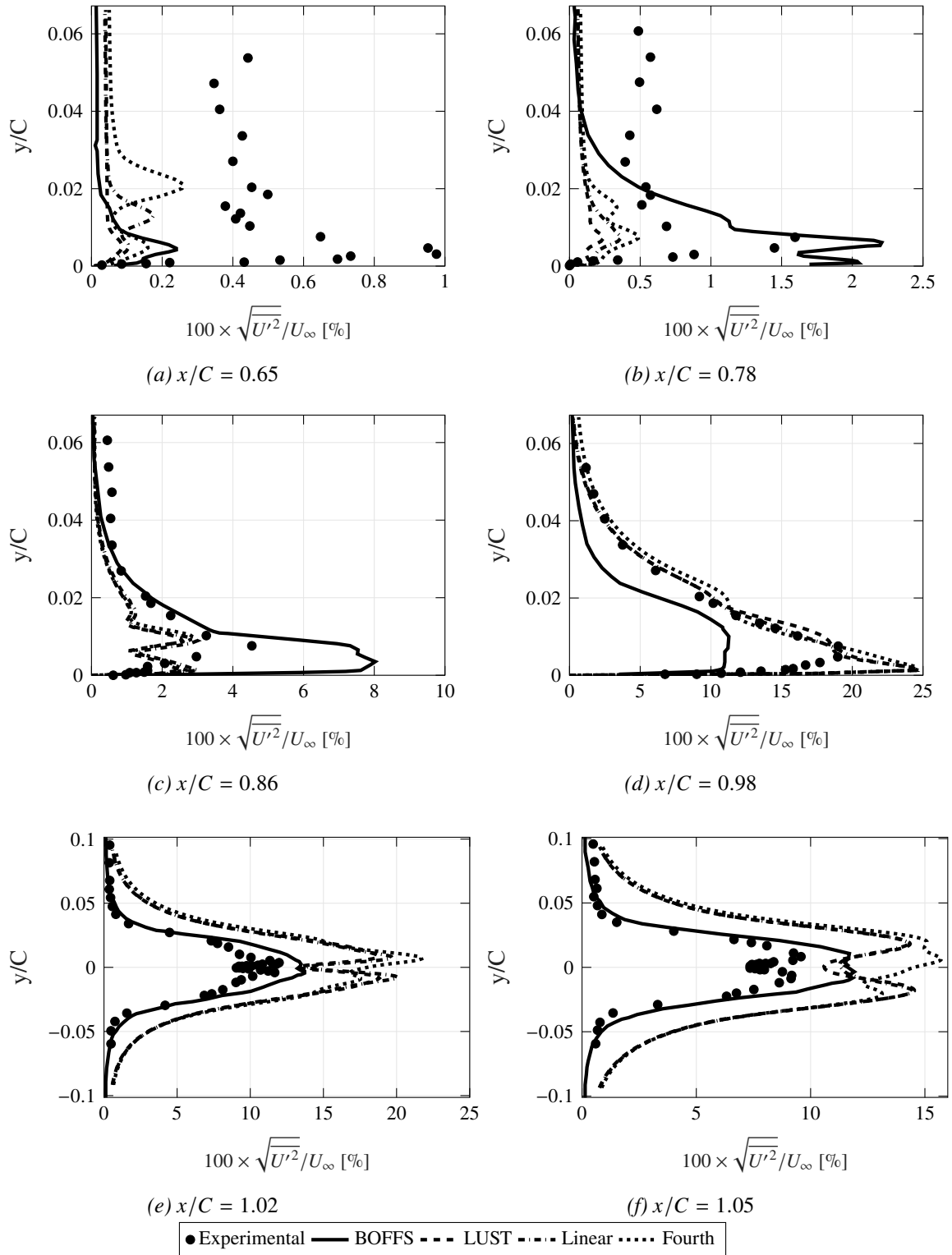


Figure B.3: Normalised streamwise mean velocity profiles for Span = 0.6C



Figure B.4: Turbulence intensity profiles for  $\text{Span} = 0.6C$