Aerodynamic and Aeroacoustic Performance of Morphing Structures

By

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For my beloved mum,
Raikana Begum,
all her pain.
has not gone in vain.
ABSTRACT

The aviation industry is one of the fastest growing industries, with the number of passengers expected to double over the next 20 years. This fast-paced commercial growth has generated a huge demand for highly efficient aircraft with improved aerodynamic and aeroacoustic performance. The aircraft wings are often designed for certain operating conditions, but the advent of morphing structures has enabled better design capabilities to expand their operating condition. In the present study, the aerodynamic and aeroacoustic behavior of morphing structures on two types of airfoils are investigated.

Firstly, experimental and numerical studies of a simple NACA 0012 airfoil fitted with two different flap profiles were successfully carried out to characterize their aerodynamic and aeroacoustic performance. The aerodynamic lift and drag measurements show improved lift-to-drag performance for the morphed flap airfoil compared to the hinged flap airfoil. The improved lift characteristics for the morphing flap airfoil was found to be due to the delayed flow separation observed in the surface flow visualization results. The flow measurement results showed that the downstream wake development can be significantly influenced by the trailing edge flap profile. Particle Image Velocimetry was used to study the flow over the flap and airfoil wake. The mean velocity contours within the airfoil wake region showed increased wake velocity deficit and turbulent kinetic energy for the morphed flap airfoil. The turbulent kinetic energy results displayed a characteristic double peak behavior, which was also the dominant content of the streamwise normal Reynolds shear stress component. Large eddy simulations were also carried out for the standard hinged and morphed airfoils and the results were validated with the experimental measurements. The unsteady flow characteristics were assessed in order to better understand the flow behavior around the morphed flap airfoil. The near-field and far-field acoustic results from the simulations showed that the morphed flap profile generates higher noise levels relative to hinged flap airfoil, which has been attributed to the increase level of surface pressure fluctuations at the trailing edge.

In the second phase of the project, the aerodynamic and aeroacoustic performance of an MDA 30P30N high-lift airfoil, fitted with slat cove fillers were examined experimentally. Measurements included lift and drag performance and mean surface pressure distribution, flow field analysis, near-field surface pressure fluctuations and far-field radiated noise. The flow measurement results show that there is no significant difference in the aerodynamic lift and drag between the standard
30P30N and that fitted with a slat cover filler. However, the slat cove filler configurations exhibit a much better lift-to-drag performance. The pressure coefficient results show that the use of slat cove fillers leads to a slight decrease in the suction peak over the main-element of the airfoil. In order to better understand the flow-field and the noise generation mechanism of the airfoil with slat cove fillers, simultaneous near-field and far-field noise measurements were carried out. The results showed that the use of the slat cove filler can generally lead to a significant reduction of the broadband noise and elimination of the tonal noise generated by the slat. The directivity pattern and the overall sound pressure level of the radiated noise have shown that a significant noise reduction can be achieved with the proper implementation of the slat cove fillers. The multiple tonal phenomena generated by the slat were also analyzed using the continuous wavelet transform method and higher order spectral analysis methods. The research carried out as part of this work has shown the great potential of morphing technologies for aerodynamically efficient and quiet airfoils and provides the impetus for further numerical and experimental work in this area.
First and foremost, I would like to thank my supervisor Dr. Mahdi Azaypeyvand, without him this whole project and this Ph.D. work wouldn’t have materialized into being and without his invaluable support, I couldn’t have successfully completed it. If I ever had an option to hand pick the supervisor that I would want to work with for my Ph.D., it would have been him without a second thought. He is someone who replies emails within few minutes at any time of the day. He is someone whom you can walk-in during his meetings and get a signature off him. He is very dedicated to his work. When it comes Mahdi, let it be the project purchases or negotiating with the school or technicians (uncountable accounts) he has always been there for us and has always taken our side at every instant. He has been completely loyal to the project and his students without expecting anything in return for years. Every plot in this thesis wouldn’t look the way it is without his input, he has always given his valuable input in every aspect of this work. Moreover, he has always patiently added the constantly missing definite articles in all my papers and reports without loosing his temper. Let it be the weekends or bank holidays, let it be 8 am or 10 pm, Mahdi has always come to the wind tunnel labs to check on us and to give his valuable inputs with the experiments (or just cruelly trigger doubts about the whole thing). He has always been a pillar of support to me giving feedbacks and corrections constantly, so this work is as much his as it is mine. I would like to wholeheartedly convey my special appreciation and sincere gratitude to Mahdi for all his support. I would also like to gratefully acknowledge the financial support from Embraer S.A. for this project.

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List of Publications

Some of the research work and results outlined in this thesis were published in peer-reviewed journals or presented at international conferences. These publications serve as a foundation of the thesis and are listed below:

Journal articles:


Conference articles:


declare that the work in this dissertation was carried out in accordance with the requirements of the University’s Regulations and Code of Practice for Research Degree Programmes and that it has not been submitted for any other academic award. Except where indicated by specific reference in the text, the work is the candidate’s own work. Work done in collaboration with, or with the assistance of, others, is indicated as such. Any views expressed in the dissertation are those of the author.

SIGNED: .................................................... DATE: ..........................................

vii
NOMENCLATURE

Roman Symbols

\( b \) flap length, m
\( c \) airfoil chord length, m
\( c_s \) slat chord length, m
\( c_0 \) speed of sound, m/s
\( C \) Courant Friedrichs Lewy number
\( C_D \) drag coefficient
\( C_L \) lift coefficient
\( C_{L,max} \) maximum lift coefficient
\( C_p \) pressure coefficient, \( C_p = (p_i - p_\infty)/(0.5 \rho U_\infty^2) \)
\( C_{pRMS} \) pressure coefficient root mean squared
\( f \) frequency, Hz
\( f_s \) sampling frequency, Hz
\( k \) turbulent kinetic energy, \( 0.5(u'u' + v'v') \), m\(^2\)/s\(^2\)
\( k_c \) Helmholtz number, \( k_c = 2\pi f \cdot c/c_0 \)
\( l \) airfoil span length, m
\( L_x \times L_y \times L_z \) cell dimensions of computational grid
\( M \) Mach number
\( N_T \) number of total records
\( \bar{p} \) average pressure, Pa
\( p' \) fluctuating surface pressure, Pa
\( p_{i} \) static pressure at the \( i^{th} \) location, Pa
\( p_{\infty} \) free-stream static pressure at the \( i^{th} \) location, Pa
\( p_{amb} \) ambient pressure, \((101.3 \times 10^3)\), Pa
\( p_{corr} \) pressure correction, dB
\( p_0 \) stagnation pressure, Pa
\( P_{Ref} \) reference pressure, \((2 \times 10^{-5})\), Pa
\( P_{RMS} \) pressure root mean squared
\( Q \) second invariant of the velocity-gradient tensor, \(1/s^2\)
\( Re_c \) chord-based Reynolds number
\( R_{p_i p_j} \) wall pressure cross-correlation coefficient between transducers \( p_i \) and \( p_j \)
\( R_{p_i p_i} \) wall pressure autocorrelation coefficient of transducers \( p_i \)
\( St_s \) slat chord based Strouhal number, \(St_s = f \cdot c_s/U_{\infty}\)
\( St_n \) mode number
\( SPL \) sound pressure level, dB
\( t_s \) sampling time, s
\( U_{\infty} \) free-stream velocity, m/s
\( U \) mean streamwise velocity, m/s
\( V \) mean crosswise velocity, m/s
\( u', v' \) streamwise and crosswise velocity fluctuations, m/s
\( \overline{u' u'} \) streamwise Reynolds normal stress component, \(m^2/s^2\)
\( \overline{v' v'} \) crosswise Reynolds normal stress component, \(m^2/s^2\)
\( -\overline{u' v'} \) Reynolds shear stress component, \(m^2/s^2\)
\( x, y, z \) streamwise, crosswise and spanwise coordinates, m
\( y^+ \) dimensionless wall distance

**Greek Symbols**
\( \alpha \) angle of attack, \(^{\circ}\)
\( \beta \) flap deflection angle, \(^{\circ}\)
\( \sigma_{boot} \) Bootstrap standard deviation
\( \rho \) air density, kg/m\(^3\)
\( \Delta f \)  frequency resolution, Hz
\( \Delta t \)  time step, s
\( \Delta St_s \)  modulation frequency, Hz
\( \delta_{ij} \)  Kronecker delta function
\( \tau \)  time delay, s
\( \phi_{uu} \)  power spectral density of velocity fluctuations, dB/Hz
\( \phi_{pp} \)  power spectral density of pressure fluctuations, dB/Hz
\( \Phi(p_i, p_i) \)  cross-power spectral density from transducer \( p_i \), Pa\(^2\)/Hz
\( \Phi(p_j, p_j) \)  cross-power spectral density from transducer \( p_j \), Pa\(^2\)/Hz
\( \gamma_{p_i, p_j}^2 \)  wall pressure coherence between transducers \( p_i \) and \( p_j \)
\( \Lambda_y \)  spanwise coherence length, m
\( \lambda_1, \lambda^2, \lambda^3 \)  eigenvalues of the Reynolds stress tensors

**Acronyms**

- **CFD**  Computational Fluid Dynamics
- **CFL**  Courant Friedrichs Lewy
- **CWT**  Continuous Wavelet Transform
- **DES**  Detached Eddy Simulation
- **FCF**  Flap Cove Filler
- **FFT**  Fast Fourier Transform
- **FOAM**  Field Operation and Manipulation
- **FoV**  Field of View
- **FTT**  Flow Through Time
- **HF**  Hinged Flap
- **H-SCF**  Half Slat Cove Filler
- **IW**  Interrogation Window
- **LES**  Large Eddy Simulation
- **MF**  Morphed Flap
- **NACA**  National Advisory Committee for Aeronautics
<table>
<thead>
<tr>
<th>Abbreviation</th>
<th>Description</th>
</tr>
</thead>
<tbody>
<tr>
<td>PISO</td>
<td>Pressure Implicit with Splitting of Operators</td>
</tr>
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<td>PIV</td>
<td>Particle Image Velocimetry</td>
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<tr>
<td>POD</td>
<td>Proper Orthogonal Decomposition</td>
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<tr>
<td>PSD</td>
<td>Power Spectral Density</td>
</tr>
<tr>
<td>OASPL</td>
<td>Over All Sound Pressure Level</td>
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<tr>
<td>RANS</td>
<td>Reynolds-Averaged Navier-Stokes</td>
</tr>
<tr>
<td>SCF</td>
<td>Slat Cove Filler</td>
</tr>
<tr>
<td>SPL</td>
<td>Sound Pressure Level</td>
</tr>
<tr>
<td>SST</td>
<td>Shear Stress Transport</td>
</tr>
<tr>
<td>STD</td>
<td>Standard Deviation</td>
</tr>
<tr>
<td>TKE</td>
<td>Turbulent Kinetic Energy</td>
</tr>
<tr>
<td>URANS</td>
<td>Unsteady Reynolds-Averaged Navier-Stokes</td>
</tr>
<tr>
<td>WS</td>
<td>Window Size</td>
</tr>
</tbody>
</table>
# Table of Contents

List of Tables  
List of Figures  
1 Introduction  
   1.1 Research motivation  
   1.2 Research objectives  
   1.3 Research outline  
   1.4 Thesis outline  
2 Literature Review  
   2.1 Morphing leading and trailing edge  
      2.1.1 Morphing leading edge  
      2.1.2 Morphing trailing edge  
   2.2 High-lift airfoil  
      2.2.1 High-lift airfoil aerodynamics  
      2.2.2 High-lift airfoil noise sources  
   2.3 Summary  
3 Experimental and Computational Setup  
   3.1 Model configurations and instrumentations  
      3.1.1 Wind tunnel facilities  
      3.1.2 NACA 0012 airfoil experimental setup  

# TABLE OF CONTENTS

3.1.3 High-lift airfoil experimental setup ........................................... 37
3.1.4 Slat cove filler design ................................................................. 43

3.2 Measurement techniques ................................................................. 44
3.2.1 Force balance measurement setup ................................................. 44
3.2.2 Pressure measurement setup .......................................................... 46
3.2.3 Particle Image Velocimetry setup ................................................... 47
3.2.4 Oil flow visualization ................................................................. 49
3.2.5 Unsteady surface pressure measurements ......................................... 49
3.2.6 Surface pressure transducer calibration ........................................... 51
3.2.7 Far-field measurement ............................................................... 54
3.2.8 Far-field microphone calibration ..................................................... 56

3.3 Definitions of measurement quantities ................................................. 57
3.4 Computational setup ................................................................. 59
3.5 Summary ................................................................. 62

4 Morphed Trailing Edges ............................................................. 63
4.1 Introduction ................................................................. 63
4.2 Experimental results ................................................................. 66
4.2.1 Aerodynamic force measurements ............................................... 66
4.2.2 Surface flow visualization ............................................................ 68
4.2.3 PIV flow visualization .............................................................. 71
4.2.4 Wake development ................................................................. 82

4.3 Computational Fluid Dynamics ............................................................. 88
4.3.1 Pressure distribution ............................................................... 92
4.3.2 Wake flow development ............................................................. 95
4.3.3 Boundary layer measurements ...................................................... 104
4.3.4 Wall pressure spectra .............................................................. 107
4.3.5 Space-time correlation ............................................................. 113
4.3.6 Wake velocity spectra .............................................................. 117
# TABLE OF CONTENTS

4.3.7 Far-field noise .................................................. 121
4.4 Conclusions ......................................................... 122

5 Slat Cove Filler ......................................................... 125
  5.1 Introduction ......................................................... 125
  5.2 Aerodynamic results ................................................. 128
    5.2.1 Aerodynamic force measurements ..................................... 129
    5.2.2 Pressure coefficient distribution ..................................... 131
  5.3 Flow field analysis ................................................ 135
    5.3.1 Flow field visualization ............................................. 135
    5.3.2 Slat wake development .............................................. 142
    5.3.3 Proper Orthogonal Decomposition .................................... 154
  5.4 Aeroacoustic results ................................................ 169
    5.4.1 Far-field spectral levels ............................................ 171
    5.4.2 Near-field spectral levels .......................................... 174
    5.4.3 Spanwise coherence ................................................. 177
    5.4.4 Continuous Wavelet Transform ...................................... 182
    5.4.5 Higher order spectral analysis ...................................... 186
    5.4.6 Persistence spectrum ............................................... 189
  5.5 Conclusions ......................................................... 190

6 Conclusions and future work ........................................ 193
  6.1 Research contribution .............................................. 197
  6.2 Future work ......................................................... 198

Bibliography .......................................................... 201

Appendices ............................................................. 217
  A Slat cove filler design ............................................... 217
    A.1 Computational setup .................................................. 217
# TABLE OF CONTENTS

A.1.1 Angle of attack of 3 degrees ............................................. 220  
A.1.2 Angle of attack of 5.5 degrees ......................................... 222  
A.1.3 Angle of attack of 8.5 degrees ......................................... 225  

B Acoustic analogies .......................................................... 229  
B.1 Lighthill’s acoustic analogy ............................................. 229  
B.2 Curle’s acoustic analogy .................................................. 231
# List of Tables

<table>
<thead>
<tr>
<th>Table</th>
<th>Page</th>
</tr>
</thead>
<tbody>
<tr>
<td>2.1</td>
<td>Typical noise source range presented in terms of slat based Strouhal number.</td>
</tr>
<tr>
<td>3.1</td>
<td>Geometrical parameters in percentage of stowed airfoil chord, $c = 0.35$ m.</td>
</tr>
<tr>
<td>3.2</td>
<td>Static pressure taps locations along the mid-span location of the MDA 30P30N airfoil model with a retracted chord length of $c = 0.35$ m.</td>
</tr>
<tr>
<td>3.3</td>
<td>Microphone locations on the MDA 30P30N airfoil.</td>
</tr>
<tr>
<td>3.4</td>
<td>The PIV setup parameters used in the current study.</td>
</tr>
<tr>
<td>4.1</td>
<td>The overall sound pressure level at observer point 1.2 m ($x/c = 1, y/c = 6$) above the trailing edge for Hinged Flap and Morphed Flap.</td>
</tr>
<tr>
<td>5.1</td>
<td>Slat trailing-edge wake measurement locations.</td>
</tr>
<tr>
<td>5.2</td>
<td>The number of resolved POD modes of the vorticity that contains 90% of the systems energy for each configuration.</td>
</tr>
<tr>
<td>5.3</td>
<td>Parameters used for tonal peak frequency prediction.</td>
</tr>
<tr>
<td>5.4</td>
<td>Pressure transducer locations on the MDA 30P30N airfoil.</td>
</tr>
<tr>
<td>5.5</td>
<td>The narrow-band frequencies observed for the Baseline case in the near-field and far-field measurements at angles of attack $\alpha = 14^\circ$ and $18^\circ$ and the labels in Figs. 5.41 and 5.44 are detailed.</td>
</tr>
</tbody>
</table>
## List of Figures

<table>
<thead>
<tr>
<th>Figure</th>
<th>Page</th>
</tr>
</thead>
<tbody>
<tr>
<td>1.1 Aircraft that successfully used span morphing wing over the years [3].</td>
<td>2</td>
</tr>
<tr>
<td>2.1 Flight control surfaces on an Airbus A380 [13].</td>
<td>9</td>
</tr>
<tr>
<td>2.2 SADE morphing leading edge with kinematic chain interior and elastic skin [19].</td>
<td>11</td>
</tr>
<tr>
<td>2.3 Deployed and retracted configuration of the slat cove filler structure [21].</td>
<td>11</td>
</tr>
<tr>
<td>2.4 Chain link trailing edge concept [4, 14].</td>
<td>12</td>
</tr>
<tr>
<td>2.5 The composite electrical trailing edge concept deveoped for Blended Wing Body aircraft design, where $M$ is the bending moment [22].</td>
<td>13</td>
</tr>
<tr>
<td>2.6 (a) Wortmann FX63-137 airfoil installed in the windtunnel and (b) lift coefficient for inlet velocity of 20 m/s and flap deflection of 20° [28].</td>
<td>15</td>
</tr>
<tr>
<td>2.7 The composite electrical trailing edge concept deveoped for Blended Wing Body aircraft design, where $M$ is the bending moment [29].</td>
<td>16</td>
</tr>
<tr>
<td>2.8 The composite electrical trailing edge concept deveoped for Blended Wing Body aircraft design, where $M$ is the bending moment [22].</td>
<td>17</td>
</tr>
<tr>
<td>2.9 Effects of slat and flap on an airfoil [36].</td>
<td>20</td>
</tr>
<tr>
<td>2.10 Typical flow around a high-lift airfoil [39].</td>
<td>20</td>
</tr>
<tr>
<td>2.11 Potential slat noise sources and flow field adopted from Choudhari and Khorrami [51].</td>
<td>27</td>
</tr>
<tr>
<td>2.12 The geometry of the slat cove cover (left) and the measured far-field noise (right) [50].</td>
<td>28</td>
</tr>
<tr>
<td>2.13 Slat noise reduction using slat extension and bulb seal (left) and the measured far-field noise reduction (right) [56].</td>
<td>28</td>
</tr>
<tr>
<td>2.14 Slat cove filler (left) and measured far-field noise reduction on full scale wing (right) [133].</td>
<td>29</td>
</tr>
</tbody>
</table>
LIST OF FIGURES

3.1 NACA 0012 airfoil model with the interchangeable trailing edge. ............... 35
3.2 NACA 0012 airfoil setup in the low-turbulence wind tunnel. .......................... 35
3.3 Geometric details of the NACA 0012 airfoil with a flap deflection angle of $\beta = 10^\circ$ named Hinged Flap and Morphed Flap airfoils. .............................. 35
3.4 NACA 0012 airfoil with side-plates setup in the large low-speed closed-circuit wind tunnel. ................................................................. 36
3.5 The camera window locations used for the PIV measurements of the NACA 0012 airfoil. 36
3.6 Exploded (top) and assembled (bottom) view of the manufacture MDA 30P30N airfoil model. ................................................................. 37
3.7 30P30N three-element airfoil geometric parameters. ................................. 38
3.8 Slat close up view of the 3D printed interchangeable leading edge. ............... 39
3.9 Static pressure taps and surface pressure transducer location on the MDA 30P30N airfoil with span of $l = 0.53$ m and a retracted chord of $c = 0.35$ m. ............... 40
3.10 The camera window locations used for PIV measurements of the 30P30N high-lift airfoil. 42
3.11 Reduced surface reflection during PIV by the use of self adhesive black vinyl sheet. 42
3.12 (a) Turbulent kinetic energy contours indicating slat shear layer profiles around 30P30N airfoil slat for an angle of attack, $\alpha = 8.5^\circ$ at $Re_c = 1.7 \times 10^6$ and (b) The 3D printed SCF fitted on the 30P30N airfoil in the low turbulence wind tunnel .......... 43
3.13 MDA 30P30N Baseline airfoil fitted with half-slat cove filler (H-SCF), slat cove filler (SCF) and flap cove filler (FCF). ................................................. 44
3.14 Probability density function of bootstrap standard deviation for the force platform signal, where $\sigma_{\text{boot}}$ is the bootstrap standard deviation. ................................. 45
3.15 AMTI OR6-7-2000 force balance used in the current experiments. ................. 46
3.16 MicroDaq pressure scanner used for static pressure measurement. ................. 47
3.17 The PIV measurement setup (a) showing the camera and model setup and (b) a close-up view of the laser sheet and illuminated particles. ................. 48
3.18 Pressure transducer used for unsteady surface pressure measurements, (a) Close up view of the stock FG-3329-P07 pressure transducer from Knowles Electronics, (b) FG-3329-P07 covered with the 3D printed surface fairing, (c) FG-3329-P07 beside the surface fairing and a scale for size comparison and (d) schematic of the surface fairing dimensions ................................................................. 50

3.19 Surface mounted FG-3329-P07 pressure transducer and pressure taps on the main-element and slat of the 30P30N high-lift airfoil, respectively ............................................. 50

3.20 FG-3329-P07 calibration setup used for in situ calibration .................................... 51

3.21 Schematic of the FG-3329-P07 calibration procedure [90, 91] ............................... 52

3.22 The broadband sensitivity of the surface mounted FG surface mounted FG-3329-P07 transducer fitted with surface fairing ............................................................. 53

3.23 Test model mounted in the aeroacoustic wind tunnel at the University of Bristol ... 55

3.24 GRAS 40PL far-field microphone setup ............................................................... 55

3.25 (a) GRAS 40PL calibration device and (b) GRAS frequency response spectra where solid line is the free field response and dashed line is the pressure response .......... 57

3.26 Large Eddy Simulation with scale separation [104] ................................................ 59

3.27 An overview of the LES computational domain and setup with a close up view of the Morphed Flap airfoil trailing edge mesh ...................................................... 61

3.28 Grid refinement close to the airfoil wall and the wake regions to capture the boundary layer transition over the flaps accurately ......................................................... 61

4.1 Geometric details of the NACA 0012 airfoil with a flap deflection angle of $\beta = 10^\circ$ named Hinged Flap and Morphed Flap airfoils .................................................... 66

4.2 Lift and drag coefficient results for the Hinged Flap and Morphed Flap airfoil at a free-stream velocity of $U_\infty = 20$ m/s ($Re_c = 2.6 \times 10^5$) ..................................... 67

4.3 Lift-to-drag coefficient ratio results and the drag polar plots for the Hinged Flap and Morphed Flap airfoil at a free-stream velocity of $U_\infty = 20$ m/s ($Re_c = 2.6 \times 10^5$) .... 68
4.4 The photographs of the oil-flow visualization patterns over the suction side of the Hinged Flap and Morphed Flap airfoil at the vicinity of the flap tested at a free-stream velocity of $U_\infty = 20$ m/s ($Re_c = 2.6 \times 10^5$) and $\cdots$ denoting the flow separation location. 70

4.5 The mean streamwise velocity contours from PIV for the Hinged Flap and Morphed Flap airfoils. 76

4.6 The mean crosswise velocity contours from PIV for the Hinged Flap and Morphed Flap airfoils. 77

4.7 The normalised turbulent kinetic energy contours for the Hinged Flap and Morphed Flap airfoils at various angles of attack for a free-stream velocity of $U_\infty = 20$ m/s ($Re_c = 2.6 \times 10^5$). 78

4.8 The normalised streamwise Reynolds normal stress contours for the Hinged Flap and Morphed Flap airfoils at various angles of attack for a free-stream velocity of $U_\infty = 20$ m/s ($Re_c = 2.6 \times 10^5$). 79

4.9 The normalised crosswise Reynolds normal stress contours for the Hinged Flap and Morphed Flap airfoils at various angles of attack for a free-stream velocity of $U_\infty = 20$ m/s ($Re_c = 2.6 \times 10^5$). 80

4.10 The normalised Reynolds shear stress contours for the Hinged Flap and Morphed Flap airfoils at various angles of attack for a free-stream velocity of $U_\infty = 20$ m/s ($Re_c = 2.6 \times 10^5$). 81

4.11 Airfoil coordinate system along with the data extraction locations in the wake region. 82

4.12 The non-dimensional mean velocity profiles (a), turbulence kinetic energy (b), and Reynolds stress tensor (c), (d) and (e) for the Hinged Flap — and Morphed Flap $\cdots$ airfoils at angles of attack $\alpha = 0^\circ$. 85

4.13 The non-dimensional mean velocity profiles (a), turbulence kinetic energy (b), and Reynolds stress tensor (c), (d) and (e) for the Hinged Flap — and Morphed Flap $\cdots$ airfoils at angles of attack $\alpha = 4^\circ$. 86

4.14 The non-dimensional mean velocity profiles (a), turbulence kinetic energy (b), and Reynolds stress tensor (c), (d) and (e) for the Hinged Flap — and Morphed Flap $\cdots$ airfoils at angles of attack $\alpha = 8^\circ$. 87
4.15 The validation from Cesar et al. [69] for (a) Pressure coefficient (LES —, Exp ◦) (b) boundary layer momentum thickness (LES ⊲, Exp □) and displacement thickness (LES ◆, Exp ◆) [79]. ................................. 89

4.16 The validation from [69] for the mean velocity profiles at various streamwise locations on the boundary layer of the NACA 0012 baseline airfoil for LES — and Exp ◆ [79]. ................................. 89

4.17 The validation from Cesar et al. [69] for the mean wake profile at the vicinity of the trailing-edge of the NACA 0012 baseline airfoil for LES — and Exp ◆ [79]. ................................. 89

4.18 The validation from Cesar et al. [69] for (a) Streamwise normal Reynolds stresses $\overline{u'u'}$ and (b) crosswise normal Reynolds stresses $\overline{v'v'}$ for LES $x/c = 0.80$ —, $x/c = 0.90$ —, $x/c = 0.98$ — and Exp $x/c = 0.80$ ◆, $x/c = 0.90$ □, $x/c = 0.98$ *. ................................. 90

4.19 The validation from Cesar et al. [69] for the wall-pressure power spectral density with $p_{Re}f = 2 \times 10^5$ Pa at various streamwise locations. (a) LES ($x/c =0.80$) —,($x/c=0.90$) —, and ($x/c=0.98$) —, (b) Experiments ($x/c =0.80$) black cross, ($x/c=0.90$) red circle, and ($x/c=0.98$) blue asterisk. ................................. 90

4.20 Wake velocity profiles for the experimental measurements and LES at angles of attack $\alpha = 0^\circ$ and $4^\circ$ at free-stream velocity of $U_\infty = 20$ m/s corresponding to a chord-based Reynolds number of $Re_c = 2.6 \times 10^5$. ................................. 91

4.21 Pressure coefficient and pressure coefficient root mean squared over the airfoil surface for Hinged Flap and Morphed Flap airfoils at angles of attack $\alpha = 0^\circ$ and $4^\circ$ at free-stream velocity of $U_\infty = 20$ m/s ($Re_c = 2.6 \times 10^5$). ................................. 94

4.22 Pressure coefficient contours root mean squared for the Hinged Flap and Morphed Flap airfoils at angles of attack $\alpha = 0^\circ$ and $4^\circ$ at free stream velocity of $U_\infty = 20$ m/s ($Re_c = 2.6 \times 10^5$). ................................. 94

4.23 Iso-surfaces of $Q$-criterion of $Q = 1 \times 10^6 s^{-2}$ for Hinged Flap and Morphed Flap airfoil with contours of vorticity magnitude at angles of attack $\alpha = 0^\circ$ and $4^\circ$ for a free-stream velocity of $U_\infty = 20$ m/s ($Re_c = 2.6 \times 10^5$). ................................. 96

4.24 The normalised instantaneous streamwise velocity contours from the LES flow for the Hinged Flap and Morphed Flap airfoils at angles of attack $\alpha = 0^\circ$ and $4^\circ$ for a free-stream velocity of $U_\infty = 20$ m/s ($Re_c = 2.6 \times 10^5$). ................................. 96
4.25 The normalised mean streamwise velocity contours for the LES flow field at mid span for the Hinged Flap and Morphed Flap airfoils at angles of attack $\alpha = 0^\circ$ and $4^\circ$ for a free-stream velocity of $U_\infty = 20$ m/s ($Re_c = 2.6 \times 10^5$). ........................................ 97

4.26 The normalised mean crosswise velocity contours for the LES flow field at mid span for the Hinged Flap and Morphed Flap airfoils at angles of attack $\alpha = 0^\circ$ and $4^\circ$ for a free-stream velocity of $U_\infty = 20$ m/s ($Re_c = 2.6 \times 10^5$). ........................................ 98

4.27 The non-dimensional turbulent kinetic energy contours for the LES flow field at mid span for the Hinged Flap and Morphed Flap airfoils at angles of attack $\alpha = 0^\circ$ and $4^\circ$ for a free-stream velocity of $U_\infty = 20$ m/s ($Re_c = 2.6 \times 10^5$). .......................... 100

4.28 The non-dimensional streamwise Reynolds normal stress contours for the LES flow field at mid span for the Hinged Flap and Morphed Flap airfoils at angles of attack $\alpha = 0^\circ$ and $4^\circ$ for a free-stream velocity of $U_\infty = 20$ m/s ($Re_c = 2.6 \times 10^5$). ......................... 100

4.29 The non-dimensional crosswise Reynolds normal stress contours for the LES flow field at mid span for the Hinged Flap and Morphed Flap airfoils at angles of attack $\alpha = 0^\circ$ and $4^\circ$ for a free-stream velocity of $U_\infty = 20$ m/s ($Re_c = 2.6 \times 10^5$). ......................... 101

4.30 The non-dimensional Reynolds shear stress contours for the LES flow field at mid span for the Hinged Flap and Morphed Flap airfoils at angles of attack $\alpha = 0^\circ$ and $4^\circ$ for a free-stream velocity of $U_\infty = 20$ m/s ($Re_c = 2.6 \times 10^5$). ......................... 101

4.31 Boundary layer velocity profiles on the suction and pressure side at various streamwise locations of the Hinged Flap and Morphed Flap airfoils at angle of attack $\alpha = 0^\circ$ for a free-stream velocity of $U_\infty = 20$ m/s ($Re_c = 2.6 \times 10^5$). ......................... 105

4.32 Boundary layer velocity profiles on the suction and pressure side at various streamwise locations of the Hinged Flap and Morphed Flap airfoils at angle of attack $\alpha = 4^\circ$, for a free-stream velocity of $U_\infty = 20$ m/s ($Re_c = 2.6 \times 10^5$). ......................... 106

4.33 Evolution of the wall-pressure spectra along the airfoil chord on the suction side with the beginning of the flap $x = 0.70c$ indicated by — and downstream locations $x = 0.75c$ and $x = 0.95c$ indicated by .... and ---, respectively. ......................... 108
4.34 Wall-pressure spectra normalised by the reference pressure $p_{Ref} = 2 \times 10^{-5}$ Pa, on the suction side at different streamwise locations for Hinged Flap and Morphed Flap airfoils at angles of attack $\alpha = 0^\circ$ and $4^\circ$. ................................. 109

4.35 Spanwise coherence of the surface pressure on the suction side, at the location $x/c = 0.95$ for Hinged Flap and Morphed Flap airfoils at angles of attack $\alpha = 0^\circ$ and $4^\circ$.................... 110

4.36 Spanwise coherence of the surface pressure on the suction side, at the location $x/c = 0.95$ for Hinged Flap (solid line) and Morphed Flap (dashed line) airfoils at angles of attack $\alpha = 0^\circ$ and $4^\circ$ for $k_c = 1$ (circles) $k_c = 10$ (asterisk) and $k_c = 20$ (triangle). ............ 111

4.37 Spanwise coherence length scales for the surface pressure on the suction side at the location $x/c = 0.95$ for Hinged Flap and Morphed Flap airfoils at angles of attack $\alpha = 0^\circ$ and $4^\circ$....................................................... 111

4.38 Auto and cross-correlation of the surface pressure fluctuation at various chord locations ($x/c$), on the suction side for Hinged Flap (solid line) and Morphed Flap (dashed line) airfoils at angles of attack $\alpha = 0^\circ$ and $4^\circ$ for probe order moving upstream of the airfoil trailing edge location $x/c = 0.99$. ....................................................... 115

4.39 Auto and cross-correlation of the surface pressure fluctuation at various chord locations ($x/c$), on the suction side for Hinged Flap (solid line) and Morphed Flap (dashed line) airfoils at angles of attack $\alpha = 0^\circ$ and $4^\circ$ for the probe order moving downstream of the flap $x/c = 0.72$. ....................................................... 116

4.40 Wake streamwise velocity spectra at location $x/c = 1.01$ close to the trailing edge for Hinged Flap and Morphed Flap airfoils at angles of attack $\alpha = 0^\circ$ and $4^\circ$....................... 119

4.41 Difference in wake streamwise velocity spectra at location $x/c = 1.01$ for the angles of attack $\alpha = 0^\circ$ and $4^\circ$....................................................... 119

4.42 Wake streamwise velocity spectra at location $x/c = 1.01$ close to the trailing edge at various crosswise locations for Hinged Flap and Morphed Flap airfoils at angles of attack $\alpha = 0^\circ$ and $4^\circ$....................................................... 120

4.43 Acoustic prediction using Curle’s analogy, sound pressure level in dB reference to $p_{Ref} = 2 \times 10^{-5}$ Pa, at observer point $1.2$ m ($x/c = 1, y/c = 6$) above the trailing edge for Hinged Flap and Morphed Flap....................................................... 121
5.1 The MDA 30P30N Baseline airfoil fitted with half-slat cove filler (H-SCF), slat cove filler (SCF) and flap cove filler (FCF). ................................................................. 128
5.2 Lift and drag coefficients for the 30P30N airfoil with various cove fillers at chord-based Reynolds number \( Re_c = 9.3 \times 10^5 \) ................................................................. 129
5.3 Lift-to-drag ratio and the drag polar plots for the 30P30N airfoil with various cove fillers at chord-based Reynolds number \( Re_c = 9.3 \times 10^5 \) .............................. 130
5.4 Pressure coefficient distribution over 30P30N Baseline airfoil for various chord-based Reynolds numbers at angle of attack \( \alpha = 12^\circ \) ........................................ 131
5.5 Pressure coefficient distribution over 30P30N Baseline airfoil for various angles of attack at a free-stream velocity of \( U_\infty = 30 \text{ m/s}, Re_c = 7.0 \times 10^5 \) [129] .......................... 132
5.6 Pressure coefficient distribution over 30P30N Baseline airfoil around the slat and flap region for a free-stream velocity of \( U_\infty = 30 \text{ m/s}, Re_c = 7.0 \times 10^5 \) ........................................ 132
5.7 Pressure coefficient distribution over 30P30N airfoil with slat modifications, at various angles of attack for a free-stream velocity of \( U_\infty = 30 \text{ m/s}, Re_c = 7.0 \times 10^5 \) .......................... 133
5.8 Pressure coefficient distribution over 30P30N airfoil with slat modifications, at various angles of attack for a free-stream velocity of \( U_\infty = 30 \text{ m/s}, Re_c = 7.0 \times 10^5 \) ........................................ 134
5.9 The normalised mean streamwise velocity \( (U/U_\infty) \) contours around the slat region for various angles of attack with a free-stream velocity of \( U_\infty = 30 \text{ m/s}, Re_c = 7.0 \times 10^5 \) .......................... 138
5.10 The normalised mean crosswise velocity \( (V/U_\infty) \) contours around the slat region for various angles of attack with a free-stream velocity of \( U_\infty = 30 \text{ m/s}, Re_c = 7.0 \times 10^5 \) ........................................ 139
5.11 The normalised streamwise Reynnolds normal stress \( (u'u'/U_\infty^2) \) contours around the slat region for various angles of attack with a free-stream velocity of \( U_\infty = 30 \text{ m/s}, Re_c = 7.0 \times 10^5 \) .......................... 140
5.12 The normalised crosswise Reynnolds normal stress \( (v'v'/U_\infty^2) \) contours around the slat region for various angles of attack with a free-stream velocity of \( U_\infty = 30 \text{ m/s}, Re_c = 7.0 \times 10^5 \) .......................... 141
5.13 Boundary layer measurement locations for the MDA 30P-30N airfoil ................................................................................................................. 142
5.14 Mean velocity profiles over the MDA 30P30N airfoil at various streamwise locations for the free-stream velocity \( U_\infty = 30 \text{ m/s}, \) for Baseline —, H-SCF —— and SCF ---- 143
L IST

OF

F IGURES

5.15 Mean velocity and turbulent kinetic energy profiles at the slat wake for α = 6◦ at the
free-stream velocity U∞ = 30 m/s, for Baseline —, H-SCF – – – and SCF – · –. . . . . . 146
5.16 Reynolds stress tensor profiles at the slat wake for α = 6◦ at the free-stream velocity
U∞ = 30 m/s, for Baseline —, H-SCF – – – and SCF – · –. . . . . . . . . . . . . . . . . . . 147
5.17 Mean velocity and turbulent kinetic energy profiles at the slat wake for α = 8◦ at the
free-stream velocity U∞ = 30 m/s, for Baseline —, H-SCF – – – and SCF – · –. . . . . . 148
5.18 Reynolds stress tensor profiles at the slat wake for α = 8◦ at the free-stream velocity
U∞ = 30 m/s, for Baseline —, H-SCF – – – and SCF – · –. . . . . . . . . . . . . . . . . . . 149
5.19 Mean velocity and turbulent kinetic energy profiles at the slat wake for α = 10◦ at the
free-stream velocity U∞ = 30 m/s, for Baseline —, H-SCF – – – and SCF – · –. . . . . . 150
5.20 Reynolds stress tensor profiles at the slat wake for α = 10◦ at the free-stream velocity
U∞ = 30 m/s, for Baseline —, H-SCF – – – and SCF – · –. . . . . . . . . . . . . . . . . . . 151
5.21 Mean velocity and turbulent kinetic energy profiles at the slat wake for α = 12◦ at the
free-stream velocity U∞ = 30 m/s, for Baseline —, H-SCF – – – and SCF – · –. . . . . . 152
5.22 Reynolds stress tensor profiles at the slat wake for α = 12◦ at the free-stream velocity
U∞ = 30 m/s, for Baseline —, H-SCF – – – and SCF – · –. . . . . . . . . . . . . . . . . . . 153
5.23 The normalised eigenvalue distribution of the first 12 POD modes of the vorticity
within the slat cove region and at the slat wake for angles of attack α = 6◦ (a,b), α = 8◦
(c,d), α = 10◦ (e,f) and 12◦ (g,h). . . . . . . . . . . . . . . . . . . . . . . . . . . . . . . . . . 158
5.24 The vorticity component of the first 4 POD eigemodes within the slat cove region for
α = 6◦ with a free-stream velocity of U∞ = 30 m/s, R e c = 7.0 × 105 . . . . . . . . . . . . . 159

5.25 The eigenvalue mode coefficient of the first 2 POD mode within the slat cove region
for angles of attack α = 6◦ . . . . . . . . . . . . . . . . . . . . . . . . . . . . . . . . . . . . . 159
5.26 The vorticity component of the first 4 POD eigemodes within the slat cove region for
α = 8◦ with a free-stream velocity of U∞ = 30 m/s, R e c = 7.0 × 105 . . . . . . . . . . . . . 160

5.27 The eigenvalue mode coefficient of the first 2 POD mode within the slat cove region
for angles of attack α = 8◦ . . . . . . . . . . . . . . . . . . . . . . . . . . . . . . . . . . . . . 160
5.28 The vorticity component of the first 4 POD eigemodes within the slat cove region for
α = 10◦ with a free-stream velocity of U∞ = 30 m/s, R e c = 7.0 × 105 . . . . . . . . . . . . 161

xxvii


5.29 The eigenvalue mode coefficient of the first 2 POD mode within the slat cove region for angles of attack $\alpha = 10^\circ$. .......................................................... 161

5.30 The vorticity component of the first 4 POD eigemodes within the slat cove region for $\alpha = 12^\circ$ with a free-stream velocity of $U_\infty = 30$ m/s, $Re_c = 7.0 \times 10^5$. ................................. 162

5.31 The eigenvalue mode coefficient of the first 2 POD mode within the slat cove region for angles of attack $\alpha = 12^\circ$. .......................................................... 162

5.32 The vorticity component of the first 4 POD eigemodes at the slat wake region for $\alpha = 6^\circ$ with a free-stream velocity of $U_\infty = 30$ m/s, $Re_c = 7.0 \times 10^5$. ................................. 163

5.33 The eigenvalue mode coefficient of the first 2 POD mode within the slat wake region for angles of attack $\alpha = 6^\circ$. .......................................................... 163

5.34 The vorticity component of the first 4 POD eigemodes at the slat wake region for $\alpha = 8^\circ$ with a free-stream velocity of $U_\infty = 30$ m/s, $Re_c = 7.0 \times 10^5$. ................................. 164

5.35 The eigenvalue mode coefficient of the first 2 POD mode within the slat cove region for angles of attack $\alpha = 8^\circ$. .......................................................... 164

5.36 The vorticity component of the first 4 POD eigemodes at the slat wake region for $\alpha = 10^\circ$ with a free-stream velocity of $U_\infty = 30$ m/s, $Re_c = 7.0 \times 10^5$. ................................. 165

5.37 The eigenvalue mode coefficient of the first 2 POD mode within the slat cove region for angles of attack $\alpha = 10^\circ$. .......................................................... 165

5.38 The vorticity component of the first 4 POD eigemodes at the slat wake region for $\alpha = 12^\circ$ with a free-stream velocity of $U_\infty = 30$ m/s, $Re_c = 7.0 \times 10^5$. ................................. 166

5.39 The eigenvalue mode coefficient of the first 2 POD mode within the slat cove region for angles of attack $\alpha = 12^\circ$. .......................................................... 166

5.40 Simplified schematic of the tonal frequency prediction model by Terracol et al. [124]. 170

5.41 Far-field noise spectra for microphone at 90° and 1.75 m above the slat trailing edge for Baseline — , H-SCF — — , SCF — — and Background noise ····. The resonance modes are listed in Table 5.5. .......................................................... 171

5.42 Directivity for the different configurations at different slat based Strouhal number, for Baseline squareblack, H-SCF triangleblue and SCF ored. .......................................................... 173
5.43 Overall sound pressure level calculated from the far-field microphones, for Baseline squareblack, H-SCF triangleblue and SCF ored. .............................. 173

5.44 Near-field noise spectra for the surface transducer M1 \((x = 22.414\, \text{mm})\) for Baseline —, H-SCF ———, SCF — and Background ..... The associated modes \(St_n\) are listed in Table 5.5. ............................................................. 175

5.45 Coherence between the reference transducer M1 and the other spanwise transducers M2-M5 (see Table 5.4), for Baseline —, H-SCF ——— and SCF ———. .................................................. 179

5.46 Spanwise coherence length scales based on the unsteady surface pressure measurement for Baseline —, H-SCF ——— and SCF ———. .................................................. 180

5.47 Auto-correlation of the surface pressure fluctuations at the near-field transducer location M1 for Baseline —, H-SCF ——— and SCF ———. .................................................. 181

5.48 The coherence between the reference near-field surface pressure transducer M1 and the far-field microphone \(90^\circ\) above the trailing edge for Baseline —, H-SCF ——— and SCF ———. .................................................. 181

5.49 The contours of the wavelet coefficient magnitude \((|W_x^2|)\) for the near-field pressure transducer M1 calculated using Morlet wavelet function. ......................... 184

5.50 Power spectral density of time signal and wavelet coefficient at selected resonance frequencies \((St_n)\) from Table 5.5 for the Baseline case. .................................................. 185

5.51 The auto-bicoherence contour for the near-field pressure transducer M1 on the main-element for the angle of attack \(\alpha = 18^\circ\) labelled with the associated modes \((St_n)\) for the Baseline case detailed in Table 5.5. .................................................. 188

5.52 The persistence spectrum contour for the near-field pressure transducer M1 on the main-element at angle of attack \(\alpha = 18^\circ\). .................................................. 190

A.1 Dense grid around 30P30N airfoil. .................................................. 218

A.2 Dense grid around slat (left) and flap (right). .................................................. 218

A.3 Full domain size used for gridding and simulation. .................................................. 219

A.4 Mean surface pressure distribution around 30P30N airfoil with angle of attack, \(\alpha = 3^\circ\) and \(Re_c = 1.7 \times 10^6\) compared to experiments by Murayama et al. [70]. .................................................. 220

xxix
A.5 Slat (left) and flap (right) mean surface pressure distribution for 30P30N airfoil with angle of attack, $\alpha = 3^\circ$ and $Re_c = 1.7 \times 10^6$ compared to experiments by Murayama et al. [70] .................................................. 220

A.6 Mean velocity contour around 30P30N airfoil with angle of attack, $\alpha = 3^\circ$. ................................. 221

A.7 Slat (left) and flap (right) mean velocity distribution for 30P30N airfoil with angle of attack, $\alpha = 3^\circ$ and $Re_c = 1.7 \times 10^6$ ............................................................. 221

A.8 Mean surface pressure distribution around 30P30N airfoil for angle of attack, $\alpha = 5.5^\circ$ and $Re_c = 1.7 \times 10^6$ compared to experiments by Murayama et al. [70] .............................. 222

A.9 Slat (left) and Flap (right) mean surface pressure distribution for angle of attack, $\alpha = 5.5^\circ$ and $Re_c = 1.7 \times 10^6$ compared to experiments by Murayama et al. [70] .............................. 222

A.10 Mean velocity contour around 30P30N airfoil with angle of attack, $\alpha = 5.5^\circ$ ............................. 223

A.11 Mean velocity distribution (left) and turbulent kinetic energy (right) for 30P30N airfoil slat with angle of attack, $\alpha = 5.5^\circ$ and $Re_c = 1.7 \times 10^6$ .......................... 223

A.12 Mean velocity distribution (left) and turbulent kinetic energy (right) for 30P30N airfoil flap for angle of attack, $\alpha = 5.5^\circ$ and $Re_c = 1.7 \times 10^6$ .......................... 224

A.13 Mean surface pressure distribution around 30P30N airfoil with angle of attack, $\alpha = 8.5^\circ$ and $Re_c = 1.7 \times 10^6$ compared to experiments by Murayama et al. [70] .......................... 225

A.14 Slat (left) and flap (right) mean surface pressure distribution for 30P30N airfoil with angle of attack, $\alpha = 8.5^\circ$ and $Re_c = 1.7 \times 10^6$ compared to experiments by Murayama et al. [70] .......................... 225

A.15 Mean velocity contour around 30P30N airfoil with angle of attack, $\alpha = 8.5^\circ$. ................................. 225

A.16 Slat (left) and flap (right) mean velocity distribution for 30P30N airfoil with angle of attack, $\alpha = 8.5^\circ$ and $Re_c = 1.7 \times 10^6$ ............................................................. 226

A.17 (a) Schematic of the manually extracted shear layer path using contours of turbulent kinetic energy around 30P30N airfoil slat for an angle of attack, $\alpha = 8.5^\circ$ at $Re_c = 1.7 \times 10^6$ and (b) The 3D printed SCF fitted on the 30P30N airfoil in the low turbulence wind tunnel .......................................................... 227
1.1 Research motivation

Aviation industry originated with the evolution of hot air balloons in the 18th century. Now, the aviation industry affords the worldwide transportation that is indispensable for universal trade and travel. It exerts a pre-eminent role in assisting economic growth, predominantly in a country's growth. It is one of the masters on the far side of globalization. The air transport industry produces around 29 million jobs globally and it is set to double by 2035 [1]. To satisfy the growing demand of the aviation industry, there is a need for new aircraft for the constantly growing passengers numbers who travel by air every year. This generates a demand for highly efficient aircrafts with better aerodynamic performance than the ones in operation.

Wind turbines are one of the cleanest forms of renewable energy that is contributing to power generation. In the present-day industrial developments, wind power has accomplished extraordinary advances. Meanwhile, the progress in aerodynamic performance, micrometeorology and the development in structural dynamics have added an annual gain of 5% in the production of energy in wind turbines all over the world [2]. Wind energy is a source of mature renewable energy which has great potential to a foremost source of energy in forthcoming years. Therefore, it is essential to further improve the aerodynamic and aeroacoustic capabilities of wind turbines.
Role of morphing technology

The Greek word “Morphos” can be defined as the process of transformation of structure or shape. The meaning of morphing in the aviation industry is the change of shape of the aircraft structure to maintain the stability of an aircraft or to attain the required lift or drag force and it has been around for over 100 years. One of the very first regularly constructed morphing prototypes was with swing-wing technology in the 1950s. Ever since then the morphing technologies has been exploited to its limits by the military aircrafts (see Fig. 1.1). Morphing technologies such as variable wing sweep has also been used in aircrafts such as F-111, US Air Force’s B-1 Lancer and US Navy’s F-14 Tomcat in the early 60s and 70s. When considering the wing on its own, for several years aircraft have employed the slats of the leading edge as well as flaps to alter the wing’s effective camber to increases the lift. Similarly, to increase the drag the retractable landing gear, as well as spoilers, are used. The advent of very light and strong composite materials over the past two decades have opened doors for redesigning the conventional wing configuration using morphing structures. Therefore, morphing continues to be a promising and empowering technology for upcoming decades, for the next-generation aerospace applications.

![Figure 1.1: Aircraft that successfully used span morphing wing over the years [3].](image-url)
1.2 Research objectives

The aim of this study is to improve our understanding of the aerodynamic and aeroacoustic performance of morphing structures for different types of airfoils. Firstly, a simple symmetric NACA 0012 airfoil with and without morphed trailing edges is thoroughly investigated. The improved understanding of the flow dynamics will aid us to better design morphing airfoils. Secondly, an MDA 30P30N high-lift airfoil is investigated with and without the application of slat cove filler to understand the slat noise generation mechanism and the most efficient way to attenuate them without compromising the aerodynamic performance. The primary objectives of this study can be categorized as follows,

[i] To understand the source of the improved aerodynamic performance for the morphed airfoil and its varying lift-to-drag performance over the wide range of angle of attacks. A deeper understanding of the surface pressure distribution and flow separation characteristics will be achieved with the aid of numerical simulations and experimental measurements.

[ii] To investigate the aeroacoustic performance of morphed airfoils and its noise generation mechanism. High fidelity numerical investigation will be performed to analyze the unsteady flow characteristics of the hinged and morphed airfoil.

[iii] To gain a deeper understanding of the slat noise sources and its flow mechanism using experimental techniques.

[iv] To understand the aerodynamic and aeroacoustic characteristics of the high-lift airfoil with and without the use of slat cove fillers.
1.3 Research outline

As part of this study, several experimental and numerical investigation were carried out. However, only a few of them are presented in this thesis. In this section, all the objectives that were met as part of this research will be detailed.

[i] The symmetric NACA 0012 airfoil with and without morphed flap trailing edge was investigated without a boundary layer trip using, lift and drag measurement, surface pressure measurements and wake measurements for five different trailing edge morphing camber profiles.

[ii] RANS simulation for both the hinged and morphed airfoil without trip was carried out for a wide range of angles of attack $\alpha = -5^\circ$ to $15^\circ$.

[iii] Detached Eddy Simulation for the hinged and morphed airfoil without trip was carried out at the angle of attack $\alpha = 0^\circ$ and $4^\circ$.

[iv] The NACA 0012 airfoil with boundary layer trip was thoroughly investigated for the morphed and hinged airfoil using lift and drag measurements, surface flow visualization and Particle Image Velocimetry measurements.

[v] Large Eddy Simulation for the NACA 0012 airfoil with the trip was performed for both the hinged and morphed airfoil at the angle of attack $\alpha = 0^\circ$ and $4^\circ$.

[vi] Experimental investigation of 30P30N airfoil including, lift and drag measurements, surface pressure measurements, unsteady surface pressure measurements, and Particle Image Velocimetry measurements were carried out for baseline, half-slat cove filler, slat cove filler and droop nose configurations at four different wind tunnels for a wide range of velocity and angles of attack.

[vii] RANS simulations for the 30P30N for the baseline, half-slat cove filler, and slat cove filler configurations were carried out for a wide range of angle of attack $\alpha = 0^\circ$ to $15^\circ$.

[viii] Large Eddy Simulation of the 30P30N airfoil with and without slat cove filler at high Reynolds number was successfully carried out at the angle of attack $\alpha = 5.5^\circ$. 


1.4 Thesis outline

Experimental and numerical investigation in order to understand the aerodynamic and aeroacoustic behavior of morphing structures were carried out. The experimental methods and the results are presented in this thesis in a systematic manner and the thesis is organized as follows. A literature review on the morphing structures focused on aerodynamic performances, high-lift airfoil basic aerodynamics and current understanding of the slat noise sources are detailed in Chapter 2. The wind tunnel setup, the model setup, and instrumentation of the NACA 0012 and 30P30N airfoil are detailed in Chapter 3. The results and discussions of the NACA 0012 airfoil with hinged and morphed flap including, the aerodynamic performance, the flow separation characteristics with respect to the angle of attack, the wake flow field with the mean velocities and the energy content are discussed in Chapter 4. This chapter also covers the unsteady flow characteristics and aeroacoustic performance of the hinged and morphed flap airfoil which includes, surface pressure fluctuations, wake velocity fluctuations and far-field noise calculations. In Chapter 5, the aerodynamic and aeroacoustic performance of 30P30N high-lift airfoils are discussed. The unsteady flow characteristics of the surface pressure fluctuation and far-field noise measurements are analyzed using higher spectral order methods. Finally, the summary of all the findings, the research contribution, and the possible future works are discussed in Chapter 6.
Aircraft noise has been a growing problem with an increasing number of airports and air travel becoming even more common. Passive noise reduction techniques currently being researched, such as serrated flap edges [6], side edge fences [7], porous flap edge [8] and edge brushes [9], have their limitations. Moreover, they may cause aerodynamic inefficiencies, such as lift reduction and increase in drag. Even promising methods, such as continuous modulus link flaps are limited as the lift change between the inner flap section and outer wing section results in the increase of induced drag of the wing resulting in low fuel efficiency [10].

These above reasons have directed the aerodynamic and aeroacoustic research toward the development of morphing wings and flaps. Morphing is the shape-varying technology which is used to attain the optimum performance in aircraft. When considering just the wing, it can be morphed in every segment since the changes in the wing area significantly influence the aerodynamic performance. Morphing wings are designed to alter its area, span, and shape of the airfoils, at times even twist and sweep. The alteration in the wing increases the flight operational conditions to fulfill various missions. For example, it can carry out loiter, cruise and perform high-speed maneuvers effectively without the use of seams or conventional control surfaces [4, 5]. In the forthcoming years, wing morphing is speculated to become as an essential change in the design of aircraft for the application of civil and military.
Morphing wing is a very attractive option as it can achieve a chordwise and spanwise differential camber variation with a single system and it also has the following benefits apart from the possible noise reduction capability [4, 11]:

[i] Increases flight operations to multiple conditions.

[ii] Differential camber variation in both chordwise and spanwise direction can be achieved with a single system.

[iii] No additional gaps for flaps provides smooth wing contour reducing drag and increasing the L/D ratio.

[iv] Higher aerodynamic efficiency resulting in higher fuel efficiency

[v] Weight loss due to fuel burn in long haul flights can be compensated using camber variation.

Active research into morphing wings began with the Mission Adaptive Wing (MAW) in 1979. It was conversely later discontinued due to its complexity and excess weight involved in this technique. Since then the materials and technology have developed significantly and mainly due to the advent of flexible smart materials the morphing wing technology once again has become an attractive option. The current approach mainly achieves this morphing in two ways, either by mechanical actuation or elastic deformation.

Developing an appropriate morphing concept, including structure design, material selection, and actuator choice covers relatively wide topic so this review here is merely a tip of the iceberg. For a desirable morphing structure, large deformation is necessary to achieve expected flow and noise control purposes. The high deformability can weaken the load-carrying ability of the structure, and the actuation requirements are then increased. Due to the conflicting nature of such requirements, anisotropic materials, such as composite materials and honeycomb cores, presenting high out-of-plane stiffness to in-plane-stiffness ratio, have attracted researchers’ attention in the recent years and are considered as strong candidates for morphing structure applications [11-29].
2.1 Morphing leading and trailing edge

Over the past decade, researchers have constantly persuaded the ability to achieve smooth continuous morphing in airfoil structures. Morphing structures are sought after as their seamless structures aid in the improved lift and reduced drag for wings with flaps. The main criteria for wing design are the ability to carry and transfer aerodynamic loads and to sustain structural integrity. In order to not compromise the strength of the wing, it is usually morphed only in sections i.e. at the leading edge or trailing edge of the airfoil. Several studies [11-29] have been carried out in order to achieve this smooth morphing structures with high strength.

![Image of Flight control surfaces on an Airbus A380](image)

**Figure 2.1: Flight control surfaces on an Airbus A380 [13].**

The basic control surfaces of a conventional aircraft is shown in Fig. 2.1. The high-lift devices in an aircraft such as the slats and flaps can be replaced with morphing structures to reduce weight and complexity, and also to improve the aerodynamic performance. When considering slats, flexible droop nose and slat cove fillers are considered as shown in Figs 2.2 and 2.3, respectively. When considering flaps, airfoil with morphing camber and morphing trailing edges are considered as shown in Fig. 2.8. Apart from the aircraft wing, the use of morphing trailing edge also extends to many other applications such as fixed wing, helicopter, tilt-rotor, wind turbine, and tidal stream turbine [30].
2.1.1 Morphing leading edge

The design for the morphing leading edge are based on the following criteria. The first is its ability to replace slat or a conventional droop nose successfully. In this scenario, the benefits come from the reduction of parasite drag and noise reduction from the smooth surface without gaps and steps. Other requirements for this scenario include the right curvature for improved aerodynamic performance and smooth leading edge contour during the cruise [16].

DLR has developed a morphing leading edge concept as part of the European FP7 SADE project [16]. They developed a kinematical mechanism for the drooping deformation of the wing leading edge. The morphing leading edge structure was evaluated for structural strength and shape accuracy of the clean and deformed shape. They also performed an optimization of the aerodynamic performance of the unmorphed and morphed leading edge configuration using RANS based simulations. It was shown that the higher aerodynamic performance required larger leading edge deflection. This high target deformation was not achievable due to the skin material strength and manufacturing constraints.

Numerical prediction of the mechanical stress on the droop nose leading edge concept was investigated by Monner et al. [17]. The numerical simulation included the skin, substructure, and kinematics of seamless gapless morphing leading edge. The material selection and the feasibility of the morphing leading edge concept were achieved from this preliminary study. Further experimental analysis of the smart morphing leading edge concept by DLR was carried out by Kintscher et al. [18]. The experiments involving stress and strain were carried out for a 2 m droop nose section. The results of the shape and strain of the leading edge in the experiments agreed well with the numerical simulations. The results showed maximum strain was concentrated at the droop nose leading edge tip location. The study also reported that the numerical predictions were overestimated due to the inconsistency of the skin thickness of the droop nose. These preliminary studies led to further implementation of the morphing leading edge on full scale models as shown in Fig. 2.2.

The leading edge slat generates a gap on the wing’s leading edge, allowing high-pressure air from the suction side to the pressure side of the wing, which adds momentum to the boundary layer, delaying and preventing separation and stall. Therefore, slat is a crucial component to
2.1. MORPHING LEADING AND TRAILING EDGE

Figure 2.2: SADE morphing leading edge with kinematic chain interior and elastic skin [19].

generate lift and delay stall at high angles of attack. This crucial slat component also generated high levels of tonal noise from the slat cavity. The noise generation mechanism and slat aerodynamics is further discussed in Sections 2.2.2 and 5.1. The noise generation mechanism in slat is dominated by the unsteady flow within the slat cavity region. Past successful methods in reducing slat noise involve the use of streamline structures to cover the slat cavity and this structure is often referred to as Slat Cove Filler (SCF) [133]. The SCF concept is based on morphing structures as it has to deform to retain the wing’s original shape in the retracted configuration as shown in Fig. 2.3.

Scholten et al. [20] carried out a design and optimization study on the use of shape memory alloy for the SCF. The thickness and flexures of the shape memory alloy were optimized to morph with minimum actuation force and moment. These preliminary tests showed the successful capability of the SCF for two main criteria, firstly to withstand aerodynamic loads at the deployed configuration and secondly, to retract-deploy quick and efficiently. Further experimental
and numerical studies aimed at reducing the actuation force for the movement of the SCF (deploy/retract) and to calculate the maximum stress on the Shape Memory Alloy (SMA) flexures at retracted position were carried out [21]. The tests were performed for the SCF profile that was optimized for maximum noise reduction. The study proved successful in reducing the actuation force to deploy and retract the SCF considerably compared to the previous study [20]. This was achieved through the optimization process which showed that a monolithic shape-memory alloy slat cove filler satisfied the design constraints. The aerodynamic and aeroacoustic characteristics of slat cove fillers available in the literature are discussed in Sections 2.2.2 and 5.1.

2.1.2 Morphing trailing edge

For conventional high-lift devices, there are many morphing ideas that are being developed. Most of them modify the entire wing structure parameters, such as the twist, wingspan, local or global camber. Apart from the research focused toward the global and local morphing capabilities on wings, airfoil trailing edge morphing has also been of high interest to researchers. Even though the entire wing morphing is a well established sophisticated morphing concept, just the trailing edge morphing has been proven to achieve enhanced aerodynamic performance.

The camber variation in airfoils are achieved by maximizing the effectiveness of the shape change at the trailing edge. The change is of pronounced structural movement and it results in distinct aerodynamic change [4, 14].

![Figure 2.4: Chain link trailing edge concept](image)

The active flexible rib [4], Rotating rib [14], Vertebrae like elements [15] are characterized through moving elements that have the ability to transmit the motion from one chain to another (see Fig. 2.4). These are some of the initial ideas aimed at smooth trailing edge deflection with the capability to be implemented on large wings. For aerospace applications, such structures
with an increased number of moving parts are considered less reliable. Moreover, these methods increase the weight of the entire structure and the skins needs added support to prevent itself from buckling under the external load actions.

**Figure 2.5:** The composite electrical trailing edge concept developed for Blended Wing Body aircraft design, where $M$ is the bending moment [22].

Recent trailing edge concepts incorporate electrically actuated composite structures, which reduces weight considerably and also increases reliability due to lesser moving parts. One such structure was proposed by Wildschek *et al.* [22] for control surfaces of Blended Wing Body aircraft. A hinged system, distributed chordwise, was linked to stringer's skin which was having the ability to refract the composite trailing edge skins and was powered through an electric actuator, as shown in Fig. 2.5. Most of the studies on morphing trailing edges are aimed at achieving a smooth curvature structurally. However, a very few number of studies were carried out to demonstrate the aerodynamic performance of the morphing trailing edges and the results from these studies are discussed below.

Barbarino *et al.* [23] proposed a new flap design for a flexible camber trailing edge. The structural performance was investigated using Finite Element Analysis on a full-scale wing. The tests were carried out on SMA to estimate the state of internal stress and temperature for a minimum activation. The SMA structure showed its suitability for a wide range of applications due to its huge structural integration and extraordinary capabilities for actuation. The morphed airfoil shapes were found to have the optimum aerodynamic load distribution for a high-lift with low weight and minimized mechanical complications. The aerodynamic performance estimated using two-dimensional vortex-lattice method showed improved lift coefficient for the morphed flap compared to the hinged flap. The lift coefficient results also showed increased difference between the two airfoils at higher angles of attack. The maximum difference in lift coefficient of
0.5 was found at angle of attack $\alpha = 24^\circ$ between the hinged flap and morphed flap airfoil.

Abdullah et al. [24] developed a deformable wing model by using Acrylonitrile butadiene styrene (ABS) plastic incorporated with SMA actuators. Aerodynamic tests were carried out at the RMIT's industrial low-speed wind tunnel to investigate the practicality of SMA actuators to change airfoil camber whilst maintaining aerodynamic performance. At the angle of attack of $\alpha = 0^\circ$, the maximum increase in the lift coefficient between before and after morphing is about 0.104. At the angle of attack $\alpha = 5^\circ$, the minimum increase in the lift coefficient was about 0.063. The results showed that at higher angles of attack the morphed airfoil has an increased lift-to-drag ratio of 0.32. These comparative results prove that the use of the SMA actuators along with the flexible skin essentially improved the aerodynamic performance of the airfoil.

Pecora et al. [25] illustrated an innovative morphing flap design on a CS-25 regional aircraft wing based on compliant rib mechanism. The aim of the project was to design a morphing rib concept and the actuator structure. They developed a fully functional flap design capable of morphing with and without acting aerodynamic loads using SMA actuators. They also verified the capability of morphing trailing edge to attain high-lift using a two-dimensional vortex-lattice method. The aerodynamic lift and drag results showed increased lift for morphed airfoils compared to hinged airfoils at increasing angles of attack. The results showed an increased lift coefficient of up to 16% at the angle of attack $\alpha = 14^\circ$, whereas at the angle of attack $\alpha = 2^\circ$ a lift increase of only 6% was observed.

Daynes and Weaver [26] used a NACA 63-418 airfoil with a chord of 1.3 m and a span of 1 m to design a full-size model of a morphing trailing edge. The flap was designed with a core of an aramid honeycomb structure along with a carbon fiber reinforced plastic (CFRP) skin on one side and silicone skin on the other side of the airfoil surface. With the CFRP rods, the flap has been incorporated. The aerodynamic investigation using XFOIL showed that the morphed trailing edge is capable of attaining the same lift coefficient as the hinged flap with 30% lesser flap deflection. Daynes and Weaver [27] conducted further testing of the wing in the wind tunnel to show the feasibility and load bearing capability of the morphing trailing edge design.
2.1. MORPHING LEADING AND TRAILING EDGE

Yokozeki et al. [28] conducted an experimental study using a Wortmann FX63-137 airfoil with a corrugated flap structure actuated by a wire (see Fig. 2.6a). The aerodynamic tests were carried out for a chord-based Reynolds number of \( Re_c = 5 \times 10^5 - 1.5 \times 10^6 \) at the low-speed wind tunnel facility at the Japan Aerospace Exploration Agency. The airfoil had a chord length of 80 cm with a flap length of 31% chord length. The results showed superior performance for the morphing flap at low angles of attack compared to the hinged flap with an increase of lift coefficient of up to 0.25 at the angle of attack \( \alpha = 10^\circ \). However, as the angle of attack is increased, the difference between the hinged and morphed flap diminishes, as shown in Fig. 2.6b. The lift-to-drag also showed superior performance for the morphed airfoil with a flap deflection angle of 20°.

Wolff et al. [29] presented a comprehensive study using two-dimensional RANS simulations on a modern turbine airfoil (DU08-W-180-6.5). The tests were carried out for various flap lengths and various flap deflection angle. The results showed that increased flap length and deflection angle resulted in increased lift. The study showed that the optimum flap length for improved aerodynamic performance is a flap length of 15% – 20% of the chord resulting in increased lift-to-drag ratio as shown in Fig. 2.7. The results here yet again confirmed that the use of a morphing trailing edge has a significant effect on lift and drag characteristics and the stall behavior of the airfoil, as seen in Fig. 2.7. The results also showed that the same lift-to-drag ratio can be
CHAPTER 2. LITERATURE REVIEW

achieved for the morphing airfoil at an angle of attack of 3° lower than the reference airfoil.

![Figure 2.7: The composite electrical trailing edge concept developed for Blended Wing Body aircraft design, where $M$ is the bending moment [29].](image)

Woods et al. [30] developed a bio-inspired Fish Bone Active Camber (FishBAC) concept for airfoil morphing. This concept has the capability to generate large bidirectional changes in airfoil camber (see Fig. 2.8a and b). The NACA 0012 airfoil with FishBAC trailing edge was tested in the wind tunnel at various angles of attack with various trailing edge deflection angles. The tests were carried out for a chord-based Reynolds number of $Re_c = 1.1 \times 10^6$. The results for the FishBAC airfoil relative to the hinged flap airfoil showed an increase of 0.72 in the lift coefficient at the angle of attack $\alpha = 0°$. The morphed FishBAC airfoil showed an increase of 1.07 at the maximum lift coefficient relative to the hinged airfoil. The morphed airfoil also showed a reduction in drag without compromising the lift. The results also showed a notable improvement in the lift-to-drag ratio on the order of 20%–25% for the morphed airfoil. The morphed airfoil also sustained the maximum lift-to-drag ratio over a wide range of angles of attack (9°) compared to the hinged airfoil (3.6°).

Wu et al. [33] performed an aerodynamic study on morphing carbon fiber composite airfoil concept with an active trailing edge. Linear ultrasonic motors were used for actuation with compliant runners enabling full freedom of movement of the trailing edge. The wind tunnel tests for the NACA 4418 airfoil with a chord length of 16 cm with active trailing edge at a chord-based $Re_c = 5 \times 10^5$ were carried out. The results were validated with existing experimental
and computational data sets. The results showed that the airfoil moment and pitch can be independently controlled by the use of morphing flap. The results showed increased camber resulted in increased lift-to-drag ratio compared to the hinged airfoil, as expected. The lift-to-drag results showed that the morphing airfoil outperformed the hinged airfoil at low angles of attack. It was suggested that for better aerodynamic performance at higher angles of attack, the design point optimization for the morphing camber was required.

Other works on morphing trailing edges that include basic aerodynamic analysis are as follows, Previtali et al. [34] showed the possibility of successfully using compliant rib mechanism as a replacement to control surfaces. The lift coefficient results calculated from Xfoil for the morphing case showed superior lift performance relative to the hinged airfoil. Raither et al. [35] used adaptive stiffness in order to morph the trailing edge and performed inviscid calculations to estimate the aerodynamic performance. The results showed a lift coefficient increase of 0.4 for the morphed trailing edge compared to the hinged case.

Tani et al. [10] carried out an experimental study at Kyushu University on a NACA23012 wing model along with an exchangeable slotted half span flap. The main objective of this experiment was to examine the noise reduction capabilities of the spanwise morphing. For the half span wing model, the beamforming noise measurement results showed a reduction in the flap side-edge noise for the morphing flap configuration. The summary of the above literature and other key papers
regarding morphing trailing edge leading to the current study will be provided in Section 4.1.

The available literature on the morphing airfoils presented above clearly shows the scarcity of the available studies on the aerodynamic and aeroacoustic performance of morphing airfoils. The prior studies were much more oriented toward the structural aspect of morphing airfoil rather than aerodynamic and aeroacoustic performance. However, it is very essential to further understand the flow field such as the surface pressure distribution, flow separation, boundary layer profiles, wake velocity profiles, wake turbulence and the unsteady characteristics of the morphing airfoils. Moreover, the morphing airfoil is thought to reduce airfoil noise but there is no strong evidence to validate the idea. Therefore, the present study will solely focus on the aerodynamic and aeroacoustic performance of hinged and morphed airfoils.
2.2 High-lift airfoil

High-lift devices play a crucial role in the aircraft performance as they increase the maximum lift coefficient during takeoff and landing. Even though the slats on high-lift devices have been used since the early 1900s, its flow physics was explained much later in the landmark work by Smith [37]. The high-lift airfoil generates extra lift by critically making use of basic flow physics. The flap generates lift by increasing the effective camber of the airfoil. The extra lift is also created by increasing the effective chord of the airfoil with the extension of the flap, as shown in Fig. 2.9. The use of slat increases the stall angle but does not influence the lift coefficient as shown in Fig. 2.9. It is well known that the airfoil stalls due to flow separation on the suction surface, this can be avoided by having thinner boundary layers and smaller pressure gradients. The use of slat makes this possible by reducing the flow velocity over the leading edge of the main-element, thus reducing the adverse pressure gradient. Even though the basics of high-lift airfoil appears straightforward, its has a very complex flow physics with large low-speed regions, strong pressure gradients, and confluent boundary layers [37]. Moreover, transonic flow regions have been identified over the slat at high angles of attack [38]. The understanding of the flow physics of the high-lift airfoils by Smith are summarized as,

[i] *The slat effect* works by reducing the pressure on the leading edge of the consecutive downstream element by reducing the velocity, thus improving just the $C_{L,max}$ by evading separation. This is due to the upstream element acting as a point vortex.

[ii] *The circulation effect* is the increased recirculation at the upstream element trailing edge to reach Kutta condition consequently increasing local velocity due to the presence of the downstream element.

[iii] *The dumping effect* is the high-velocity wake of the upstream element insulating the boundary layer of the downstream element with the free-stream, thus reducing the pressure gradient and avoiding separation over the downstream element.

[iv] *The off-the-surface pressure recovery* is the efficient mixing of the isolated high-velocity upstream element wake without interacting with the wall of the downstream element.
[v] The fresh boundary-layer effect is the ability to sustain stronger adverse pressure gradients with the thin boundary that are generated at the leading of each individual element. These are the basic key points of the high element flow field with the deployed slat as shown in Fig. 2.9. As part of the current study, MDA 30P30N airfoil is used for further understanding the aerodynamics and aeroacoustics of the high-lift airfoil (see Section 3.1.3). The following section discusses the available literature on the experimental campaigns on MDA 30P30N airfoil.

Figure 2.9: Effects of slat and flap on an airfoil [36].

Figure 2.10: Typical flow around a high-lift airfoil [39].
2.2. HIGH-LIFT AIRFOIL

2.2.1 High-lift airfoil aerodynamics

NASA and MDA in a collaborative study carried out an extensive experimental program [40] in order to establish a detailed experimental database for mean flow and turbulence characteristics of shear flow about realistic three-element high-lift airfoil under realistic flight conditions. The mean velocity profiles results showed in detail the features of merging shear layers, flow curvatures in the near wake and flow properties in the flap-well area. The boundary layer showed no separation as the friction coefficient was never smaller than 0.0015. They observed considerable variation in static pressure over various part of the shear layers due to confluent wakes of the consequent elements thickening the boundary layer. The slat wake develops within the large pressure gradient on the leading edge of the main-element. The surface pressure results showed a pressure gradient spike just downstream of the main-element and flap due to the strong acceleration of flow on the lower surface. The magnitude of the turbulent stresses was observed to be very high in the near wake region for both the configurations. The observed streamwise fluctuations were large at all station but the spanwise fluctuations were larger for some stations. In general, the overall results showed that the main-element had the strongest wake and other two wakes of the slat and flap were absorbed into it, widening the overall wake width. They also expected flow reversal on the flap wake at the trailing edge vicinity rather than on the boundary layer on the flap (see Fig. 2.10). Shear flow observed in high-lift airfoils was shown to be much different from classical shear layers in airfoils.

As a part of the above mentioned NASA-MDA collaborative study a detailed investigation was carried out to assess the maximum lift capability and effects of high Reynolds numbers on an MDA high-lift airfoil by Valarezo et al. [41]. An optimization study was also carried out on deflection angles, gap and overhang positions for the slat and flap of the MDA high-lift airfoil. The results showed that the slat positions were sensitive to the Reynolds number and the optimized slat gap position resulted in a higher maximum lift at higher Reynolds numbers. The results of the optimized flap positon delivered a maximum lift coefficient in excess of 4.5. Even though higher flap deflection angles produced higher lift coefficient, it resulted in the large scale flow separation over the flap [41]. Also, the maximum obtainable lift showed a significant dependence on Mach number at a given Reynolds number. The results from the study showed that to produce high-lift
coefficient the optimized MDA three-element airfoil should have a slat deflection, $\delta_s = 30^\circ$ and flap deflection, $\delta_f = 30^\circ$ and it is often referred to as "30P30N airfoil".

A continuation of the above study detailing on the issues involved in very high Reynolds number experimentation along with airfoil model design and hardware requirements for the 30P30N high-lift airfoil was published by Valarezo [42, 43]. This study also stressed the importance of material selection of the high-lift airfoil as the expected dynamic load on the individual elements were high. The main-element was manufactured using aluminum whereas the smaller slat and flap elements were made from steel in order to minimize deflection at high aerodynamic loads. The results showed that for the landing configuration with slat deflection angle of $\delta_s = 30^\circ$, and flap deflection angle of $\delta_f = 30^\circ$, the maximum lift was found at $Re_c = 9 \times 10^6$ and the lift dropped at higher $Re_c = 16 \times 10^6$. But, for the take-off configuration with slat deflection angle of $\delta_s = 20^\circ$, and flap deflection angle of $\delta_f = 10^\circ$, the maximum lift was not affected for Reynolds number ranging between $Re_c = 5 \times 10^6$ and $20 \times 10^6$. However, the drag was affected for the take-off configuration with differences of up to 10% for the same range of Reynolds numbers. The flow field results indicated a total merging of the slat and main-element wake, implying a very slow-moving air over the flap. The observed flow also indicated flow reversal close to the flap trailing edge even though it was absent on the flap surface pressure results.

Further flow field measurements around 30P30N and 30P30AD high-lift airfoils configurations optimized by Valarezo et al. [41, 42] were reported by Chin et al. [44], in order to improve the understanding of the factors influencing the optimization of the three-element high-lift airfoils. The only difference between the 30P30N and 30P30AD airfoil is that the latter has an increased flap gap distance $g_f$ of 0.23%. The results showed that the flap was highly loaded at the low angles of attack but as the angle of attack was increased the loading on the slat and main-element increased. The trends associated with the presence of adverse pressure gradient over the flap were observed here as observed in earlier experiments [40–42]. The slat wake was found to be more prominent at high angles of attack, which is consistent with higher slat loading. At higher angles of attack, the main-element wake diverges notably and the entire wake structure grows significantly large. However, the lift is seen to increase as the Reynolds number was increased but the stall angle was not affected. For lower angles of attack, the boundary
2.2. HIGH-LIFT AIRFOIL

Layer results showed a tendency of the flow to separate over the flap at lower Reynolds number with a larger wake and decreased merging between the slat and main-element. The results of the total pressure coefficient revealed the differences in the wake between the tree-elements and wake merging regions in detail. Even the small difference in flap gap of 0.23% between the 30P30N and 30P30AD configurations showed prominent differences in the results. Between the angles of attack, \( \alpha = 12^\circ \) to \( 16^\circ \), a reduction in the lift and an increase in the drag was observed, however, these results were reversed at higher angles of attack. The 30P30AD configuration was observed to have an increase in the \( C_{L,max} \) of approximately 0.05 and it increased the stall angle by one degree compared to the 30P30N configuration. Moreover, the 30P30AD configuration also displayed lesser wake merging between the different multi-element components along with lesser off-body flow reversal at higher \( \alpha \) resulting in slightly higher \( C_{L,max} \).

The understanding of the flow physics of the high-lift airfoils were further improved by skin friction (surface shear stress) measurements made at high Reynolds number by Klausmeyer and Lin [47]. The skin friction coefficient \( (C_f) \) results for the 30P30N airfoil showed large values at the leading edge of the main-element due to high-velocity flow with thin boundary layers. Moreover, the \( C_f \) values steadily increased as the angle of attack \( (\alpha) \) was increased and the \( C_f \) decreased at downstream locations over the main-element with steady decrease with \( \alpha \). For \( Re_c = 5 \times 10^6 \), at the main-element downstream location \( x/c = 0.825 \), \( C_f \) rapidly dropped after \( \alpha = 21^\circ \) \( (C_{L,max}) \) with flow separation, this was shown much more clearly at \( \alpha = 24^\circ \) with negative \( C_f \). The flap showed high \( C_f \) with peak values at locations near the flap leading edge but the \( C_f \) values rapidly dropped at further downstream locations. For \( Re_c = 5 \times 10^6 \), at \( \alpha = 8^\circ \) regions of negative \( C_f \) were observed, which characterised flow separation at the flap trailing edge location \( x/c = 1.094 \), but as the Reynolds number increased the separation disappeared. The \( C_f \) was seen to increase at the flap trailing edge after \( \alpha = 16^\circ \) for all the tested Reynolds numbers. Since the results showed flow attachment over the flap, even at high angles of attack past the \( C_{L,max} \), it was suggested that the stall might be characterized by the flow reversal in the main-element wake, which was previously confirmed by the experimental velocity measurement results [44]. The skin friction results presented by Klausmeyer and Lin [47] lead to better insight into the complex flow behavior over the three-element high-lift airfoils.
CHAPTER 2. LITERATURE REVIEW

A much more crucial flow physics at high Reynolds number was reported by Spaid et al. [45, 46] pertaining to the aerodynamic performance characteristics of the 30P30N airfoil with different slat gap, flap gap, and flap deflection angle. When the slat gap was reduced from $g_s = 2.95\%$ to $2.48\%$, the boundary layer profiles showed negligible changes with very small differences in the slat gap. However, a notable loss in the lift beyond stall and a slight increase in the suction peak on the main-element was observed. The change in flap gap from $g_f = 1.27\%$ to $1.50\%$ resulted in lift benefits beyond $\alpha = 16^\circ$, with a marginally higher $C_{L,max} = 0.05$ and it delayed the stall angle by one degree, as previously reported by Chin et al. [44]. This shows the crucial role of the spreading/merging of the multi-element wakes, the unloading of the flap due to spreading of wake above the flap results in the unloading of the aft portion of the main-element resulting in slightly higher $C_{L,max}$. The results for larger flap deflection ($\delta_f = 35^\circ$) did not show any increase in lift for the intermediate angles of attack. The results for the larger flap deflection also showed a slight increase in the drag coefficient $C_D$ and no substantial increase in $C_{L,max}$. The velocity profiles for larger flap deflection clearly showed evidence of separation at low to moderate angles of attack but at higher angles of attack, the flow remains attached. The results for cases with larger flap deflection angles did not show any increase in the flow turning angle, however, an increase in the boundary displacement thickness about the flap was observed. The results showed that the increased flap wake spreading with larger flap deflection angles limited the capability to attain higher $C_{L,max}$.

Paschal et al. [48] investigated the 30P30N flow field using Particle Image Velocimetry (PIV) measurements. Several measures were taken in order to maintain flow two-dimensionality such as, tangential blowing, smoke flow, and oil flow visualization to check surface streamlines. The PIV result showed spanwise vortex structures on the spatial scale of approximately $5mm$ present in the slat wake due to the slat cove flow unsteadiness at $\alpha = 4^\circ$. However, this unsteadiness in the slat wake was not as frequent at higher angles of attack. High levels of normalized wall normal stress were observed at lower angles of attack of $\alpha = 4^\circ$, which correlated with the mean velocity profiles in the slat wake. Overall, the results showed that the unsteady components of the local flow field in the slat cove play an important role on the aerodynamic performance of the high-lift airfoil especially from $\alpha = 4^\circ$ to $10^\circ$. 

24
In order to further improve the understanding of the flow field at the slat cove, slat cusp and slat wake regions, PIV measurements were carried out by Jenkins et al. [49]. At the slat cove region, results for mean flow field streamwise velocity, crosswise velocity, and turbulent kinetic energy showed well-defined shear layers with equal levels of vorticity inside the slat cove for all the tested angles of attack. These results also correlated with the negative streamwise velocity observed on the slat lower surface for all the angles of attack. The shear layer trajectory leaving the slat cusp was seen to be affected by the negative velocity, which also changed the reattachment point of the shear layer at all the angles of attack. The instantaneous flow fields showed spanwise vortices leaving the slat cusp with the shear layer trajectory and impinging on the slat lower surface, where some vortices are entrained into the recirculation region while others left through the slat gap. It was also suggested that this type of flow behavior might contribute to higher sound pressure levels. At the slat trailing edge region different flow behavior were observed, including, vortices from slat trailing edge without slat cove vortices passing through slat gap, ejection of the positive vortices from the slat cove along with trailing edge vortex shedding and finally negative sign vortices from slat trailing edge along with positive sign vortices from slat cove passing through slat gap.

2.2.2 High-lift airfoil noise sources

The importance and the flow physics of leading edge slat were discussed in Section 2.2. Even though the slat plays a crucial part in increasing the stall angle of the aircraft, it is identified as a prominent airframe noise source during approach and landing [50]. Slat noise is a complex problem as it generates both broadband and tonal noise. Cavity resonance is believed to play a major role in the generation of the tonal narrow band peaks. The noise sources pertaining to the slat and the flow field associated with it were outlined by Choudhari and Khorrami [51], see Fig. 2.11. As shown in Fig. 2.11, the accelerated flow coming from the slat leading edge passes the slat cusp with the same high energy. This flow impinges on the slat lower side, resulting in a recirculation region with the impingement position moving to upstream locations with increasing angle of attack. A shear layer develops between the regions of varying velocities, which gives rise to the low-speed cove recirculation and high-speed slat gap flow. As a result of the velocity
CHAPTER 2. LITERATURE REVIEW

Table 2.1: Typical noise source range presented in terms of slat based Strouhal number.

<table>
<thead>
<tr>
<th>Source</th>
<th>Slat Strouhal number</th>
</tr>
</thead>
<tbody>
<tr>
<td>Slat cove oscillation</td>
<td>$St_s = 0.5$</td>
</tr>
<tr>
<td>Shear layer instabilities and slat cusp vortex shedding</td>
<td>$1 \leq St_s \leq 5$</td>
</tr>
<tr>
<td>Slat trailing edge vortex shedding</td>
<td>$St_s \approx 30$</td>
</tr>
</tbody>
</table>

shear between the two varying velocities between the fluid, two-dimensional Kelvin-Helmholtz instabilities form in the shear layer, resulting in coherent structures that impinge on the slat lower surface.

Terracol et al. [52] performed a series of computational studies on a LEISA F-16 high-lift airfoil. The results showed high levels of turbulence within the slat cove with large streamwise oriented vortices leaving the slat cusp. These vortices along with hairpin vorticial structures were seen to impinge on the slat lower surface, before mixing into the slat cove vortex or leaving through the slat gap. These expanding vortices are likely to be contributing to the broadband noise component. The slat cove flow phenomenon was also observed in previous PIV studies by Paschal et al. [48]. Apart from the low-frequency noise generation from the shear layer impingement on the slat lower surface, there is evidence of low-frequency oscillation of the slat recirculation vortex that is linked to shear layer flapping corresponding to $St_s = 0.15$ [54].

Khorrami et al. [53] identified the sources of the high-frequency slat noise with the aid of URANS simulations. This study confirmed the presumption that high-frequency noise was related to vortex shedding from the slat trailing edge. The observed high-frequency tonal peak was related to the flow velocity and the trailing edge thickness. The changing impingement location on the slat lower side at the vicinity of the slat trailing edge also plays a role in the intensity of the vortex shedding. Further experiments by Olson et al. [55] confirmed the source of the high-frequency noise as the slat trailing edge vortex shedding. There is also presumption that the proximity of the main-element to the slat cove and slat trailing edge would result in additional interference effects [9]. The frequency related with the slat noise mechanisms are summarised in Table 2.1.
2.2. HIGH-LIFT AIRFOIL

Figure 2.11: Potential slat noise sources and flow field adopted from Choudhari and Khorrami [51].

The above mentioned slat noise sources and physical mechanisms were taken into account and several slat noise reduction methods were suggested over the past decade. Dobrzynski et al. [50] performed far-field measurements for a 1/7.5 scaled A320 model and showed high levels of slat noise radiated predominantly on the rear arc at low frequencies. Their approach attenuated the slat noise by separating the cove vortex and the shear layer leaving the slat cusp by using a slat cove cover (see Fig. 2.12a) to reduce the turbulence merging between them. The results (see Fig. 2.12b) showed a noise reduction up to 2 dB can be achieved with the application of the slat cove cover.

Khorrami and Lockard [56] investigated the possibility of slat noise reduction using blade seal extensions with the aid of URANS simulations on a Boeing 777 high-lift system (see Fig. 2.13a). The blade seal extensions significantly reduced the strength of the shear layer leaving the slat cusp region and also by moving the shear layer impingement closer to the slat trailing edge. The results also showed the weakening of the shear layer vortices. The slat trailing edge vortex shedding also showed reduced energy due to the presence of the reattachment point closer to the slat trailing edge. The blade seal extension showed significant levels of noise reduction at low and high frequency, as shown in Fig. 2.13b.
Horne et al. [133] approached the slat noise problem by covering the slat cove cavity with a streamlined shaped solid slat cove filler (see Fig. 2.14a). The slat cove filler covers the cavity region thus completely eliminating the flow unsteadiness within the slat cavity. The far-field measurements using microphone array showed the capability of slat cove filler in reducing strong slat noise sources. The results showed noise reduction except at a mid-range frequency of 1200 Hz which is suggested to be due to the lift discontinuity caused by vortical structures. The far-field measurements revealed significant noise reduction up to 4-5 dB for the slat cove filler compared to the baseline slat as shown in Fig. 2.14b. Due to its high noise reduction capabilities the
2.3 Summary

The literature review presents an overview of the need for morphing in the aviation industry and also the importance of high-lift device. The review also presents the progress made in the morphing structures and their aerodynamic performance. With respect to leading edge morphing, the available research on droop nose and slat cove fillers were summarised. The review showed the high-lift and high lift-to-drag characteristics of morphing trailing edges compared to the hinged configuration. The study involving spanwise morphing flap showed the possibility of noise reduction using morphing trailing edges. This morphing literature review presented here clearly shows the research gap in the aerodynamic and aeroacoustic performance of morphing trailing edge, which this current study aims to fulfill using experimental and computational methods.

The importance of high-lift device for an aircraft was explained in detail. The aerodynamic performance of the slat on a high-lift airfoil was reviewed. The review clearly points out the importance of having the confluent boundary layer over the main-element and the flap isolated to avoid early separation. The review of the 30P30N experimental study showed the high influence of the slat gap and flap gap on the high-lift aerodynamic performance. The slat was shown to be a prominent noise contributing component with regards to airframe noise making it significantly

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Figure 2.14: Slat cove filler (left) and measured far-field noise reduction on full scale wing (right) [133].

Acoustic characteristics of the slat cove filler were further studied by other researchers which is summarized in Section 5.1.
important to further understand the physical mechanism behind it. The current understanding of the noise generation and physical mechanism behind slat noise was discussed in detail. The slat noise reduction mechanism developed over the past decade showed that high acoustic performance can be achieved with the use of slat cove filler. Although there are a few studies on the aeroacoustic performance of the slat cove filler, there is no aerodynamic measurements, near-field unsteady surface pressure measurements and near to far-field coherence studies available for slat cove fillers. The current study aims to provide that and also further improve the understanding of slat noise and noise reduction mechanisms with and without the use of slat cove filler using high fidelity experimental measurements.
A series of experimental studies were performed to evaluate the aerodynamic and aeroacoustic capabilities of morphed structures on airfoils. Composite structures capable of morphing for aerospace application are highly sought after over the past decade. However, the aerodynamic and aeroacoustic behavior of such morphing structures have not been thoroughly investigated. Therefore, in order to investigate these two different type of morphing structure applications were selected. A simple morphed trailing edge configuration that is used for aileron and flap of a wing and a slat cove filler that used for slat noise reduction on the high-lift airfoil. The study is mainly focused on the aerodynamic and aeroacoustic behavior of morphed structures on wings.

This chapter provides a detailed outline of the experimental facilities, model instrumentation and measurement techniques employed in the current study. The configuration and capabilities of the wind tunnel facilities used for the experiments are described in Section 3.1.1. A symmetric NACA 0012 airfoil model and instrumentation used for the simple morphed trailing edge study are discussed in Section 3.1.2. The detailed manufacturing method and instrumentation of the 30P30N high-lift airfoil and the slat cove filler used for slat noise study is described in Section 3.1.3. The aerodynamic characteristics are studied using lift and drag measurements and pressure distribution measurements which are discussed in Section 3.2.1 and 3.2.2, respectively. The flow structure in the wake and the surface flow visualization are measured using Particle
Image Velocimetry and oil flow visualization, as detailed in Section 3.2.3 and 3.2.4, respectively. The pressure transducers used for unsteady surface pressure measurement are discussed in Section 3.2.5. The far-field noise measurement setup is described in detail in Section 3.2.7. The method used for calibrating the near-field transducers and far-field microphones are described in Section 3.2.6 and 3.2.8, respectively.

3.1 Model configurations and instrumentations

To study the aerodynamic and aeroacoustic performance of morphing technologies in various aircraft applications two different experimental rigs were designed and manufactured, namely NACA 0012 airfoil and 30P30N high-lift airfoil. The NACA 0012 airfoil is used for studying the aerodynamic and aeroacoustic effects of surface camber close to the wing’s trailing edge. The high-lift airfoil 30P30N rig is used for studying noise generated from the high-lift device and ways to suppress them while maintaining the aerodynamic performance of the high-lift device.

3.1.1 Wind tunnel facilities

The aerodynamic and aeroacoustic measurements were carried out in different wind tunnel facilities at the University of Bristol. The wind tunnels used in the current study are the open jet wind tunnel, a large wind tunnel, low turbulence wind tunnel, and the aeroacoustic wind tunnel. A description of the wind tunnel used in the current study is provided below.

[i] Open jet wind tunnel: The open jet wind tunnel is a closed-loop wind tunnel with an open test section of length 2 m. The nozzle has a diameter of 1.1 m. The wind tunnel is capable of reaching a maximum reliable flow speed of 30 m/s and with a turbulence intensity of 0.5%. This wind tunnel was used for oil flow visualization, as well as preliminary set-up preparation and aerodynamic force measurements.

[ii] Large wind tunnel: The large wind tunnel is a low-speed closed-circuit wind tunnel with a contraction ratio of 3:1. It has an octagonal working section of 2.1 m×1.5 m×2.7 m and can deliver a stable flow velocity range of 10 m/s to 60 m/s. This wind tunnel is used for lift and drag, and surface pressure distribution measurements.
3.1. MODEL CONFIGURATIONS AND INSTRUMENTATIONS

[iii] **Low turbulence wind tunnel**: The low turbulence closed-circuit wind tunnel has an octagonal working section of 0.8 m × 0.6 m × 1 m. The contraction nozzle has a ratio of 12:1. The wind tunnel's maximum velocity is 100 m/s with a minimum turbulence level of 0.05% [57]. This wind tunnel was used for Particle Image Velocimetry measurements and surface pressure measurements.

(iv) **Aeroacoustic wind tunnel**: The aeroacoustic wind tunnel is a closed circuit open-jet wind tunnel in an anechoic chamber with dimensions of 4.6 m × 5.0 m × 7.9 m (Height × Width × Length). The chamber is fully anechoic down to 160 Hz, achieved by internally padding the chamber with wedges having a base dimension of 0.3 m × 0.3 m and a length of 0.34 m. The contraction nozzle has an exit dimension of 0.5 m × 0.775 m with a contraction ratio of 8.4:1. The wind tunnel is capable of flow velocities up to 45 m/s with turbulence levels as low as 0.25% [58]. This wind tunnel was used for aeroacoustic measurements.
3.1.2 NACA 0012 airfoil experimental setup

The NACA 0012 airfoil has been chosen to study the aerodynamic and aeroacoustic capabilities on morphing trailing edges as an alternate to standard hinged trailing edge flaps. A NACA 0012 airfoil model with a chord length of $c = 0.2$ m and a span length of $l = 0.45$ m was manufactured using RAKU-TOOL® WB-1222 polyurethane tool board. The tool board was machined to the NACA 0012 profile using a 3-axis CNC machine. The airfoil was designed with an interchangeable trailing-edge section (flap) with a chord-wise length of $b = 0.06$ m ($0.3c$), see Fig. 3.1. The flap deflection angle ($\beta$) is defined as the ratio of the morphing flap length, $b$ and tip deflection length while maintaining the same flap surface area and the flap had a deflection angle of $\beta = 10^\circ$, as shown in Fig. 3.3. The boundary layer was tripped at location $x/c = 0.1$ (see Fig. 3.2) on both sides of the airfoil using a serrated turbulator tape with a height of 0.5 mm and serration angle of 60° [59, 60]. The three-dimensionality effects of the flow over the airfoil were reduced by the use of circular side-plates with a radius of 0.17 m, see Fig. 3.4. Ai et al. [61, 62] designed various flaps with increasing surface curvature (see Fig. 3.3) and used Xfoil-BPM model to assess the aerodynamic and aeroacoustic performance of a NACA 0012 airfoil with novel morphed flaps. The results from these studies were then used for the design of the morphed flap camber profiles for further experimental and computational studies [63–67]. In the current study, the NACA 0012 airfoil with morphed flap profiles (see Fig. 3.3) with the lowest $C_L$ (Hinged Flap, HF) and highest $C_L$ (Morphed Flap, MF) for the same flap deflection angle ($\beta = 10^\circ$) from the previous studies [63–67] were thoroughly investigated to improve the understanding of the aerodynamic, aeroacoustic and wake turbulence characteristics of such airfoil configurations.

The NACA 0012 airfoil was equipped with 38 pressure taps along the chord with 19 on the suction and pressure side (15 on the leading edge and 4 on the interchangeable morphing trailing edges). The pressure taps made out of brass tubes with an inner and outer diameter of 1.6 mm and 0.6 mm, respectively. The pressure taps are adhered to the airfoil into well distributed 2 mm×2 mm grooves that run along the span of the airfoil. To study the flow field around the airfoil, Particle Image Velocimetry (PIV) technique and oil flow visualization were carried out, the details of which are presented in Sections 3.2.3 and 3.2.4, respectively. The airfoil was covered in a self-adhesive black vinyl sheet with a matt finish (see Fig. 3.2) to reduce the surface reflection.
for the PIV measurements and to capture the flow patterns in oil flow visualizations.

Figure 3.1: NACA 0012 airfoil model with the interchangeable trailing edge.

Figure 3.2: NACA 0012 airfoil setup in the low-turbulence wind tunnel.

Figure 3.3: Geometric details of the NACA 0012 airfoil with a flap deflection angle of $\beta = 10^\circ$ named Hinged Flap and Morphed Flap airfoils.
CHAPTER 3. EXPERIMENTAL AND COMPUTATIONAL SETUP

Figure 3.4: NACA 0012 airfoil with side-plates setup in the large low-speed closed-circuit wind tunnel.

Figure 3.5: The camera window locations used for the PIV measurements of the NACA 0012 airfoil.
3.1.3 High-lift airfoil experimental setup

An overview of the existing experimental study on MDA 30P30N high-lift airfoil was discussed in Section 2.2.1. Prior experimental studies had pointed out that the material consideration is of high importance due to the possible high aerodynamic loads on the tested high-lift airfoil. The multi-element airfoil was manufactured using aluminium to avoid any slat or flap deflection during testing at high flow speeds, which might affect the slat or flap gap \( o_s \) and \( o_f \). It was also pointed out in the previous studies \([41–44]\) that the two-dimensionality over the multi-element model is important as to avert non-essential vortices and noise, hence, no clamps were used in the spanwise direction for the slat and flap to avoid three-dimensionality in the flow in the design of the MDA 30P30N model.

![Exploded and assembled view of the manufacture MDA 30P30N airfoil model.](image)

The MDA 30P30N three-element airfoil model with a span of \( l = 0.53 \) m and a retracted chord length of \( c = 0.35 \) m was made from aluminium alloy 6082 which has the highest strength of the 6000 series alloys and with an excellent corrosion resistance. The aluminium plates were
machined to the required aerodynamic profiles using a 4-axis CNC machine. The main-element (ME) of the MDA 30P30N airfoil is made of two separate pieces (see 3.6), the upper and the lower with a hollow centre and skin thickness of ≈ 4 mm to reduce weight and have space to incorporate microphones within the airfoil. The slat was manufactured with an interchangeable slat cusp, which was achieved by a spanwise slot with a depth of 15 mm.

![Figure 3.7: 30P30N three-element airfoil geometric parameters.](image)

Table 3.1: Geometrical parameters in percentage of stowed airfoil chord, $c = 0.35$ m.

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Symbol</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Slat chord</td>
<td>$c_s$</td>
<td>0.15c</td>
</tr>
<tr>
<td>Main-element chord</td>
<td>$c_{me}$</td>
<td>0.83c</td>
</tr>
<tr>
<td>Flap chord</td>
<td>$c_f$</td>
<td>0.3c</td>
</tr>
<tr>
<td>Slat deflection angle</td>
<td>$\delta_s$</td>
<td>30°</td>
</tr>
<tr>
<td>Flap deflection angle</td>
<td>$\delta_f$</td>
<td>30°</td>
</tr>
<tr>
<td>Slat gap</td>
<td>$g_s$</td>
<td>2.95%</td>
</tr>
<tr>
<td>Flap gap</td>
<td>$g_f$</td>
<td>1.27%</td>
</tr>
<tr>
<td>Slat overhang</td>
<td>$o_s$</td>
<td>-2.5%</td>
</tr>
<tr>
<td>Flap overhang</td>
<td>$o_f$</td>
<td>0.25%</td>
</tr>
</tbody>
</table>
### 3.1. MODEL CONFIGURATIONS AND INSTRUMENTATIONS

**Slat and Flap Configuration:** Early studies by NASA [41–44] have shown variation in the overall aerodynamic performance of 30P30N airfoil due to slat and flap gap/overhang and Reynolds number. The studies concluded that the optimum slat deflection angle was 30° and that the gap/overhang position was sensitive even at high chord-based Reynolds numbers ($Re_c < 4 \times 10^6$). The optimum flap deflection angle was found to be 30° producing maximum lift coefficient $C_{L,max}$ excess of 4.5. Even though higher flap deflection angles produced higher lift coefficient $C_L$, it resulted in the large-scale flow separation over the flap [41, 42]. Therefore, for the current experimental study, the geometrical parameters for the flap and slat locations were determined from these optimisation studies and it is detailed in Fig. 3.7 and Table 3.1. It is also noteworthy that most of the experimental and computational studies on the 30P30N airfoil in the past decade were carried out for the above-mentioned landing configuration, shown in Table 3.1. The coordinates system used to describe the high-lift airfoil is shown in Fig. 3.9, where $x$ and $y$ are the streamwise and crosswise axes from the retracted leading edge point of the airfoil and $z$ is the spanwise axis from the mid-span location of the airfoil.

![Figure 3.8: Slat close up view of the 3D printed interchangeable leading edge.](image)

In order to perform steady surface pressure measurements, the airfoil was equipped with extensive amounts of static pressure taps. The static pressure tap locations on the airfoil are shown in Fig. 3.9 and are listed in Table 3.2. The pressure taps were made from brass tubes with an outer diameter of 1.6 mm and an inner diameter of 0.6 mm. The pressure taps were installed along the span of the airfoil into 2 mm × 2 mm grooves and adhered with the Belzona® 1111 composite glue based on a ceramic steel reinforced polymer. At the mid-span of the airfoil, a hole of 0.4 mm was applied on the pressure taps. The slat is equipped with 23 pressure taps, one of which is present on the 3D printed interchangeable leading edge. The main-element is equipped with 41 pressure taps, while the flap has 39 pressure taps. In total the three-element
Figure 3.9: Static pressure taps and surface pressure transducer location on the MDA 30P30N airfoil with span of $l = 0.53$ m and a retracted chord of $c = 0.35$ m.

The unsteady surface pressure measurements were carried out using miniature Knowles FG-3329-P07 pressure transducers. A brief explanation of the technique used for installation and measurement is provided in Section 3.2.5. The placement of the pressure transducers on the airfoil was chosen based on the existing literature [70]. The recent experimental study by JAXA [70] has provided a comprehensive report on the noise source maps around MDA 30P30N airfoil giving us an overview of the critical regions on the 30P30N airfoil. The pressure transducers locations are listed in Table 5.4 and shown in Fig. 3.9. The main-element transducers $M1 - M5$ are placed in a location where there would be a possible impingement of flow from the slat to the main-element at high angles of attack. Murayama et al. [70] has also shown that the pressure transducers on the main-element are sufficient enough to capture the pressure fluctuations within the slat cove. The pressure transducer $M6$ was placed to acquire the convective velocity and streamwise correlation between the main-element leading and trailing edge. The flap has two rows of pressure transducers $F1 - F5$ and $F6 - F10$ along the span located at two different streamwise locations. All of the mentioned pressure transducers locations are shown in Fig. 3.9.

In order to get a smoothly distributed distance for all sensor pairs and to cover a large range of length scales [71], the spanwise spacing of the transducers was arranged with reference to a potential function $z_i/z_{max} = a^i/a^N$, $i = 1..N$, where $a$ is the coefficient value and $N$ is the maximum
### 3.1. MODEL CONFIGURATIONS AND INSTRUMENTATIONS

Table 3.2: Static pressure taps locations along the mid-span location of the MDA 30P30N airfoil model with a retracted chord length of $c = 0.35$ m.

<table>
<thead>
<tr>
<th>No.</th>
<th>Slat Upper $(mm)$</th>
<th>Slat Lower $(mm)$</th>
<th>Main-Element Upper $(mm)$</th>
<th>Main-Element Lower $(mm)$</th>
<th>Flap Upper $(mm)$</th>
<th>Flap Lower $(mm)$</th>
</tr>
</thead>
<tbody>
<tr>
<td>1</td>
<td>-4.613</td>
<td>-3.955</td>
<td>15.126</td>
<td>17.622</td>
<td>305.761</td>
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</tr>
<tr>
<td>2</td>
<td>-8.117</td>
<td>-5.328</td>
<td>17.374</td>
<td>23.851</td>
<td>309.236</td>
<td>307.043</td>
</tr>
<tr>
<td>3</td>
<td>-12.15</td>
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<td>29.778</td>
<td>312.655</td>
<td>309.161</td>
</tr>
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<td>4</td>
<td>-17.348</td>
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<td>36.991</td>
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<tr>
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<td>73.26</td>
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</tr>
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<td>-25.733</td>
<td>-14.847</td>
<td>72.672</td>
<td>102.483</td>
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<td>131.551</td>
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<td>336.338</td>
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<td>10</td>
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<td>131.551</td>
<td>160.601</td>
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<td>11</td>
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<td>393.483</td>
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<td>290.782</td>
<td>394.871</td>
<td></td>
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<td></td>
<td>292.792</td>
<td></td>
<td></td>
<td>292.84</td>
</tr>
</tbody>
</table>

distance to be resolved.

The flow field around the high-lift airfoil was captured using the Particle Image Velocimetry (PIV) technique. A detailed explanation of the PIV technique used for measurement is provided in Section 3.2.3. A total of 10 camera windows around the high-lift airfoil was used for capturing the flow field. The airfoil was inverted during measurement to capture the flow over the suction surface. The windows had two different sizes, the first one had a size of 6.3 cm × 6.3 cm and the second one had a size of 9.5 cm × 9.5 cm. The measurement windows around the airfoil are shown in Fig. 3.10, where the windows on the pressure side are shown by dashed lines and the windows on the suction side is shown by solid lines. The airfoil was covered in a self-adhesive black vinyl sheet with a matt finish to reduce surface reflection, as shown in Fig. 3.11.
CHAPTER 3. EXPERIMENTAL AND COMPUTATIONAL SETUP

Table 3.3: Microphone locations on the MDA 30P30N airfoil.

<table>
<thead>
<tr>
<th>No.</th>
<th>$x$ (mm)</th>
<th>$z$ (mm)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Main-Element</td>
<td>M1</td>
<td>22.414</td>
</tr>
<tr>
<td></td>
<td>M2</td>
<td>22.414</td>
</tr>
<tr>
<td></td>
<td>M3</td>
<td>22.414</td>
</tr>
<tr>
<td></td>
<td>M4</td>
<td>22.414</td>
</tr>
<tr>
<td></td>
<td>M5</td>
<td>22.414</td>
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<tr>
<td></td>
<td>M6</td>
<td>239.701</td>
</tr>
<tr>
<td>Flap</td>
<td>F1</td>
<td>308.844</td>
</tr>
<tr>
<td></td>
<td>F2</td>
<td>308.844</td>
</tr>
<tr>
<td></td>
<td>F3</td>
<td>308.844</td>
</tr>
<tr>
<td></td>
<td>F4</td>
<td>308.844</td>
</tr>
<tr>
<td></td>
<td>F5</td>
<td>308.844</td>
</tr>
<tr>
<td></td>
<td>F6</td>
<td>349.301</td>
</tr>
<tr>
<td></td>
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<td></td>
<td>F8</td>
<td>349.301</td>
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<td>F9</td>
<td>349.301</td>
</tr>
<tr>
<td></td>
<td>F10</td>
<td>349.301</td>
</tr>
</tbody>
</table>

Figure 3.10: The camera window locations used for PIV measurements of the 30P30N high-lift airfoil.

Figure 3.11: Reduced surface reflection during PIV by the use of self adhesive black vinyl sheet.
3.1.4 Slat cove filler design

As part of the noise reduction study of the MDA airfoil, a slat cove-filler (SCF) was designed using a similar strategy introduced by Imamura et al. [72, 73] for experimentation purposes. Initially, a preliminary Reynolds-averaged Navier–Stokes (RANS) steady-state simulation (see Appendix A) for the Baseline case was performed at the angle of attack $8.5^\circ$. The slat shear layer trajectory profile with high turbulent kinetic energy (TKE) extracted from the results were used to define the shape of the SCF profile (see Fig. 3.12). Another configuration with a Half slat cove filler (H-SCF) was also considered, which exhibits good aerodynamic and noise reduction properties, as shown computationally by Tao [74]. Both the slat cove-fillers and the flap cove-fillers (SCF-FCF) were manufactured using 3D printing technology and were made in four different sections that could be slid along the span of the slat and flap cove. The solid SCF is fitted with 6 pressure taps and the solid FCF is fitted with 3 pressure taps along the span of the high-lift airfoil for surface pressure measurements. A schematic of the different 30P30N high-lift airfoil configurations used in the present study are shown in Fig. 3.13.

![TKE RANS, $\alpha = 8.5^\circ$](image1)

![SCF Insert](image2)

Figure 3.12: (a) Turbulent kinetic energy contours indicating slat shear layer profiles around 30P30N airfoil slat for an angle of attack, $\alpha = 8^\circ$ at $Re_c = 1.7 \times 10^6$ and (b) The 3D printed SCF fitted on the 30P30N airfoil in the low turbulence wind tunnel
3.2 Measurement techniques

To unravel the flow physics around the NACA 0012 airfoil and 30P30N high-lift airfoil various flow and noise measurement techniques were employed in the current study. In this section a detailed description of the flow and noise measurement techniques used such as, force balance measurement setup, pressure measurement setup, particle image velocimetry setup, oil flow visualization setup, unsteady surface pressure measurement setup and far-field measurement setup are presented. The steps taken to minimise the uncertainty level of the experimental results are also discussed in detail.

3.2.1 Force balance measurement setup

The aerodynamic lift (L) and drag (D) force measurements were carried out using an AMTI OR6-7-2000 3-axis force platform from Advanced Mechanical Technology Inc. The airfoil was secured to the force platform with two teardrop shaped metal side-arms to minimise the drag forces on the support arms, see Fig. 3.4. The force platform records the force-induced voltage,
which is measured through the AMTI MSA-6 strain gauge amplifiers and processed using the in-built LabView V18 system design software. The data were collected through a 16-bit A/D card for a period of 16 seconds at a sampling frequency of 2000 Hz. The uncertainty analysis was performed using the bootstrap technique [75] and was found to be $\approx 4.5\%$ for all the tested angles of attack. The lift coefficient $C_L$ and drag coefficient $C_D$ are calculated by,

\[
C_L = \frac{L}{0.5\rho U_\infty^2 S},
\]

\[
C_D = \frac{D}{0.5\rho U_\infty^2 S},
\]

where $L$ is the lift force acting on the direction normal to the airfoil, $D$ is the drag force acting on the streamwise direction of the airfoil and $S$ is the wing section area of their respective airfoil model.

Figure 3.14: Probability density function of bootstrap standard deviation for the force platform signal, where $\sigma_{\text{boot}}$ is the bootstrap standard deviation.
3.2.2 Pressure measurement setup

The static pressure measurements around the airfoil were carried out using MicroDaq pressure scanners manufactured by Chell Instruments, shown in Fig. 3.16. The scanners have a full-scale measuring capacity of 1 Psi with a system accuracy of ±0.05%. Each MicroDaq system has 32 channels and two scanners in parallel with a total of 64 channels were used for the pressure measurements. The pressure taps were made from brass tubes with an outer diameter of 1.6 mm and an inner diameter of 0.6 mm. A hole with a diameter of 0.4 mm was made on the surface of the pressure taps at the mid-span location of the airfoils for pressure measurement (see Fig. 3.16). The scanners were connected to the pressure taps using clear PVC tube with an internal diameter of 0.8 mm resulting in a tight fit. The scanners were always connected to a pitot tube to calculate the free-stream stagnation and static pressure (used as reference pressure). The pitot tube was placed away from the tunnel wall boundary layer. To obtain accurate pressure distribution results, measurements were carried out for a period of 60 seconds with a sampling frequency of 500 Hz. The uncertainty analysis for the collected data at all the tested angles of attack were found to be ≈2−9% depending upon the pressure tap location with increased values at the suction peak locations. The averaged measured data were used to calculate the $C_p$ pressure coefficient using,

$$C_p = \frac{\overline{p} - p_\infty}{0.5\rho U_\infty^2} = \frac{\overline{p} - p_\infty}{\overline{p_0} - p_\infty},$$

(3.3)

where $p$ is the measured static pressure at the measurement ports, $p_0$ is the measured stagnation pressure and $p_\infty$ is the free-stream static pressure.
3.2. MEASUREMENT TECHNIQUES

3.2.3 Particle Image Velocimetry setup

The flow field around both the NACA 0012 airfoil and 30P30N high-lift airfoil were studied using two-dimensional ($x - y$ plane) two-component Particle Image Velocimetry (PIV) in the low turbulence closed-circuit wind tunnel. The low turbulence wind tunnel is equipped with clear perspex bottom window, through which the PIV laser sheet enters the tunnel to illuminate the seed particles (see Figs. 3.11 and 3.17) and a clear perspex side windows through which the camera captures the images. The PIV experimental setup is shown in Fig. 3.17 with the laser sheet positioned on the mid-span of the airfoil models parallel to the flow and the camera mounted on a traverse system placed aside the wind tunnel's side window perpendicular to the laser sheet.

Table 3.4: The PIV setup parameters used in the current study.

<table>
<thead>
<tr>
<th>Setup parameters</th>
<th>NACA 0012 and 30P30N</th>
<th>units</th>
</tr>
</thead>
<tbody>
<tr>
<td>Snapshot time interval</td>
<td>9-15</td>
<td>(Δ$t$), µs</td>
</tr>
<tr>
<td>Repetition rate</td>
<td>10-15</td>
<td>Hz</td>
</tr>
<tr>
<td>Field of view</td>
<td>63 × 63 and 95 × 95</td>
<td>mm²</td>
</tr>
<tr>
<td>Interrogation window</td>
<td>16 × 16</td>
<td>pixel</td>
</tr>
<tr>
<td>Interrogation window overlap</td>
<td>50</td>
<td>%</td>
</tr>
<tr>
<td>Spatial resolution</td>
<td>32.5 and 21.5</td>
<td>pixel/mm</td>
</tr>
<tr>
<td>Vector spacing</td>
<td>0.5 and 0.74</td>
<td>mm</td>
</tr>
</tbody>
</table>
A Dantec DualPower 200 mJ Nd:YAG laser with a wavelength of 532 nm was used to produce 1 mm thick laser sheet with a time interval between each snapshot of 9-15 μs and a repetition rate of 10-15 Hz. A mixture of Polyethylene glycol 80 with a mean diameter of 1 μm was used to seed the air inside the low turbulence wind tunnel. A total of 2400 images for each measurement was captured using a FlowSense 4 MP CCD camera with a resolution of 2048 × 2048 pixels and 14 bit. The corresponding field of view are 63 mm × 63 mm and 95 mm × 95 mm, as illustrated in Fig. 3.5 and 3.10. The uncertainty in the pixel displacements was found to be below ≈ 1% [76]. The images were analysed with the DynamicStudio software from Dantec. The iterative process yielded grid correlation window of 16 × 16 pixels with an overlap of 50%, resulting in a field vector spacing of 0.5 mm and 0.74 mm for the two fields of view, respectively. The PIV setup parameters used in the current study is summarised in Table 3.4

Figure 3.17: The PIV measurement setup (a) showing the camera and model setup and (b) a close-up view of the laser sheet and illuminated particles.
3.2.4 Oil flow visualization

To capture the surface flow pattern over the airfoil, the oil-flow visualization technique was used. The airfoil was covered with matt black vinyl sheet to visualize the flow better. A mixture of paraffin and oleic acid in a ratio of 20:1 along with the white pigimenting agent titanium dioxide was used in the present study. In order to effectively capture the boundary layer behavior, the consistency of the oil mixture was tested to have the inertial forces of the oil mixture lower than viscous and surface tension forces [77, 78].

At first, the oil-flow visualization mixture was applied on the airfoil and then the wind tunnel was operated for 60 seconds. The airflow evaporates the paraffin in the flow visualization mixture leaving the oily oleic acid and the white pigmented titanium oxide behind in flow patterns over the airfoil surface. The high-quality images of the surface flow patterns were then captured. The airfoil was then thoroughly cleaned and the procedure was repeated for a range of angles of attack $\alpha = -5^\circ$ to $15^\circ$ on both the suction and pressure side of the airfoil.

3.2.5 Unsteady surface pressure measurements

FG-3329-P07 pressure transducer from Knowles Electronics (see Fig. 3.18) were used for the measurement of the unsteady surface pressure over the airfoil. These surface pressure transducers due to their excellent frequency response were proven successful in capturing near-field aeroacoustic in several previous experimental studies [79–83]. The transducer has a diameter of 2.5 mm and a height of 2.5 mm with a sensing area of 0.8 mm. In order to reduce the measurement errors that arise due to the spatial integration of the signal, a surface fairing with a reduced sensing area of 0.4 mm was used, as shown in Fig. 3.18. The surface fairing was manufactured using 3D printing technology. The pressure transducers along with the surface fairing were flush mounted to the surface of the airfoil as shown in Fig. 3.19. The FG-3329-P07 transducer has a manufacturer provided a sensitivity of 22.4 mV/Pa (45 Pa/V) in the flat region of the transducer response. The sensitivity for the current experiments are calculated from an in situ calibration setup for the surface pressure transducers, which is explained in Section 3.2.6. The nominal frequency of the pressure transducer and the frequency response with the surface fairing is shown in Fig. 3.22.
Figure 3.18: Pressure transducer used for unsteady surface pressure measurements, (a) Close up view of the stock FG-3329-P07 pressure transducer from Knowles Electronics, (b) FG-3329-P07 covered with the 3D printed surface fairing, (c) FG-3329-P07 beside the surface fairing and a scale for size comparison and (d) schematic of the surface fairing dimensions.

Figure 3.19: Surface mounted FG-3329-P07 pressure transducer and pressure taps on the main-element and slat of the 30P30N high-lift airfoil, respectively.
3.2.6 Surface pressure transducer calibration

The FG-3329-P07 transducers are calibrated based on a procedure formulated in previous studies [90, 91]. The calibration was carried out by comparing the signal of the FG-3329-P07 transducer against the signal from a reference GRAS 40PL microphone with a known sensitivity. The reference microphone has a manufacturer provided frequency response between 10 Hz and 10 kHz in the flat region of the microphone response (see Fig. 3.25). The sensitivity and the transfer function values of the FG-3329-P07 transducers were acquired using a calibrator, as shown in Fig. 3.20. The calibrator unit consists of an extension cone, an extension metal tube, loudspeaker, reference microphone GRAS 40PL and acoustic termination tube (see Fig. 3.20). The cone was manufactured with the smallest angle possible to ensure a plane wave propagation through the whole channel. In order to ensure the attenuation of higher-order modes and any external noise, the cone was filled with the polyurethane porous foam material. A small microphone holder was placed at the end of the extension metal tube, where the reference microphone was mounted at the circumference of the metal tube, with a short silicon tube placed at the other end of the microphone holder to transmit the pressure waves to the FG-3329-P07 transducers. This allows both of the transducers to be held at an equal distance from the central axis of the cone. A long extension tube is connected to the microphone holder for the purpose of acoustic termination of the noise traveling through the extension metal pipe. The flow chart describing the calibration procedure is shown in Fig. 3.21.

![Calibration Setup Diagram](image)

Figure 3.20: FG-3329-P07 calibration setup used for in situ calibration.
The FG-3329-P07 transducer calibration method was performed by acquiring data from the reference GRAS microphone and the surface pressure FG-3329-P07 transducer. Simultaneous measurement of the broadband white noise source \( V_a^1(t) \) and the GRAS microphone signals \( V_a^2(t) \) was carried out to measure the response of the speaker \( S_p(f) \). The output signal from the speaker, \( S_p(t) \) can be found from,

\[
S_p(t) = \frac{V_a^2(t)}{S_{GRAS}} [Pa],
\]

where \( S_{GRAS} \) is the broadband sensitivity response of the GRAS microphone. The speaker response in the frequency domain can be calculated from,

\[
S_p(f) = \frac{E[P_{S_p}(f) \cdot V_a^2(f)]}{E[V_a^2(f) \cdot V_a^2(f)]} = \frac{G V_a^2 V_a^2}{G V_a^2 V_a^2} \cdot \frac{1}{S_{GRAS}} \cdot \frac{1}{V_a^1} [Pa],
\]

where \( E \) is the expected operator value, \( G \) is the cross-spectrum between the white noise and GRAS microphone output signals.

A simultaneous measurement of the broadband white noise source \( V_b^1(t) \) and the FG-3329-P07 transducer signals \( V_b^2(t) \) was performed. The FG-3329-P07 transducer frequency response can be found from,
Since $S_p(f)$ is known from Eq. 3.5, the FG-3329-P07 transducer frequency response can be calculated from,

$$S_{FG}(f) = \frac{G_{V_2^b V_1^b}}{G_{V_1^b V_1^b}} \cdot \frac{G_{V_1^b V_0^b}}{G_{V_2^b V_1^b}} \cdot S_{GRAS} \left[ \frac{V}{Pa} \right].$$ \hspace{1cm} (3.7)
CHAPTER 3. EXPERIMENTAL AND COMPUTATIONAL SETUP

3.2.7 Far-field measurement

The experimental setup of the high-lift airfoil for the aeroacoustic measurements at the University of Bristol aeroacoustic facility is shown in Fig. 3.23. The far-field noise measurements were carried out using an array of 22 GRAS 40PL piezoelectric free-field microphones distributed over a circular arc at a radius of 1.75 m from the trailing edge of the slat in the aeroacoustic wind tunnel. A close-up view of the GRAS microphone attached to the arc is shown in Fig. 3.24. The GRAS microphone transmits the voltage signal to the National Instrument PXIe-4499 data acquisition system through a 15 m coaxial SMB (Sub-Miniature version B)-BNC (Bayonet Neill–Concelman) cable that has an excellent electrical performance from DC to 4 GHz. The microphone array covers a range of polar angles between 35° to 140°, with a regular interval of 5°. The microphone has a flat frequency response at frequencies from 10 Hz to 10 kHz, with a dynamic range of 142 dB (see Section 3.2.8). The far-field noise spectra were computed for 16 seconds using a sampling frequency of $f = 2^{16}$ Hz. The acoustic data were recorded for a wide range of flow velocities of up to 40 m/s.

The pressure spectrum results were obtained using the power spectral density (PSD) of the pressure signals with Hanning window and the acquired data were averaged for 200 times to yield a frequency resolution of $\Delta f = 6.25$ Hz. The sound pressure level (SPL) spectrum can be calculated using,

$$\text{SPL} = 20 \cdot \log_{10} \left( \frac{p_{RMS}}{p_{Ref}} \right), \quad (3.8)$$

where $p_{RMS}$ is the root-mean-square of the acoustic pressure and $p_{Ref}$ is the reference pressure at 20 $\mu$Pa. The power spectrum of the acoustic pressure signal is corrected to a reference distance of 1 m.

The overall sound pressure level (OASPL) are obtained using,

$$\text{OASPL} = 10 \cdot \log_{10} \left[ \frac{\int PSD(f) \, df}{p_{Ref}^2} \right], \quad (3.9)$$

where PSD is the power spectral density based on the unsteady pressure $p'(t)$ (where $p'(t) = p(t) - \bar{p}$, see Section 3.3). The resolved frequency $f$ ranges from 100 Hz to 32 kHz.
3.2. MEASUREMENT TECHNIQUES

Apart from the SPL and OASPL noise, the measurements in the anechoic chamber enables a wide range of other fundamental research, such as, spanwise coherence, near-field to far-field coherence, bi-coherence and persistence spectrum. The results are provided and discussed in Chapter 5.

Figure 3.23: Test model mounted in the aeroacoustic wind tunnel at the University of Bristol.

Figure 3.24: GRAS 40PL far-field microphone setup.
3.2.8 Far-field microphone calibration

The GRAS 40PL microphones are used in this study for the far-field noise measurements of the high-lift airfoil. The calibration of the GRAS 40PL microphones were carried out using the GRAS pistonphone Type 42AA, shown in Fig. 3.25. The pistonphone produces a constant sound pressure level of 114 dB within ±0.5 dB, under reference condition, which induces a pressure of 10 Pa at 250 Hz (or equivalent to 105.4 dB(A)). The pistonphone is operated by battery and it works by generating a sinusoidal pressure signals using a precision-machined rotating cam actuating two reciprocating pistons and it has a accuracy of ±0.5 dB at reference pressure of 20 µPa [92]. The pistonphone is limited to a low frequency of 250 Hz, this output voltage was also calculated using fast Fourier Transform (FFT) and the obtained amplitude was used to calculate the microphone sensitivity \( S_{GRAS} \) from,

\[
S_{GRAS} = \frac{V_{pp}}{20 \cdot 10^{-6} \cdot 10^{L_{pp}/20}}.
\]  

(3.10)

Where \( V_{pp} \) is the output voltage of the pistonphone and \( L_{pp} \) is the sound pressure level generated by the pistonphone. The pressure correction \( p_{corr} \) can be calculated from,

\[
p_{corr} = 20 \cdot \log_{10} \left( \frac{p_{amb}}{p_{Ref}} \right) \ [dB],
\]  

(3.11)

Where \( p_{amb} \) and \( p_{Ref} \) are the ambient and reference pressure, respectively. When necessary the ambient pressure corrections could be made and it is measured using the manufacturer provided Barometer ZC0002K shown in Fig. 3.25. The correction factors are not applied for the current study at a static ambient pressure of 101.3 kPa.
3.3. DEFINITIONS OF MEASUREMENT QUANTITIES

The results are always presented after normalisation. The mean values of the streamwise ($U$) and crosswise ($V$) velocities are calculated by averaging the velocity time signals $u(t)$ and $v(t)$ as,

\[ U = \frac{1}{N} \sum_{i=1}^{N} u(t_i), \]
\[ V = \frac{1}{N} \sum_{i=1}^{N} v(t_i). \]  

(3.12)

The velocity fluctuations, $u'(t)$ and $v'(t)$ are evaluated by subtracting the mean component of the velocity $U$ and $V$ as,

\[ u'(t) = u(t) - U, \]
\[ v'(t) = v(t) - V, \]  

(3.13)

where $N$ is the total number of the data samples in time and $t_i$ refers to the time history of the collected data.

Figure 3.25: (a) GRAS 40PL calibration device and (b) GRAS frequency response spectra where solid line is the free field response and dashed line is the pressure response.
The streamwise $u'u'$ and crosswise $v'v'$ Reynolds stress and the Reynolds shear stress $u'v'$ components were calculated from,

$$
\begin{align*}
\overline{u'u'} &= \frac{1}{N} \sum_{i=1}^{N} u'(t_i)u'(t_i), \\
\overline{v'v'} &= \frac{1}{N} \sum_{i=1}^{N} v'(t_i)v'(t_i), \\
\overline{u'v'} &= \frac{1}{N} \sum_{i=1}^{N} u'(t_i)v'(t_i),
\end{align*}
$$

(3.14)

The mean and fluctuating velocity components are normalized by dividing it by the mean free-stream velocity $U_{\infty}$. The turbulent kinetic energy (TKE) $k$ is calculated using,

$$
k = \frac{1}{2}(u'u' + v'v').
$$

(3.15)

The Reynolds stress components and the turbulent kinetic energy are normalized by dividing them by the mean free-stream velocity squared $U_{\infty}^2$. The mean and root mean square of the pressure components are defined as,

$$
P = \frac{1}{N} \sum_{i=1}^{N} p(t_i),
$$

(3.16)

and

$$
P_{RMS} = \sqrt{\frac{1}{N} \sum_{i=1}^{N} (p'(t_i))^2},
$$

(3.17)

where $p'(t)$ is the pressure time signal, $P$ is the mean pressure, $N$ is the total number of the data samples in time, $t_i$ refers to the time history of the collected data and $p'(t) = p(t) - P$ is the pressure fluctuation.
3.4 Computational setup

The basic theory and idea of Large Eddy Simulation (LES) was formulated by Smagorinsky [84]. The LES method resolves the eddies in the flow that are larger than the grid used in the computational domain and the eddies smaller than the grid are modelled using an analytical method. It is important to directly calculate the large energy scales since most of the flow energy transport, whereas the smaller energy scales behave more uniformly and thus can be easily modelled. In the Smagorinsky model, the eddy viscosity $v_{SGS}$ is obtained by assuming that the small scales are in equilibrium, so that energy production and dissipation are in balance, which yields:

$$v_{SGS} = 2\bar{\rho}(C_s \Delta)^2 |S| = 2\bar{\rho}(C_s \Delta)^2 (2S_{ij}S_{ij})^{1/2},$$ \hspace{1cm} (3.18)

where $C_s$ is the Smagorinsky constant, $\Delta$ equals the filter cutoff width, i.e. the characteristic length scale of the SGS eddies and $|S|$ represents the absolute value of the shear strain tensor. When using the Smagorinsky model, the suggested value of the constant $C_s$ ranges between 0.065 and 0.25. Due to the limitations of the Smagorinsky model, a more general dynamic subgrid-scale model [85] was developed. The dynamic model allows the Smagorinsky constant $C_s$ to vary in space and time. $C_s$ is calculated locally at each timestep based upon two filtering (grid filter and the test filter) applied to the flow variables.

Figure 3.26: Large Eddy Simulation with scale separation [104].
Large Eddy Simulation based on the spatially filtered, incompressible Navier-Stokes equations with the dynamic Smagorinsky subgrid-scale model \cite{85} was implemented in OpenFOAM V2.0.1 for the NACA 0012 airfoil. A ‘pressure implicit with splitting of operator’ (PISO) algorithm was used to resolve the incompressible Navier–Stokes. The convective fluxes were obtained by employing a second-order discretization method. A second-order centered scheme was used for the viscous terms and a second-order implicit time-stepping method was used to estimate the temporal terms. Reynolds averaged Navier–Stokes (RANS) and Detached Eddy Simulation (DES) studies for the NACA 0012 airfoil with and without morphed trailing edges were previously completed to comprehend their aerodynamic and aeroacoustic characteristics \cite{64–67}.

Unlike the previous studies \cite{64–67}, in the current LES work, the airfoil was tripped to avoid any potential laminar boundary layer instability issues. A step-trip with a height of 0.8 mm (0.004c) and a length of 3 mm (0.015c) was placed at the location 0.1c downstream of the leading edge on both the sides of the airfoil. The airfoil was well spaced within a three dimensional C-H type computational domain with 10 chord length in the streamwise direction and 5 chord length in the crosswise direction as shown in Fig. 3.27. The three-dimensional domain had a spanwise thickness of 0.1 chord length. The span length was deemed sufficient based on several other similar previous studies \cite{86–89, 104}. The step-trip airfoils had a cell distribution of $L_x \times L_y \times L_z = 704 \times 40 \times 64$. The airfoil was set to have a $y^+ \approx 0.5 - 1$, which corresponds to an airfoil wall distance of $y = 0.035c$ populated with 40 grid points. The far-field region around the airfoil had a cell distribution of $L_x \times L_y \times L_z = 352 \times 150 \times 32$. A grid spacing resulting in a $x^+ \approx 15$ and $z^+ \approx 20$ was set along the streamwise and spanwise direction, respectively. The mesh in the streamwise direction was clustered toward the leading and trailing edge of the airfoil and the wake region (up to 1.5c) was further refined with 400 grid points to capture the wake behavior accurately. The final computational mesh had 11.5 million cells in total (see Fig. 3.28). A time-step of $\Delta t = 2.2 \times 10^{-6}$ s was employed to maintain the Courant-Friedrichs-Lewy value below $C_{max} \leq 1$. The simulations were carried out for 30 flow through time (FTT), which corresponds to 0.30 seconds. The LES data were captured only for a period of 10 FTT with a time interval of $\Delta t = 2.2 \times 10^{-6}$ s for the purpose of acoustic analysis.
3.4. COMPUTATIONAL SETUP

Figure 3.27: An overview of the LES computational domain and setup with a close up view of the Morphed Flap airfoil trailing edge mesh.

Figure 3.28: Grid refinement close to the airfoil wall and the wake regions to capture the boundary layer transition over the flaps accurately.
3.5 Summary

The chapter provided a brief description of the experimental setup, instrumentation of the airfoils and the measurement techniques employed in the present study. A wide range of experiments were carried out in the present study to capture the various aspects of the aerodynamic and aeroacoustic behavior of the two tested airfoils, namely, a NACA 0012 symmetric airfoil and 30P30N high-lift airfoil. The manufacturing techniques and the instrumentation of the airfoils were described in detail. This chapter described the multiple measurement techniques used such as the force balance, static pressure measurement, unsteady pressure measurement, Particle Image Velocimetry and far-field noise measurements in detail. The experimental uncertainty for each of the measurement technique is described in detail in its respective sub-sections. The calibration of the near- and far-field microphones were discussed in detail. This chapter also reported the three different wind tunnels in detail, which is used for studying the performance of the airfoils. A general introduction to the LES and the Smagorinsky model used for the simulations were given. The computational domain and setup used for the LES simulation were also described in detail.
CHAPTER 4

MORPHED TRAILING EDGES

4.1 Introduction

The advent of shape-adaptive structures with reduced weight and complexity have improved the performance of wind turbine blades and aeroplane wings. The ability of these shape-adaptive structures to remain conformal to the flow by maintaining continuous smooth geometric changes have increased their appeal within the field of aerodynamics. These compliant lightweight control surfaces are increasingly known as morphing structures. As such, significant noise and drag reduction are envisaged through morphing structures. It is, therefore, of fundamental importance in the concept synthesis of morphing structures to thoroughly investigate the flow behavior and mechanisms of performance improvement.

Several studies on the implementation of morphing structures on airfoils have shown that the smooth airfoil curvature has significant effects on their aerodynamic performance. The full-scale capabilities of morphing wing with improved lift performance during takeoff and landing along with its drag reduction capabilities were shown in NASA’s projects, such as the Elastically Shape Future Vehicle project [94] and Adaptive Compliant Trailing edge [95, 96]. The significant aerodynamic benefits of a smooth variable camber flap were presented by Hetrick et al. [95], where the results showed a 3.3% improvement in the lift-to-drag ratio and a possible 15% savings
in the fuel costs. The study also showed that variable camber flaps required 33% less actuation force and 40% increased control authority compared to hinged flaps. The aerodynamic studies using biologically inspired Fish Bone Active Camber morphing flap in comparison with hinged flap by Woods et al. [30] showed improvements in the maximum obtainable lift-to-drag ratio in the order of 20 – 25% for the morphing flap. The ability of the morphing flap to produce the same lift as that of the hinged flap with a 30% less flap tip deflection for the same flap length was illustrated by Daynes et al. [26, 27]. Wind tunnel tests by Yokozeki et al. [28] using morphing airfoil made from corrugated structures demonstrated preferable high lift coefficients compared to the conventional flap. This superior performance was believed to be due to the seamless morphing deformation. An innovative trailing edge morphing mechanism that uses a honeycomb core with axial variable stiffness developed by Ai et al.[61, 62], was used as the morphing profile for the experimental and computational studies carried out by Jawahar et al. [64–67]. The results showed the superior aerodynamic capabilities of the morphing flap. The results also showed that the surface camber of the morphing flap has a high influence on the aerodynamic performance of the airfoil and that the effect of the morphing flap camber increases with increasing flap deflection.

Most of the studies available on morphing airfoils have focused on the morphing mechanism and the internal structures [26–28, 30, 61, 62, 94–99] than the aerodynamic characteristics of the airfoil. Previous studies have presented only the basic aerodynamic characteristics such as the lift and drag coefficients. A recent study by Jawahar et al. [67] showed that significant improvement in the aerodynamic behavior of the airfoil can be brought about by the application of morphed flap compared to hinged flap. The study also showed that the flap camber profile played an important role in achieving improved aerodynamic behavior. The lift and drag for the morphed airfoil with high flap camber resulted in increased lift and drag compared to the airfoil with low flap camber and hinged flap. The boundary layer behavior showed delayed separation for the morphed flap airfoil relative to the conventional hinged flap configuration. The turbulence kinetic energy levels at the wake were also found to be significantly altered with the morphed flap having higher intensity compared to hinged flap configuration. This study concluded that independent morphing of the airfoil’s upper and lower surfaces would aid the favourable delayed
4.1. INTRODUCTION

separation while reducing the unfavourable increased drag. The use of morphing trailing edges has been proposed for aerodynamic improvement and trailing edge noise reduction in multiple previous studies [64–67, 100–102]. The use of morphing surfaces aims to address transition and separation delay, lift enhancement, drag reduction, turbulence augmentation and noise suppression [103]. However, the ability of the morphing trailing edges to reduce noise is yet to be proven and documented. An ideal method of morphing should achieve the control goal without affecting other goals adversely. However, in reality, continuous compromises and trade-offs have to be made for a particular design goal as it is almost impossible to decouple the interlinked flow behavior [103], i.e. lift and drag forces, and noise emission in the case of the high-lift systems.

Complimenting the previous studies, the current experimental and numerical study investigates the aerodynamic and aeroacoustic behavior of a morphed and hinged NACA 0012 airfoil. This chapter presents a detailed experimental aerodynamic study with the lift and drag measurements for hinged and morphed airfoil in Section 4.2.1. Then Section 4.2.2 presents a detailed account of the flow separation regions at different angles of attack using surface flow visualization technique. The flow development in the airfoil was studied using Particle Image Velocimetry and it is presented in Section 4.2.3. In order to further study in detail the unsteady flow characteristics over the airfoil surface and at the airfoil wake, high fidelity Large Eddy Simulations were also performed. The pressure coefficient results for the two airfoils are shown in The results for the detail contours of the steady and unsteady velocity components at the airfoil wake are presented in Section 4.3.2. Section 4.3.3 discusses the boundary layer development over the airfoils. The unsteady wall pressure and far-field noise estimation for the tested morphed and hinged airfoils are presented in Sections 4.3.4 and 4.3.7
CHAPTER 4. MORPHED TRAILING EDGES

4.2 Experimental results

A symmetric NACA 0012 airfoil with Hinged and Morphed trailing edges configurations are investigated experimentally. The NACA 0012 airfoil had a chord length of \( c = 0.2 \) m and a span length of \( l = 0.45 \) m. A detailed explanation of the airfoil manufacturing and instrumentation process are discussed in Section 3.1.2. The Hinged Flap (HF) and Morphed Flap (MF) configurations with a flap length of \( b = 0.3c \) and a flap deflection angle of \( \beta = 10^\circ \) are shown in Fig. 4.1. The A serrated turbulator tape (see Fig. 3.2 in Section 3.1.2) was used on both sides of the airfoil surface at location \( x/c = 0.1 \) to ensure a turbulent boundary layer flow over the trailing edge area. As a part of this experimental study, lift and drag measurements, oil-flow visualization and Particle Image Velocimetry technique were employed. The results are discussed in detail below.

![](image)

Figure 4.1: Geometric details of the NACA 0012 airfoil with a flap deflection angle of \( \beta = 10^\circ \) named Hinged Flap and Morphed Flap airfoils.

4.2.1 Aerodynamic force measurements

The aerodynamic lift and drag measurements are crucial in order to evaluate the aerodynamic capabilities of any airfoil. AMTI OR6-7-2000 3−axis force platform was used to measure the aerodynamic life and drag force for the NACA 0012 airfoil configurations. A detailed explanation of the force measurement setup is provided in Section 3.2.1. The results of the aerodynamic force measurements for the NACA 0012 airfoil fitted with the Hinged Flap and Morphed Flap having a flap deflection angle of \( \beta = 10^\circ \) is presented in Fig. 4.2. The tests were carried out for the angles of attack ranging from \( \alpha = -5^\circ \) to \( 15^\circ \). The results of the lift coefficients \( C_L \) − \( \alpha \), presented in
4.2. EXPERIMENTAL RESULTS

Fig. 4.2a, show a better overall lift performance for the MF airfoil compared to the HF airfoil. The $C_L$-$\alpha$ curve of the MF airfoil has an average lift increase of $\Delta C_L = 0.19$ compared to the HF airfoil over the entire range of the tested angles of attack. The largest difference in the lift coefficient between the HF and MF airfoil, $\Delta C_L = 0.22$, was produced at $\alpha = -5^\circ$ and the lowest $\Delta C_L = 0.16$ was found at the stall angle of attack $\alpha = 13^\circ$, which corresponds to a 14% increase in the lift coefficient for the MF airfoil relative to the HF airfoil at $C_{L,max}$. The MF airfoil reduces the angle of attack for a given lift coefficient by $\alpha = 3^\circ$ compared to the HF airfoil. The stall angle of both the airfoils remained unchanged, which is consistent with the results in the literature [105].

The lift-to-drag coefficient ratio results and the drag polar curves for both the HF and MF airfoils are presented in Fig. 4.3. The lift-to-drag coefficient ratio results in Fig. 4.3a show a large difference between the HF and MF airfoil at low angles of attack. This difference in the aerodynamic performance between the two airfoils decreases as the angle of attack is increased.
The MF airfoil has an overall superior aerodynamic lift performance compared to the HF airfoil. There is no or insignificant difference in the $C_L/C_D$ performance between the airfoils for the angles of attack larger than $\alpha > 7^\circ$. The largest difference in the lift-to-drag coefficient ratio of $\Delta(C_L/C_D) = 0.98$ was found between the HF and MF airfoil at the angle of attack $\alpha = -5^\circ$ with the MF airfoil portraying better performance with larger values of $C_L/C_D$. As the angle of attack is increased to $\alpha = 0^\circ$ and $5^\circ$, the differences in the $\Delta(C_L/C_D)$ value reduces to 0.85 and 0.33, respectively, which corresponds to an increased performance of up to 26% and 6% for the MF airfoil relative to the HF airfoil. The largest value of $C_L/C_D = 5.60$ was achieved by the MF airfoil at $\alpha = 5^\circ$. The drag polar curves presented in Fig. 4.3b clearly illustrate the aforementioned phenomenon of the MF airfoil reducing the effective angle of attack for a given lift coefficient by $\alpha = 3^\circ$. The drag polar plot also shows that the MF airfoil produces slightly higher lift and a lower drag at $\alpha = 10^\circ$ compared to the $\alpha = 13^\circ$ of the HF airfoil.

### 4.2.2 Surface flow visualization

The oil-flow visualization technique was used on both the suction and pressure sides of the airfoil to visualize the boundary layer behavior and flow separation region over the airfoil flap surfaces. The airfoil setup and the oil flow visualization technique are discussed in detail in Section 3.2.4. The tests were carried out for the angles of attack ranging from $\alpha = -4^\circ$ to $16^\circ$ in increments of $2^\circ$ for a free-stream velocity of $U_{\infty} = 20$ m/s ($Re_c = 2.6 \times 10^5$). However, for the purpose of
4.2. EXPERIMENTAL RESULTS

brevity, the results are presented only for the suction side of the airfoil at the angles of attack of $\alpha = 0^\circ, 2^\circ, 4^\circ, 6^\circ$ and $8^\circ$, see Fig. 4.4. The flow visualization results for the airfoil pressure side are not presented due to the absence of any unique flow feature or separation on the pressure side of the airfoil flap surfaces for the entire range of tested angles of attack.

The photographs of the oil-flow visualization are presented in Fig. 4.4 at the vicinity of the flap with the red dotted line (---) denoting the flow separation location. The results of the HF and MF airfoils are presented in the left and right columns, respectively, with the different rows showing different angles of attack. The application of the serrated turbulator at $0.1c$ on the airfoil formed turbulent wedges downstream of the serrated turbulator, this was also observed in previous flow visualization studies [59, 60]. Small recirculation regions were found in-between the wedges of the serrated turbulator, which may have served to trip the flow and led to the formation of the turbulent wedges. These are seen as horizontal lines in the photographs in Fig. 4.4. The vertical lines seen in Fig. 4.4 are the shade arising due to the pressure taps that lay beneath the black vinyl used to cover the airfoil.

The surface flow visualization results over the flap trailing edge clearly show that for the HF configuration the boundary layer flow on the suction side of the airfoil separates at the hinge point of the flap at $x/c = 0.7$ for all the presented angles of attack. For the MF airfoil, the flow does not separate over the flap surface until $x/c = 0.95$ for the angles of attack of $\alpha = 0^\circ$ and $2^\circ$. As the angle of attack is increased, the flow separation point over the flap surface for the MF airfoil moves to upstream locations $x/c = 0.8, 0.76$ and $0.7$ for the angles of attack $\alpha = 4^\circ, 6^\circ$ and $8^\circ$, respectively. The results for the angles of attack $\alpha > 8^\circ$, the flow separation point moved to further upstream locations $x/c < 0.7$ and remained the same for both the HF and MF airfoil. The surface flow visualization results presented here clearly show that the shape of the flap strongly influences the flow separation over the flap region for both the HF and MF airfoil. The results also correspond to the increased $C_L/C_D$ seen in Section 4.2.1 for the MF airfoil compared to the HF airfoil at angles of attack $\alpha < 8^\circ$. This improved aerodynamic performance seen earlier could be due to the delayed separation observed in the surface flow visualization results.
Figure 4.4: The photographs of the oil-flow visualization patterns over the suction side of the Hinged Flap and Morphed Flap airfoil at the vicinity of the flap tested at a free-stream velocity of $U_\infty = 20$ m/s ($Re_c = 2.6 \times 10^5$) and \(\ldots\) denoting the flow separation location.
4.2. EXPERIMENTAL RESULTS

4.2.3 PIV flow visualization

In order to better understand the flow separation, flow structure and wake development of the HF and MF airfoils, PIV measurements over the flap region and airfoil wake were carried out. The PIV measurement setup was presented in detail in Section 3.2.3, see Fig. 3.17. The camera window location at the trailing edge and wake of the airfoil are shown in Section 3.1.2 in Fig. 3.5. The results of the mean velocity contours at the HF and MF airfoil flap and wake region, measured using the PIV technique are presented in Fig. 4.5. The tests were carried out for a free-stream velocity of $U_\infty = 20 \text{ m/s}$ at angles of attack of $\alpha = 0^\circ$, $2^\circ$, $4^\circ$, $6^\circ$ and $8^\circ$. The equations applied to the measured quantities to calculate the presented flow variables are detailed in Section 3.3. The results from the normalized flow field contours from the PIV for the tested angles of attack are shown through Figs. 4.5 to 4.10 and the results for the HF and MF airfoils are presented in the left and right columns, respectively and with the increasing angles of attack presented in different rows.

The normalized streamwise velocity contour ($U/U_\infty$) results over the flap and wake region presented in Fig. 4.5 show that at the angles of attack $\alpha = 0^\circ$ and $2^\circ$, the flow separates at the flap hinge location $x/c = 0.7$ for the HF airfoil but remains close to the wall, whereas for the MF airfoil the flow separates only at $x/c = 0.95$, consistent with the observations in surface flow visualization results in Fig. 4.4. The results for the MF airfoil show a region of negative velocity close to the trailing edge, indicating a separation bubble, which is absent in the case of the HF airfoil. The results for the angle of attack $\alpha = 4^\circ$ and $6^\circ$, in Figs. 4.5e, 4.5f, 4.5g, and 4.5h, show that the flow separates at the hinge point $x/c = 0.7$ for the HF airfoil, while in the MF airfoil case the flow completely separates only at $x/c > 0.8$. The results for the angle of attack $\alpha = 8^\circ$, show that the separation point for the HF airfoil remains unchanged but for the MF airfoil, the separation point moves upstream to $x/c = 0.85$. The separation bubble close to the trailing edge of the MF airfoil at $\alpha = 8^\circ$ is substantially larger compared to the HF airfoil.

The normalized crosswise velocity contour ($V/U_\infty$) results for the HF and MF airfoils are shown in Fig. 4.6 for all the tested angles of attack. At first glance, the results clearly show that the MF airfoil has increased negative velocity over the flap region relative to the HF airfoil, indicating increased downwash velocity. The separation over the flap region for both the HF
and MF airfoils are also seen quite evidently in the presented results. For the HF airfoil, even though the flow separates at the flap hinge location $x/c = 0.7$ for all the angles of attack, the flow remains close to the airfoil surface at low angles of attack $\alpha = 0^\circ$ and $2^\circ$, as seen in Figs. 4.6a and 4.6c. As the angle of attack is increased, the flow separation is much more distinct, as seen in Figs. 4.6e, 4.6g and 4.6i for angles of attack $\alpha = 4^\circ$, $6^\circ$ and $8^\circ$, respectively. The results of the normalized crosswise velocity contour show the flow separation at the vicinity of the trailing edge more prominently for the MF airfoil relative to the HF airfoil for all the presented angles of attack. The adverse flow velocity seen at the trailing edge vicinity for the MF airfoil at low angles of attack $\alpha = 0^\circ$ and $2^\circ$ (see Figs. 4.6b, 4.6d) is located at close proximity to the trailing edge $x/c > 0.8$ compared to the HF airfoil, for which, the increased negative velocity begins at the streamwise location $x/c = 0.7$. As the angle of attack is increased to $\alpha = 8^\circ$, the flow separation location at $x/c = 0.7$ can also be clearly seen. The results of the mean velocity contours presented in Figs. 4.5 and 4.6 has shown that flow separation and acceleration over the flap region for both the HF and MF airfoils very clearly. The separation regions seen here are also in agreement with surface flow visualization results and discussion in Section 4.2.2.

The results of the non-dimensional turbulent kinetic energy ($k/U_\infty^2$, TKE) contours over the flap and wake region of the HF and MF airfoils are presented in Figs. 4.7. The results for the HF and MF airfoil are presented in the left and right columns of the figures, respectively, with different angles of attack present in different rows. The same color scales are used for each angle of attack to facilitate comparison of the contour plots for the presented results. The boundary layer separation locations over the flap region, which were discussed earlier can be seen much more distinctively in the turbulent kinetic energy contours of both the airfoils at all the presented angles of attack. These results clearly indicate highly turbulent flow behavior in the flap boundary layer and in the near wake regions. The results of the TKE contours in Fig. 4.7 clearly show two distinct regions of high TKE in the near-wake region for all the presented angles of attack and configurations. toward the suction side of both the airfoil configurations, a high TKE region arises from the flow separation over the flap. toward the pressure side, another high TKE region arises from the interaction of the oncoming flow with the trailing edge. The results clearly show that the overall TKE at the near-wake region is higher for the MF airfoil compared to the HF airfoil
for all the presented angles of attack. At the low angles of attack, $\alpha = 0^\circ$ and $2^\circ$, the high TKE toward the suction and pressure side can be found at the near-wake regions but as the angle of attack is increased, $\alpha = 4^\circ$, $6^\circ$ and $8^\circ$ the TKE in the wake reduces toward the suction side region but increases toward the pressure side region. This is due to the direct interaction of the oncoming inflow with the trailing edge. The increased high TKE toward the pressure side at the near-wake location is higher for the MF airfoil compared to the HF airfoil.

The turbulent kinetic energy results primarily consist of the streamwise and crosswise Reynolds stress components. In order to identify the prime contributor to the increased TKE over the airfoil flap and the trailing edge, it is important to further visualize the streamwise and crosswise Reynolds stress components individually. The results of the non-dimensional streamwise normal Reynolds stress components ($u'u' / U_\infty^2$) in Fig. 4.8 clearly show two distinct regions of high $u'u'$ values found at the wake location for both the HF and MF airfoil at all the presented angles of attack. The flow aft of the trailing edge, at the further downstream location, the wake profile undergoes a fast relaxation process. The two distinct regions with high TKE at the vicinity of the trailing edge seen earlier are much more pronounced in the $u'u'$ results. At the low angles of attack, $\alpha = 0^\circ$ and $2^\circ$, the two distinct regions of high $u'u'$ are found on both the upper and lower side of the wake. The increased regions of $u'u'$ extend up to $x/c = 1 - 1.4$ into the wake region for the MF airfoil, whereas, for the HF airfoil it extends only up to $x/c = 1 - 1.2$ into the wake. The stresses toward the suction side of the wake decay quicker compared to the pressure side of the wake at the further downstream locations for both the airfoils. The MF airfoil has higher values of $u'u'$ on the upper side relative to the HF airfoil for the low angle of attack. At the higher angles of attack $\alpha = 4^\circ$, $6^\circ$ and $8^\circ$, the values of $u'u'$ are wider compared to the previously discussed lower angles of attack. The early separation at this angle of attack gives rise to higher values of $u'u'$ over the flap surfaces for both the airfoils compared to lower angles of attack. The lower side of the airfoils have higher values of $u'u'$ compared to the upper side at higher angles of attack. When considering only the increased $u'u'$ on the lower side the MF airfoil has higher values of $u'u'$ compared to the HF airfoil. These high values of $u'u'$ for the MF airfoil are due to the trailing edge tip interacting with the flow at a higher angle compared to the trailing edge tip of the HF airfoil.
CHAPTER 4. MORPHED TRAILING EDGES

The contours of the non-dimensional crosswise normal Reynolds stress component \( \overline{v'v'} / \overline{U_\infty^2} \) presented in Fig. 4.9 shows increased magnitude of \( \overline{v'v'} \) at regions closer to the trailing edge for the MF airfoil at the near-wake regions \( x/c = 1 - 1.3 \) for all the presented angles of attack. The HF airfoil does not show increased values of \( \overline{v'v'} \) at the trailing edge vicinity except at the angle of attack \( \alpha = 0^\circ \). The increased \( \overline{v'v'} \) close to the airfoil trailing edge, at the wake region, arises from the trailing edge tip location \( x/c = 1 \) for both the airfoil configurations. This is due to the inflow interacting with the airfoil's trailing edge. Since the angle at which the trailing edge tip interacts with the inflow is higher for the MF airfoil compared to the HF airfoil, the MF airfoil has higher \( \overline{v'v'} \) at the airfoil wake. These results with reduced \( \overline{v'v'} \) at the airfoil's wake clearly imply that the main contributor to the increased TKE is the streamwise normal Reynolds stress components \( \overline{u'u'} \). Moreover, the dominant characteristic of the TKE for both the airfoil at the wake is the two distinct regions with increased TKE values, which is predominantly contributed by the \( \overline{u'u'} \).

The non-dimensional Reynolds shear stress contours \( -\overline{u'v'}/U_\infty^2 \) for both the HF and MF airfoils are presented in Fig. 4.10. The Reynolds shear stress \( -\overline{u'v'} \) contours for the MF airfoil is wider, with higher values compared to the HF airfoil at the near-wake locations for low angles of attack \( \alpha = 0^\circ, 2^\circ \). The HF airfoil has high \( -\overline{u'v'} \) at the near-wake locations \( x/c = 1 - 1.3 \), whereas, for the MF airfoil the high values of \( -\overline{u'v'} \) extend up to the wake locations \( x/c = 1 - 1.4 \). However, the shear stresses at far-wake downstream locations are similar between the two airfoil configurations for the low angles of attack. The relatively high Reynolds stresses at the airfoil wake are produced due to high mean shear and eventually, the stresses reduce in the far-wake regions due to their diffusive and dissipative nature. At the increased angles of attack \( \alpha = 4^\circ, 6^\circ \) and \( 8^\circ \), the \( -\overline{u'v'} \) contours for the MF airfoil show much wider regions of high \( -\overline{u'v'} \) values compared to the HF airfoil at near- and far-wake regions. This is very evident at \( \alpha = 8^\circ \) in Figs. 4.10i and 4.10j at near- and far-wake regions. The high \( -\overline{u'v'} \) that arise from the trailing edge tip interaction with the inflow for the MF airfoil extends well into the far-wake locations. The results also quite evidently show the increased wake deflection angle for the MF airfoil compared to the HF airfoil.

74
The non-dimensional flow field contours presented in Figs. 4.5 to 4.10 clearly shows the delayed flow separation for the MF airfoil relative to the HF airfoil. These results of the delayed flow separation for the MF airfoil also correspond to the surface flow visualization results seen earlier in Section 4.2.2. The TKE results showed increased turbulence intensity with a wider wake for the MF airfoil at the vicinity of the trailing edge and in the near-wake region compared to the HF airfoil. The TKE results show two distinctive regions of high TKE at the trailing edge vicinity. The major contributor to the two distinct regions of high turbulence arises from the streamwise normal Reynolds stress tensors ($\overline{u'u'}$) components. The crosswise-normal Reynolds stress component ($\overline{v'v'}$) and the Reynolds shear stress component ($-\overline{u'v'}$) for the MF airfoil is higher compared to the HF airfoil.
Figure 4.5: The mean streamwise velocity contours from PIV for the Hinged Flap and Morphed Flap airfoils.
4.2. EXPERIMENTAL RESULTS

Figure 4.6: The mean crosswise velocity contours from PIV for the Hinged Flap and Morphed Flap airfoils.
Figure 4.7: The normalised turbulent kinetic energy contours for the Hinged Flap and Morphed Flap airfoils at various angles of attack for a free-stream velocity of $U_{\infty} = 20$ m/s ($Re_c = 2.6 \times 10^5$).
Figure 4.8: The normalised streamwise Reynolds normal stress contours for the Hinged Flap and Morphed Flap airfoils at various angles of attack for a free-stream velocity of $U_\infty = 20$ m/s ($Re_c = 2.6 \times 10^5$).
Figure 4.9: The normalised crosswise Reynolds normal stress contours for the Hinged Flap and Morphed Flap airfoils at various angles of attack for a free-stream velocity of $U_\infty = 20$ m/s ($Re_c = 2.6 \times 10^5$).
4.2. EXPERIMENTAL RESULTS

Figure 4.10: The normalised Reynolds shear stress contours for the Hinged Flap and Morphed Flap airfoils at various angles of attack for a free-stream velocity of $U_\infty = 20$ m/s ($Re_c = 2.6 \times 10^5$).
4.2.4 Wake development

The wake flow field of both the HF and MF airfoils were studied experimentally using the PIV technique. The tests were carried out for a free-stream velocity of \( U_{\infty} = 20 \text{ m/s} \) \( (Re_c = 2.6 \times 10^5) \) at angles of attack \( \alpha = 0^\circ, 2^\circ, 4^\circ, 6^\circ, \) and \( 8^\circ \). Detailed line data were extracted in the mid-span position of the airfoil from the contour plots of the PIV measurements in Section 4.2.3. The results are presented and investigated for six downstream locations (see Fig. 4.11) in the airfoil wake region, \( x/c = 1.025, 1.05, 1.10, 1.20, 1.40, \) and 1.60 with the airfoil leading-edge tip as the chord-wise datum point \( (x/c = 0) \). The line plots of the normalized mean velocity, turbulent kinetic energy, and Reynolds stress tensors are presented in Figs. 4.12, 4.13 and 4.14. For the purpose of brevity, the results are presented only for angles of attack \( \alpha = 0^\circ, 4^\circ \) and \( 8^\circ \).

Figure 4.11: Airfoil coordinate system along with the data extraction locations in the wake region.

The non-dimensional mean streamwise velocity profiles \((U/U_{\infty})\) for the HF and MF airfoil at angles of attack \( \alpha = 0^\circ, 4^\circ, \) and \( 8^\circ \) are presented in Figs. 4.12a, 4.13a, and 4.14a, respectively. The mean velocity profiles at the near-wake location \( x/c = 1.025 \) show an increased velocity deficit for the MF airfoil relative to the HF airfoil for all the tested angles of attack. The velocity deficit for the MF airfoil relative to the HF airfoil reduces at downstream locations \( x/c = 1.05 \) and 1.10. At the further downstream far-wake locations \( x/c = 1.20, 1.40 \) and 1.60, the velocity deficit between the HF and MF airfoils are similar. At the far-wake locations, the overall results for all the presented angles of attack show that the wake velocity for the MF airfoil has an increased flow deflection angle compared to the HF airfoil. The increased velocity deficit for the MF airfoil
relative to the HF airfoil is prominent only at all the downstream near-wake locations.

The non-dimensional turbulent kinetic energy profiles \((k/U_\infty^2, \text{TKE})\) results extracted from the PIV contours for the HF and MF airfoils at the angles of attack \(\alpha = 0^\circ, 4^\circ,\) and \(8^\circ\) are presented in Figs. 4.12b, 4.13b, and 4.14b, respectively. The results of the TKE magnitude clearly show the characteristic double peak wake TKE behavior at both the near- and far-wake location. The double peak behavior is much more prominent at the vicinity of the trailing edge due to the two eminent different boundary layers that arise from the upper and lower side of the airfoil. For all the tested angles of attack, at the near-wake location, \(x/c = 1.025\) and \(x/c = 1.05\) close to the trailing edge, a peak toward the lower surface with high TKE magnitude was observed for the MF airfoil. This is due to the high angle at which the MF airfoil’s trailing edge tip interacts with the free-stream inflow. The TKE magnitude for the HF airfoil is up to 50% lower than that of the MF airfoil for the peak \((y/c = 0)\) toward the lower surface at the streamwise locations \(x/c = 1.025\) and \(x/c = 1.05\). The TKE magnitude peak for the HF airfoils, at the locations \(x/c = 1.10\) and \(x/c = 1.20\), is also observed to be up to 20% lower compared to the MF airfoil. This difference in the TKE magnitude between the two airfoils reduces at the further far-wake downstream locations \(x/c = 1.40\) and \(1.60\). At the far-wake locations, \(x/c = 1.40\) and \(1.60\), the results show a wider wake toward the airfoil pressure side for the MF airfoil relative to the HF airfoil.

The non-dimensional streamwise Reynolds stress tensors \((\overline{u'u'}/U_\infty^2)\) for the HF and MF airfoil at the angles of attack \(\alpha = 0^\circ, 4^\circ,\) and \(8^\circ\) are presented in Figs. 4.12c, 4.13c, and 4.14c, respectively. The \(u'u'\) results are dominated by the characteristic double peak behavior at both the near- and far-wake locations as observed earlier in the TKE results. The MF airfoil shows higher \(u'u'\) compared to the HF airfoil at both the near- and far-wake locations. These differences between the airfoils are much more prominent at the near-wake locations \(x/c = 1.025, 1.05\) and \(1.10\).

The non-dimensional crosswise Reynolds stress tensors \((\overline{v'v'}/U_\infty^2)\) for the HF and MF airfoil at the angles of attack \(\alpha = 0^\circ, 4^\circ,\) and \(8^\circ\) are presented in Figs. 4.12d, 4.13d, and 4.14d, respectively. The \(v'v'\) results here also show the characteristic double peak behavior but only at the near-wake locations \(x/c = 1.025, 1.05\) and \(1.10\). Even though the two distinct peaks are observed they are not as prominent as the TKE and \(u'u'\) magnitude. At the far-wake locations downstream locations \(x/c = 1.20, 1.40\) and \(1.60\), the double peak is absent in the \(v'v'\) results. For all the near-
and far-wake locations the MF airfoil has increased $\overline{u'v'}$ magnitude compared to the HF airfoil irrespective of to which if it possesses the double peak behavior or not. The increased downwash angle for the MF airfoil relative to the HF airfoil can be readily observed at the fat-wake location $x/c = 1.60$ for all the tested angles of attack.

The Reynolds shear stress components ($-\overline{u'v'}/U_{\infty}^2$) for the HF and MF airfoil at the angles of attack $\alpha = 0^\circ$, $4^\circ$, and $8^\circ$ are presented in Figs. 4.12e, 4.13e, and 4.14e, respectively. The $-\overline{u'v'}$ magnitude is higher toward the pressure side ($y/c < 0$) of the MF airfoil relative to the HF airfoil at the near-wake locations $x/c = 1.025$, $x/c = 1.05$ and $x/c = 1.10$ for all the tested angles of attack. The $-\overline{u'v'}$ magnitude at the downstream far-wake locations are similar between the two cases for all the tested angles of attack but with the MF airfoil having a wider shear relative to the HF airfoil.

The aerodynamic characteristics studies seen in the previous Section 4.2.1 showed increased $C_L/C_D$ for the MF airfoil compared to the HF airfoil. The surface flow visualization results in Section 4.2.2 and the wake flow behavior in Section 4.2.3 shows us that the MF airfoil has delayed separation on the flap and increased wake deflection angle at the far-wake locations. The increased wake deflection angle for the MF airfoil aids in the improved lift performance seen before and the longer region of attached flow near the trailing edge of the MF airfoil aids in the reduction of form drag resulting in higher $C_L/C_D$ relative to the HF airfoil. This hypothesis is also supported by the flow behavior seen previously in the Gurney flap studies [106, 107].

The PIV flow visualization results in Section 4.2.3 and wake profile analysis in Section 4.2.4 gave an insight into the energy content within the airfoil’s wake flow. The results were presented for TKE, $\overline{u'u'}$, $\overline{v'v'}$, and $-\overline{u'v'}$. The TKE results showed characteristic double peak behavior for both the HF and MF airfoils at all the tested angles of attack. The results for the $\overline{u'u'}$ and $\overline{v'v'}$ magnitude showed that the energy for the characteristic double peak behavior was predominantly contributed by the $\overline{u'u'}$ and not the $\overline{v'v'}$ for both the HF and MF airfoils at all the tested angles of attack. The TKE, $\overline{u'u'}$ and the $-\overline{u'v'}$ magnitude is higher for the MF airfoil relative to the HF airfoil and toward the pressure side ($y/c < 0$) compared to the suction side ($y/c > 0$) and the HF airfoil. This is due to the angle of the trailing edge tip of the MF airfoil and its direct interaction with the free-stream inflow.
Figure 4.12: The non-dimensional mean velocity profiles (a), turbulence kinetic energy (b), and Reynolds stress tensor (c), (d) and (e) for the Hinged Flap — and Morphed Flap — airfoils at angles of attack $\alpha = 0^\circ$. 

85
Figure 4.13: The non-dimensional mean velocity profiles (a), turbulence kinetic energy (b), and Reynolds stress tensor (c), (d) and (e) for the Hinged Flap — and Morphed Flap airfoils at angles of attack $\alpha = 4^\circ$. 
Figure 4.14: The non-dimensional mean velocity profiles (a), turbulence kinetic energy (b), and Reynolds stress tensor (c), (d) and (e) for the Hinged Flap — and Morphed Flap - - - airfoils at angles of attack $\alpha = 8^\circ$. 

87
4.3 Computational Fluid Dynamics

In addition to the experimental measurements and the discussions presented in the preceding sections, detailed Large Eddy Simulation (LES) were also carried out for both the HF and MF airfoils at the selected angles of attack of $\alpha = 0^\circ$ and $4^\circ$ for free-stream velocity of $U_\infty = 20$ m/s, corresponding to a chord-based Reynolds number of $Re_c = 2.6 \times 10^5$. The LES investigation for the HF and MF airfoils are essential as to inspect the unsteady characteristics of the flow field over the flap and trailing edge. The results from the LES are thoroughly analyzed and presented in this section.

The airfoil geometries, domain and the computational setup for the LES are presented and the results are validated with the PIV measurements in Section 3.4. The pressure coefficient and the pressure coefficient root mean squared distribution around the airfoils are presented in Section 5.2.2. In Section 4.3.2 the results are presented for the wake flow development which includes, iso-contours of Q-criterion, instantaneous and means velocity, and Reynolds stress tensors. The boundary layer development is presented in Section 4.3.3 in order to further investigate the flow separation region over the flap. Further unsteady flow characteristics study was carried out by analyzing the wall pressure spectra, spanwise coherence, length scale, space-time correlation, wake velocity spectra, and far-field noise, which are presented in Sections 4.3.4 to 4.3.7, respectively. The following results will aid us to better understand the flow structures and aeroacoustic behavior of both the HF and MF airfoils.

The LES computational setup used in the current study was validated with available experimental data [79] for the baseline NACA 0012 airfoil at a chord based Reynolds number of $Re_c = 4 \times 10^5$. The results for which are already presented in a collaborative study with Cesar et al. [69]. The results of the pressure coefficient ($C_p$), boundary layer displacement and momentum thickness, mean boundary layer profiles, mean wake velocity profile, Reynolds stress components ($u' u'$ and $v' v'$), and wall pressure spectra are presented in Figs. 4.15 to 4.19. The presented LES results show very good agreement with the experimental data from Sagrado [79].
Figure 4.15: The validation from Cesar et al. [69] for (a) Pressure coefficient (LES —, Exp ○) (b) boundary layer momentum thickness (LES †, Exp □) and displacement thickness (LES ♂, Exp ○) [79].

Figure 4.16: The validation from [69] for the mean velocity profiles at various streamwise locations on the boundary layer of the NACA 0012 baseline airfoil for LES — and Exp ○ [79].

Figure 4.17: The validation from Cesar et al. [69] for the mean wake profile at the vicinity of the trailing-edge of the NACA 0012 baseline airfoil for LES — and Exp ○ [79].
Figure 4.18: The validation from Cesar et al. [69] for (a) Streamwise normal Reynolds stresses $u'u'$ and (b) crosswise normal Reynolds stresses $v'v'$ for LES $x/c = 0.80$ -- , $x/c = 0.90$ ----, $x/c = 0.98$ --- and Exp $x/c = 0.80$ ◻, $x/c = 0.90$ □, $x/c = 0.98$ ★.

Figure 4.19: The validation from Cesar et al. [69] for the wall-pressure power spectral density with $p_{Ref} = 2 \times 10^5$ Pa at various streamwise locations. (a) LES $(x/c = 0.80)$ -- , $(x/c = 0.90)$ ----, and $(x/c = 0.98)$ ---, (b) Experiments $(x/c = 0.80)$ black cross, $(x/c = 0.90)$ red circle, and $(x/c = 0.98)$ blue asterisk.
In order to further show the validity of the LES results, the non-dimensional mean velocity profiles at the wake region for the HF and MF airfoils from the LES are compared with the PIV measurements from Section 4.2.4 and the results are presented in Fig. 4.20. The results from the dynamic Smagorinsky model accurately predicts the wake velocity deficit and the dip locations for both the angles of attack ($\alpha = 0^\circ$ and $4^\circ$). The velocity deficit magnitude is found to be slightly under-predicted by the simulation at the near-wake locations $x/c = 1.025$, $1.05$, and $1.10$ compared to its respective experimental results. The velocity profile comparison also shows that the LES simulations have captured the wake width accurately at all the presented wake locations.
4.3.1 Pressure distribution

In the previous Section 4.2.1, the improved aerodynamic capabilities of the MF airfoil compared to the HF airfoil was observed. It is essential to further investigate the steady and unsteady pressure coefficient distribution around the two airfoils to identify the source of improved aerodynamic behavior. The results of the steady surface pressure coefficient \( (C_p) \) distribution around both the HF and MF airfoils at angles of attack \( \alpha = 0^\circ \) and \( 4^\circ \) are presented in Figs. 4.21a and 4.21b, respectively. The \( C_p \) is calculated using the Equation 3.3 in Section 3.2.2. A suction peak in the \( C_p \) distribution can be observed at the vicinity of the leading edge of the airfoil for both the airfoil configurations. A sudden secondary peak in the \( C_p \) distribution at the step-trip location for both the HF and MF airfoils on the suction and pressure sides can be distinctly seen in the presented results. This is due to the flow acceleration over the step trip. The effect of the trip quickly fades away with the \( C_p \) distribution for both the airfoils reconciling with the normal trend just after \( x/c < 0.25 \).

The suction peak of the MF airfoil at the angle of attack \( \alpha = 0^\circ \) is 11% higher relative to the HF airfoil while at \( \alpha = 4^\circ \), the suction peak of the MF airfoil is 8% higher than that of the HF airfoil. The results show that the change in the mean camber of the flap profile of length \( b = 0.3c \) at the trailing edge has an effect on the suction peak close to the airfoil leading edge. The \( C_p \) distribution over the flap at regions \( x/c = 0.7 − 1.0 \) for the HF and MF airfoil show significant differences between the two airfoil configurations. A third small peak in the pressure distribution on the suction side of the HF airfoil can be seen at \( x/c = 0.7 \), which was also observed in Sections 4.2.2 and 4.2.3. This arises due to the impingement of the flow on to the small bulge from the hinge point of the HF airfoil and can be seen for both the presented angles of attack. This third peak on the suction side is absent for the MF airfoil due to the smooth curvature of the flap profile. The results clearly show that the MF airfoil experiences a larger pressure difference over the entire flap region compared to the HF airfoil. This increased suction peak on the leading edge and the increased pressure difference over the flap surfaces for the MF airfoil compared to the HF airfoil corresponds with the increased \( C_L \) for the MF airfoil previously seen in Section 4.2.1.
4.3. COMPUTATIONAL FLUID DYNAMICS

The coefficient of pressure root-mean-squared ($C_{p\text{RMS}}$) over the airfoil surface is presented in Figs. 4.21c and 4.21d, for the angles of attack $\alpha = 0^\circ$ and $4^\circ$, respectively. The unsteady pressure distribution over the flap surface can also provide some comparative insight into the trailing edge noise generation mechanisms of these airfoils. All the peaks in the results seen close to the trailing edge arise on the suction side of the airfoil. The results on the suction side of the HF airfoil at the hinge point location ($x/c = 0.7$) show increased surface pressure fluctuations but the pressure fluctuations subside after chord location $x/c \approx 0.85$. For the MF airfoil, highly unsteady surface pressure can be seen within the chord-wise regions of $x/c = 0.9 - 1.0$, close to the trailing edge, for both the angles of attack.

The contours of the $C_{p\text{RMS}}$ over a slice on the mid-span of the LES computational domain for both the HF and MF airfoils at the angles of attack $\alpha = 0^\circ$ and $4^\circ$ are presented in Fig. 4.22. The results for the HF airfoil show increased pressure fluctuations just after the flap hinge point for both the presented angles of attack. For the MF airfoil, two regions of increased pressure fluctuations are observed close to the trailing edge. There is a large region of increased fluctuation toward the suction side extending from $x/c = 0.9 - 1.2$ and another small region of increased fluctuation closer to the trailing edge point extending from $x/c = 1 - 1.2$. The regions of increased pressure fluctuations close to the trailing edge for the MF airfoil extends well into the wake region up to $x/c \approx 1.3$ for both the presented angles of attack. For the HF airfoil, the increased pressure fluctuation in the wake region extends no further than $x/c = 1$. The overall results of the $C_{p\text{RMS}}$ over the flat and at the vicinity of the trailing edge show increased pressure fluctuations with higher intensity for the MF airfoil compared to the HF airfoil. The $C_p$ and $C_{p\text{RMS}}$ results presented here clearly show us that the geometric profile of the flap has a strong influence on the flow at the vicinity of the trailing edge. The higher pressure fluctuations closer to the trailing edge of the MF airfoil would potentially lead to unfavorable higher noise production compared to the HF airfoil. This will be further discussed in detail in Sections 4.3.4 to 4.3.7.
Figure 4.21: Pressure coefficient and pressure coefficient root mean squared over the airfoil surface for Hinged Flap and Morphed Flap airfoils at angles of attack $\alpha = 0^\circ$ and $4^\circ$ at free-stream velocity of $U_\infty = 20$ m/s ($Re_c = 2.6 \times 10^5$).

Figure 4.22: Pressure coefficient contours root mean squared for the Hinged Flap and Morphed Flap airfoils at angles of attack $\alpha = 0^\circ$ and $4^\circ$ at free stream velocity of $U_\infty = 20$ m/s ($Re_c = 2.6 \times 10^5$).
4.3. COMPUTATIONAL FLUID DYNAMICS

4.3.2 Wake flow development

In this section the instantaneous and mean wake flow fields are investigated in order to further understand the wake behavior of the HF and MF airfoil and also to gain more confidence in the presented LES results. A well established standard method commonly used to identify the turbulent coherent structures formed around an airfoil is the $Q$-criterion visualization, where $Q$ is the second invariant of the velocity gradient tensor, which is defined by,

$$Q = \frac{1}{2}(\Omega_{ij}\Omega_{ij} - S_{ij}S_{ij}), \quad (4.1)$$

where the $\Omega_{ij}$ and $S_{ij}$ are the anti-symmetric and symmetric part of the velocity gradient, respectively, that is:

$$\Omega_{ij} = \frac{1}{2} \left( \frac{\partial u_i}{\partial x_j} - \frac{\partial u_j}{\partial x_i} \right), \quad (4.2)$$

$$S_{ij} = \frac{1}{2} \left( \frac{\partial u_i}{\partial x_j} + \frac{\partial u_j}{\partial x_i} \right). \quad (4.3)$$

The $Q$-criterion thus represents the balance between the rate of vorticity $\Omega^2 = \Omega_{ij}\Omega_{ij}$ and the rate of strain $S^2 = S_{ij}S_{ij}$. At the core of a vortex, $Q > 0$, as vorticity increases as the center of the vortex is approached. Thus, regions of positive $Q$-criterion correspond to vortical structures. This type of visualization allows the identification of rotational motion from non-rotational motions. The flow field over the airfoils and in the wake region are visualized using iso-surfaces of $Q$-Criterion and are presented in Fig. 4.23 for the HF and MF airfoil at angles of attack $\alpha = 0^\circ$ and $4^\circ$. The results are presented with contours of vorticity magnitude and iso-surfaces of $Q$-criterion of $Q = 1 \times 10^6 s^{-2}$ for both the airfoils at the angles of attack $\alpha = 0^\circ$ and $4^\circ$. The results show that the simple step-trip $(x/c = 0.1)$ was sufficient to make the flow turbulent over the airfoil and prevent the formation of 2D spanwise rollers over the airfoil that was observed in the author’s previous studies [66]. The green-colored iso-surfaces of $Q$-criterion indicate flow separated regions. The separated regions can be seen right behind the step-trip $x/c = 0.1$ and also right after the flap hinge point $x/c = 0.7$ for the HF airfoil and close to the trailing edge, $x/c > 0.8$, for the MF airfoil.
CHAPTER 4. MORPHED TRAILING EDGES

Figure 4.23: Iso-surfaces of $Q$-criterion of $Q = 1 \times 10^6 s^{-2}$ for Hinged Flap and Morphed Flap airfoil with contours of vorticity magnitude at angles of attack $\alpha = 0^\circ$ and $4^\circ$ for a free-stream velocity of $U_\infty = 20$ m/s ($Re_c = 2.6 \times 10^5$).

Figure 4.24: The normalised instantaneous streamwise velocity contours from the LES flow for the Hinged Flap and Morphed Flap airfoils at angles of attack $\alpha = 0^\circ$ and $4^\circ$ for a free-stream velocity of $U_\infty = 20$ m/s ($Re_c = 2.6 \times 10^5$).
4.3. COMPUTATIONAL FLUID DYNAMICS

Figure 4.25: The normalised mean streamwise velocity contours for the LES flow field at mid span for the Hinged Flap and Morphed Flap airfoils at angles of attack $\alpha = 0^\circ$ and $4^\circ$ for a free-stream velocity of $U_\infty = 20$ m/s ($Re_c = 2.6 \times 10^5$).

The results of the non-dimensional instantaneous ($u'/U_\infty$) and mean ($U/U_\infty$) streamwise velocity contours extracted along the mid-span location ($z/c = 0$) of the LES domain for both the HF and MF airfoil at the angles of attack $\alpha = 0^\circ$ and $4^\circ$ are presented in Figs. 4.24 and 4.25. The results are presented in terms of the normalized chord distance ($x/c$) and normalized crosswise distance ($y/c$) and the contours show the non-dimensional instantaneous and mean streamwise velocity ($u'/U_\infty$ and $U/U_\infty$). The instantaneous velocity contours for the MF airfoil at chord-wise location $x/c = 0.8 - 1$ shows a higher level of unsteadiness on the pressure side of the flap compared to the HF airfoil. The flow unsteadiness over the airfoils is quite evident, which shows that the step trip placed at $x/c = 0.1$ was sufficient enough to induce turbulence within the boundary layer on both the suction and pressure surfaces.
The non-dimensional mean streamwise velocity contour plots from the LES simulations are very similar to the PIV results seen previously in Section 4.2.3. The flow separation point for the HF airfoil is right after the flap hinge location ($x/c = 0.7$) for both the angles of attack, whereas for the MF airfoil the flow separates very close to the trailing edge at only about $x/c = 0.9$. It can also be observed that flow separation point over the MF airfoil flap at both the angles of attack $\alpha = 0^\circ$ and $4^\circ$ is at the location $x/c = 0.9$, whereas in the experimental results in Sections 4.2.2 and 4.2.3 the separation point for the MF airfoil at angles of attack $\alpha = 0^\circ$ and $4^\circ$ are at $x/c = 0.95$ and $x/c = 0.8$, respectively.

![Figure 4.26](image)

Figure 4.26: The normalised mean crosswise velocity contours for the LES flow field at mid span for the Hinged Flap and Morphed Flap airfoils at angles of attack $\alpha = 0^\circ$ and $4^\circ$ for a free-stream velocity of $U_\infty = 20 \text{ m/s (Re}_c = 2.6 \times 10^5)$. 

98
4.3. COMPUTATIONAL FLUID DYNAMICS

The non-dimensional mean crosswise velocity \( \left( \frac{V}{U_\infty} \right) \) from the LES simulations extracted at the mid-span location \( (z/c = 0) \), for both the HF and MF airfoils at the angles of attack \( \alpha = 0^\circ \) and \( 4^\circ \) are presented in Fig. 4.26. The results are presented in terms of the normalized streamwise and crosswise coordinates \( (x/c \) and \( y/c) \). The results are presented with the same scale and color map for all the cases to facilitate the comparison of the contour plots. At first glance, the overall results show increased negative crosswise velocity on the suction side of the flap for all the presented cases. The results for the angles of attack \( \alpha = 0^\circ \) show that the region of increased negative crosswise velocity is located at slightly different streamwise locations for both the HF and MF airfoil. For the HF airfoil, the increased velocity region is present close to the flap hinge \( (x/c = 0.7 - 0.85) \), whereas, for the MF airfoil the increased negative crosswise velocity region is located closer to the trailing edge \( (x/c = 0.9 - 1) \). These regions of increased negative crosswise velocity show the region at which the flow starts to deflect downward. From the results at \( \alpha = 0^\circ \) in Figs. 4.26a and 4.26b, it can be seen that in the case of HF airfoil the flow starts to deflect at location \( x/c = 0.7 \), whereas in the case of the MF airfoil the flow starts to deflect only at chord-wise location \( x/c = 0.9 \) but with higher intensity than the HF airfoil.

At the angle of attack \( \alpha = 4^\circ \), it is evident from the results that the region with increased negative crosswise velocity covers the entire flap region for both the HF and MF airfoils. The results also show negative crosswise velocity with higher intensity over a larger chord-wise region compared to that of the results at the lower angle of attack \( \alpha = 0^\circ \). This larger chord-wise region with higher flow intensity is present at chord-wise regions \( x/c = 0.7 - 0.95 \) for the HF airfoil but for the MF airfoil this region is located at a downstream chord-wise region \( x/c = 0.8 - 1.1 \) as seen in Figs. 4.26c and 4.26d. The results presented here show that there is an increased downwash velocity for the MF airfoil especially due to the increased negative crosswise velocity region present closer to the trailing edge. The delayed flow separation for the MF airfoil relative to the HF airfoil is also very evident from the crosswise velocity.
CHAPTER 4. MORPHED TRAILING EDGES

Figure 4.27: The non-dimensional turbulent kinetic energy contours for the LES flow field at mid span for the Hinged Flap and Morphed Flap airfoils at angles of attack $\alpha = 0^\circ$ and $4^\circ$ for a free-stream velocity of $U_\infty = 20$ m/s ($Re_c = 2.6 \times 10^5$).

Figure 4.28: The non-dimensional streamwise Reynolds normal stress contours for the LES flow field at mid span for the Hinged Flap and Morphed Flap airfoils at angles of attack $\alpha = 0^\circ$ and $4^\circ$ for a free-stream velocity of $U_\infty = 20$ m/s ($Re_c = 2.6 \times 10^5$).
4.3. COMPUTATIONAL FLUID DYNAMICS

Figure 4.29: The non-dimensional crosswise Reynolds normal stress contours for the LES flow field at mid span for the Hinged Flap and Morphed Flap airfoils at angles of attack $\alpha = 0^\circ$ and $4^\circ$ for a free-stream velocity of $U_\infty = 20$ m/s ($Re_c = 2.6 \times 10^5$).

Figure 4.30: The non-dimensional Reynolds shear stress contours for the LES flow field at mid span for the Hinged Flap and Morphed Flap airfoils at angles of attack $\alpha = 0^\circ$ and $4^\circ$ for a free-stream velocity of $U_\infty = 20$ m/s ($Re_c = 2.6 \times 10^5$).
The flow characteristics over the airfoil flap surface and within the wake region can be further investigated by visualizing the turbulent kinetic energy and Reynolds stresses of the flow field at the flap and wake regions. The LES flow field at the mid-span location of the simulation domain is extracted to further visualize the mentioned TKE and Reynolds stresses. The non-dimensional turbulent kinetic energy \( (k/U_\infty^2) \) for both the HF and MF airfoil configurations at the angles of attack \( \alpha = 0^\circ \) and \( 4^\circ \) are presented in Fig. 4.27. The results for both the tested angles of attack \( \alpha = 0^\circ \) and \( 4^\circ \), clearly show increased TKE for the HF airfoil immediately after the flap hinge point \( (x/c > 0.75) \), while for the MF airfoil the increased TKE is observed only close to the trailing edge \( (x/c > 0.9) \). This believed to be is due to the delayed flow separation for the MF airfoil compared to the HF airfoil as previously observed in Section 5.2.2. The prominent difference between the \( \alpha = 0^\circ \) and \( 4^\circ \) is that the \( \alpha = 4^\circ \) has thicker boundary layer region with high TKE after the flow separation point for both the HF and MF airfoils. The two distinct regions with high TKE in the near-wake region seen earlier in the PIV measurements (see Fig. 4.7) are also observed here. For the MF airfoil in the PIV measurements the high TKE can be seen up to wake location \( x/c = 1.4 \) but for the LES the TKE dissipates much quicker by the wake location \( x/c = 1.2 \) for both the angles of attack \( \alpha = 0^\circ \) and \( 4^\circ \).

The results of the non-dimensional streamwise normal Reynolds stress components \( (u'\bar{u}'/U_\infty^2) \) obtained from the LES flow field are presented in Fig. 4.28. The results clearly show two distinct regions of high \( u'\bar{u}' \) at the vicinity of the trailing edge, developing into the wake flow field for both the HF and MF airfoil at the angles of attack \( \alpha = 0^\circ \) and \( 4^\circ \). The wake profile undergoes a fast relaxation process just as discussed before in the PIV results in Section 4.2.3 for all the presented LES cases. For both the airfoil configurations at the angles of attack \( \alpha = 0^\circ \) and \( 4^\circ \), the regions of high \( u'\bar{u}' \) values are found on both the suction and pressure sides of the airfoil’s trailing edge in the near wake region. The MF airfoil has higher values with larger regions of \( u'\bar{u}' \) on the suction side of the flap relative to the HF airfoil for both the presented angles of attack. The pressure side of the MF airfoil has the highest values of \( u'\bar{u}' \) compared to the pressure side of the HF airfoil for both angles of attack. These high \( u'\bar{u}' \) for the MF airfoil are due to the trailing edge tip interacting with the flow at a higher angle compared to the trailing edge tip of the HF airfoil, which is more oriented with the free-stream direction.
The contours of the non-dimensional crosswise normal Reynolds stress component $\frac{\bar{v}'v'k}{U_\infty^2}$ are presented in Fig. 4.29. The results show an increased magnitude of $\bar{v}'v'$ at regions closer to the trailing edge. The MF airfoil produces higher values of $\bar{v}'v'$ closer to the trailing edge relative to the HF airfoil for both the presented angles of attack $\alpha = 0^\circ$ and $4^\circ$. But the HF airfoil has a larger region of increased $\bar{v}'v'$ over the flap chord compared to the MF airfoil. At the wake region the two distinct regions seen earlier in the TKE and $\bar{u}'u'$ values are not seen here. This shows that those two distinct regions with increased values of TKE and $\bar{u}'u'$ are dominated by the streamwise velocity. The $\bar{v}'v'$ values in the wake region extend for a longer region in the streamwise direction further into the far-wake region for the HF airfoil at the angle of attack $\alpha = 4^\circ$ compared to the MF airfoil. The HF airfoil also possesses a much narrower and increased values of $\bar{v}'v'$ relative to the MF airfoil even at the further downstream far-wake region at locations $x/c = 1.4$ (see Figs. 4.29c and 4.29d).

The non-dimensional Reynolds shear stress contours ($-\frac{\bar{u}'v'k}{U_\infty^2}$) for both the HF and MF airfoils are presented in Fig. 4.30. The non-dimensional Reynolds shear stresses $-\bar{u}'v'$ for the MF airfoil is larger than that of the HF airfoil at the near-wake locations for both the presented angles of attack. However, the relatively high Reynolds stresses at far-wake downstream locations diffuses and dissipates much earlier for the MF airfoil compared to the HF airfoil for the presented angles of attack. The results of the TKE and the Reynolds stress components obtained from the LES simulations follow the same trend as the PIV measurement results presented and discussed in Section 4.2.3.
4.3.3 Boundary layer measurements

The flow separation over the flap surfaces for both the HF and MF airfoil can be further studied by analysing the non-dimensional streamwise boundary layer velocity \((U/U_\infty)\) profiles at different streamwise locations \((x/c = 0.65, 0.75, 0.85, 0.90, 0.95\) and \(0.99\)) over the flap surface. The results of the boundary layer velocity profiles on the suction and pressure sides of the flaps from the LES simulations are presented for the angles of attack \(\alpha = 0^\circ\) and \(4^\circ\) in Figs. 4.31 and 4.32, respectively. The results are presented in terms of non-dimensional velocity \((U/U_\infty)\) and the non-dimensional wall distance \((y/c)\).

For the angles of attack \(\alpha = 0^\circ\) and \(4^\circ\), at the streamwise location \(x/c = 0.65\), just before the flap hinge point, the two airfoils exhibit very similar flow behavior on both the suction and pressure surface. At the streamwise location, \(x/c = 0.75\), the flow appears to be still attached to the surface for both the HF and MF airfoils with the HF airfoil having increased velocity deficit. At the further downstream locations, \(x/c = 0.85\) and \(0.95\), on the suction surface a sudden decrease in the velocity gradient \((\delta U/\delta y)\) for the HF airfoil shows the onset of the boundary layer separation. For the MF airfoil, the boundary layer separation can be seen only near the trailing edge locations \(x/c = 0.95\) and \(0.99\). These results correspond to the flow separation phenomenon previously seen in the surface flow visualization results and PIV measurements in Figs. 4.4 and 4.5. The same trend of delayed separation for the MF airfoil compared to the HF airfoil was also observed in the pressure measurements in Figs. 4.21 and 4.22, where the increased fluctuations denoted the flow separation points. The boundary layer velocity profile on the pressure side for both airfoils show increased velocity deficit compared to the suction side at both the angles of attack \(\alpha = 0^\circ\) and \(4^\circ\). The streamwise boundary velocity profiles on the pressure side for the MF airfoil at both the angles of attack \(\alpha = 0^\circ\) and \(4^\circ\) show increased velocity deficit at location \(x/c = 0.85\) and \(0.90\) compared to the other streamwise locations due to the partially separated flow on the pressure side within the flap curvature. This velocity deficit on the pressure side of the flap is no observed for HF airfoil.
4.3. COMPUTATIONAL FLUID DYNAMICS

Figure 4.31: Boundary layer velocity profiles on the suction and pressure side at various streamwise locations of the Hinged Flap and Morphed Flap airfoils at angle of attack $\alpha = 0^\circ$ for a free-stream velocity of $U_\infty = 20$ m/s ($Re_c = 2.6 \times 10^5$).
Figure 4.32: Boundary layer velocity profiles on the suction and pressure side at various streamwise locations of the Hinged Flap and Morphed Flap airfoils at angle of attack $\alpha = 4^\circ$, for a free-stream velocity of $U_\infty = 20 \text{ m/s}$ ($Re_c = 2.6 \times 10^5$).
4.3. COMPUTATIONAL FLUID DYNAMICS

4.3.4 Wall pressure spectra

The velocity flow field and the turbulence over the flap surface were investigated in the previous sections. In this section, the strong near-field pressure fluctuation created by the turbulence will be investigated in detail. Amiet [111] has shown that the noise propagating to the far-field scattered at the trailing edge are directly proportional to the product between the power spectral of the surface pressure fluctuation and the spanwise extent of the turbulence length scales. Therefore it is important to thoroughly investigate the power spectral of the near-field surface pressure fluctuations, spanwise coherence and length scales.

The power spectral density $\Phi_{pp}$ of the surface pressure normalized by the reference pressure ($p_{Ref} = 2 \times 10^{-5} \text{ Pa}$) on the suction side of the HF and MF airfoils at the angles of attack $\alpha = 0^\circ$ and $4^\circ$ are presented in Figs. 4.33 and 4.34. The Welch power spectral density (PSD) of the pressure fluctuations were performed based on the time-domain LES pressure signals using Hamming windowing for segments of equal length with 50% overlap. The results presented in Fig. 4.33 show the evolution of the wall-pressure spectral density along the airfoil chord region $x/c = 0.6 - 1$ at the frequency range $10^2 \text{ Hz} < f < 10^4 \text{ Hz}$. Two important streamwise locations, $x/c = 0.75$ (....) right after the flap hinge and $x/c = 0.95$ (-----) close to the trailing edge are highlighted in Fig. 4.33 and detailed comparison of the results at these locations are presented in Fig. 4.34. The wall-pressure spectra at the selected locations are important for the better understanding of the noise generation mechanism from the trailing edge area.

The evolution of wall-pressure spectra in Fig. 4.33 clearly shows that for both angles of attack, just downstream of the flap hinge at location $x/c = 0.75$, a sharp increase in the wall-pressure spectra for the HF airfoil is observed. For the MF airfoil, the wall-pressure spectra increase only at the regions close to the trailing edge. The results in Fig. 4.34 for angles of attack $\alpha = 0^\circ$ and $4^\circ$, at $x/c = 0.75$ show that the spectra are up to 20 dB higher for the HF airfoil over the entire presented frequency range (100 Hz to 10 kHz) relative to the MF airfoil. This corresponds with the increased $C_{p, RMS}$ seen at the location $x/c = 0.75$ in Fig. 4.21. At the further downstream locations close to the trailing edge $x/c = 0.95$, the MF airfoil has up to 10 dB higher spectral levels at low to medium frequency range ($f \leq 2 \text{ kHz}$) compared to the HF airfoil. This is also consistent with the higher unsteady surface pressure and separated flow close to the trailing edge seen.
earlier in the boundary layer results (Fig. 4.31), $C_{p, RMS}$ results (Fig. 4.21) and the surface flow visualization results (Fig. 4.4).

Figure 4.33: Evolution of the wall-pressure spectra along the airfoil chord on the suction side with the beginning of the flap $x = 0.70c$ indicated by — and downstream locations $x = 0.75c$ and $x = 0.95c$ indicated by .... and - - -, respectively.

The spanwise coherence function $\gamma_{pp_{i,j}}^2$ based on the fluctuating surface pressure are calculated using Eq. 4.4,

$$\gamma_{pp_{i,j}}^2(f, \Delta z) = \frac{|\Phi_{pp_{i,j}}(f)|^2}{\Phi_{pp_{i,i}}(f)\Phi_{pp_{j,j}}(f)} \text{ for } i = 1 \text{ and } j = 1, 2, 3, ..., N,$$

(4.4)

where the symbol $|\cdot|$ denotes the absolute value, $N$ is the number of transducers along the span of the airfoil and $\Phi_{pp_{i,j}}$ is the cross-spectral density between two pressure signals $p_i$ and $p_j$.

The iso-coherence contours as the function of frequency ($f$) and spanwise separation distance ($z/c$) for the suction side of the airfoil are presented in Fig. 4.35. The results clearly show that
Figure 4.34: Wall-pressure spectra normalised by the reference pressure $p_{Ref} = 2 \times 10^{-5}$ Pa, on the suction side at different streamwise locations for Hinged Flap and Morphed Flap airfoils at angles of attack $\alpha = 0^\circ$ and $4^\circ$.

The use of an MF airfoil for both the angles of attack $\alpha = 0^\circ$ and $4^\circ$ at the location $x/c = 0.95$ has increased the coherence of the flow structures for frequencies below 3 kHz relative to the HF airfoil. The surface pressure coherence in the spanwise direction on the suction side at the location $x/c = 0.95$ for different Helmholtz numbers ($kc = 2\pi f \cdot c/343.2$) is presented in Fig. 4.36. The coherence drops considerably within location $z/c = 0.015$ for all frequencies except for the MF airfoil at $kc = 1$ ($f = 273$ Hz) for both the presented angles of attack. These results also confirm that the present span length of the LES domain ($z = 0.1c$) is sufficient enough for computing the airfoil noise over a wide frequency range.
Figure 4.35: Spanwise coherence of the surface pressure on the suction side, at the location $x/c = 0.95$ for Hinged Flap and Morphed Flap airfoils at angles of attack $\alpha = 0^\circ$ and $4^\circ$. 
4.3. COMPUTATIONAL FLUID DYNAMICS

Figure 4.36: Spanwise coherence of the surface pressure on the suction side, at the location \( x/c = 0.95 \) for Hinged Flap (solid line) and Morphed Flap (dashed line) airfoils at angles of attack \( \alpha = 0^\circ \) and \( 4^\circ \) for \( k_c = 1 \) (circles) \( k_c = 10 \) (asterisk) and \( k_c = 20 \) (triangle).

Figure 4.37: Spanwise coherence length scales for the surface pressure on the suction side at the location \( x/c = 0.95 \) for Hinged Flap and Morphed Flap airfoils at angles of attack \( \alpha = 0^\circ \) and \( 4^\circ \).

In order to estimate the noise generated through the interaction of advecting pressure over the airfoil surface, the spanwise correlation length of the wall pressure fluctuations were calculated at the vicinity of the trailing edge location \( x/c = 0.95 \). The spanwise correlation length of the flow structures can be calculated using the following equation,

\[
\Lambda_y(f) = \int_0^\infty \gamma_{p_i p_j}(f, \Delta z)d\Delta z.
\]  

(4.5)

The length scale \( \Lambda_y \) as a function of frequency is calculated using the spanwise coherence
results ($\gamma$) between the surface pressure signals $p_i$ and $p_j$ and are presented in Fig. 4.37. The normalized length scale ($\Lambda_{\gamma}/c$) results are presented as a function of frequency. At the angle of attack $\alpha = 0^\circ$ the MF airfoil shows larger integral length scales relative to the HF airfoil with values varying from $\Lambda_{\gamma}/c = 0.012$ to $0.032$ at low frequencies of $f = 250 – 700$ Hz and the largest difference in length scales of $\Delta \Lambda_{\gamma}/c = 0.02$ at $f = 300$ Hz. At angle of attack $\alpha = 4^\circ$ the MF airfoil yet again shows larger integral length scales varying from $\Lambda_{\gamma}/c = 0.012$ to $0.032$ compared to the HF airfoil at a frequency range of $f = 250 – 1000$ Hz with a average difference in length scales of $\Delta \Lambda_{\gamma}/c = 0.005$. The results clearly show that the MF airfoil has larger coherent structures relative to the HF airfoil.
4.3.5 Space-time correlation

In this section auto-correlation and cross-correlation of the unsteady surface pressure signals at the various chord-wise location will be investigated to better understand the spatial features of the turbulence structure. The space-time correlation provides information regarding the convection characteristic of the turbulence structures generated by the surface pressure fluctuations. The space-time correlation of the flow over the Hinged and morphed airfoils was acquired from the cross-correlation of the unsteady surface pressure signals, defined as,

$$R_{p_i p_j} (\tau) = \frac{p_i(t+\tau)p_j(t)}{p_{iRMS}p_{jRMS}}$$

for \(i = 1\) and \(j = 1, 2, 3, ..., N\), \(4.6\)

where \(p_i\) is the reference surface pressure signal and \(p_j\) is the upstream/downstream surface pressure signals at various location along the chord, \(p_{iRMS}\) and \(p_{jRMS}\) are the reference and upstream/downstream surface pressure root mean squared, \(\tau\) is the time-delay and the time-average is represented by the overbar. The results for the space-time correlation \(R_{p_i p_j}\) are presented for the flow moving upstream in Fig. 4.38 and the flow moving downstream in Fig. 4.39, as a function of the normalised time-delay \(\tau U_\infty / c\). The results are presented for a measurement probe separation distance of \(\Delta x/c = 0.02\) along the chord-wise locations of \(x/c = 0.72\) to 0.99. For the results presented in Figs. 4.38 and 4.39, the reference probe \((p_i)\) is always the probe at the first mentioned chord location in the legend, i.e. \(x/c = 0.99\) for Figs. 4.38, \(x/c = 0.72\) for Figs. 4.39a&b, \(x/c = 0.80\) for Figs. 4.39c&d, and \(x/c = 0.90\) for Figs. 4.39e&f. The black colored line in every plot represents the auto-correlation function of the respective reference probe at the respective probe location along the chord and shown in the respective legend.

The results presented in Fig. 4.38 shows the auto and cross-correlation function of the surface pressure fluctuation at the trailing edge location \(x/c = 0.99\) of the HF and MF airfoils up to the upstream location \(x/c = 0.90\) for both the angles of attack \(\alpha = 0^\circ\) and \(4^\circ\). The results for the angle of attack \(\alpha = 0^\circ\) and \(4^\circ\) are presented in the left and right side of the plots respectively. The cross-correlation \(R_{p_i p_j}\) for all the presented cases show negative \(\tau c/U_\infty\) since the \(R_{p_i p_j}\) is calculated for the flow upstream of the trailing edge. The auto-correlation function shows quick decay for the HF airfoil compared to the MF airfoil at both the presented angles of attack. This shows that the MF airfoil has a larger pressure generating structure compared to the HF airfoil at
the chord-wise location $x/c = 0.99$. At the angle of attack $\alpha = 0^\circ$, cross-correlation of the pressure signals can be seen for the HF airfoil only at the locations $x/c = 0.99$ and 0.98, and not any other upstream locations. For the MF airfoil at $\alpha = 0^\circ$, the cross-correlation of the pressure signals $R_{p_i,p_j}$ can be for location $x/c = 0.99$, 0.98, and 0.96. This confirms the larger pressure generated structures for the MF airfoil compared to the HF airfoil. At the angle of attack $\alpha = 4^\circ$, the results do not show any cross-correlation of the pressure signals between any of the locations.

The downstream movement of the pressure signals can also be investigated to see the size and speed of the turbulent flow structure over the airfoil. The fluctuating surface pressure was investigated at three regions over both the airfoils separately. The first region is at chord-wise location $x/c = 0.72 – 0.78$, which is from a location just before the flap hinge point to a location after the flap hinge. The second region is at chord-wise location $x/c = 0.80 – 0.88$, which is from a location after the flap hinge point to a location just before the trailing edge. The third region is at the vicinity of the trailing edge $x/c = 0.90 – 0.99$. The results of the cross-correlation of the surface pressure signals at various mid-span location along the chord starting at location $x/c = 0.72$ for the above mentioned three regions for both the HF and MF airfoils are presented in Fig. 4.39. The results for the first, second and third regions are presented in Figs 4.39a&amp;b, 4.39c&amp;d and 4.39e&amp;f, respectively. The cross-correlation $R_{p_i,p_j}$ for all the presented cases show positive $\tau_{c/U_\infty}$, which indicates that the flow travels in the direction of the free-stream flow. The auto-correlation function for all the three regions shows quick decay for all the presented cases.

For the HF airfoil at angles of attack $\alpha = 0^\circ$ and $4^\circ$, the cross-correlation results show that the pressure wave propagates slowly relative to the MF airfoil at the first region $x/c = 0.72 – 0.78$. This can be seen from the slow shift of the correlation peaks for the HF airfoil compared to the MF airfoil in the first region in Figs 4.39a&amp;b. The higher cross-correlation $R_{p_i,p_j}$ and the quick shift in the correlation peak between chord locations for the MF airfoil indicates the existence of a long-lasting energy field in the surface pressure compared to the HF airfoil. At the second region $x/c = 0.80 – 0.88$, for the angle of attack $\alpha = 0^\circ$, the MF airfoil in Fig. 4.39c shows fast-moving pressure wave with the attached flow. In the case of HF airfoil, signs of flow separation is clearly seen in the cross-correlation results with low cross-correlation values $R_{p_i,p_j} > 0.5$ especially at locations $x/c = 0.84 – 0.88$. This suggests that the turbulent structures
convected in the downstream direction are primarily dominated by a short-lived energy field \((R_{pi,pj} \text{ drops to 0.1 within } \tau U_\infty/c = 3)\).

For the angle of attack \(\alpha = 4^\circ\), the cross-correlation of the pressure fluctuations indicates flow separation for both the HF and MF airfoil. At the third region \(x/c = 0.90-0.99\) (see Figs. 4.39e&f), close to the trailing edge, the results for all the cases indicates separated flow as it has low cross-correlation values \(R_{pi,pj} > 0.5\) with slow shifting correlation peaks, especially at locations \(x/c > 0.92\), this also corresponds to the results seen in the Sections 4.2.3 and 4.3.2. The auto-correlation results of the MF airfoil exhibit a slow decaying behavior compared to the HF airfoil at the location \(x/c = 0.90\) for the angle of attack \(\alpha = 0^\circ\). The overall cross-correlation results in Fig. 4.39 show that the pressure wave propagates much faster over the flap surface for the MF airfoil compared to the HF airfoil. The flow separation is slightly delayed to the further downstream location for the MF airfoil \((x/c = 0.86)\) relative to HF airfoil \((x/c = 0.75)\) especially at angles of attack \(\alpha = 0^\circ\). The results here also corresponds to the observations made in Sections 4.2.3, 4.3.2 and 4.3.4.

Figure 4.38: Auto and cross-correlation of the surface pressure fluctuation at various chord locations \((x/c)\), on the suction side for Hinged Flap (solid line) and Morphed Flap (dashed line) airfoils at angles of attack \(\alpha = 0^\circ\) and \(4^\circ\) for probe order moving upstream of the airfoil trailing edge location \(x/c = 0.99\).
Figure 4.39: Auto and cross-correlation of the surface pressure fluctuation at various chord locations $(x/c)$, on the suction side for Hinged Flap (solid line) and Morphed Flap (dashed line) airfoils at angles of attack $\alpha = 0^\circ$ and $4^\circ$ for the probe order moving downstream of the flap $x/c = 0.72$. 

116
4.3. COMPUTATIONAL FLUID DYNAMICS

4.3.6 Wake velocity spectra

In order to further understand and characterize the wake turbulence behavior at the trailing edge of the MF and HF airfoil, the energy-frequency content of the turbulence structures has been studied. The power spectral density of the streamwise wake velocity ($\Phi_{uu}$) on the suction side at the trailing edge location $x/c = 1.01$ for the HF and MF airfoils at angles of attack $\alpha = 0^\circ$ and $4^\circ$ are presented in Fig. 4.40. The Welch power spectral density of the streamwise velocity fluctuations were performed based on the time-domain LES velocity signals using Hamming windowing for segments of equal length with 50% overlap. The results are presented in terms of the frequency and normalised crosswise distance ($y/c$) above the trailing edge, with 40 equally spaced data points within the crosswise distance of $y/c = 0 - 0.2$. The trailing edge of the airfoil is considered as the datum point, i.e. $y/c = 0$. The results presented in Fig. 4.40 for all the presented cases show an increased velocity spectra at low frequencies up to 2 kHz for a normalised distance of up to $y/c = 0.5$ above the airfoil trailing edge. The results are also presented in the form of line plots at selected location ($y/c = 0.05, 0.10, 0.15$) for the angle of attack $\alpha = 0^\circ$ and $4^\circ$ in Fig. 4.42.

In order to better visualize and understand the differences in the wake velocity spectra between all the cases, the difference between the streamwise velocity spectra of the MF airfoil ($\Phi_{uu})_{MF}$ and the HF airfoil ($\Phi_{uu})_{HF}$ was calculated and the results are presented in Fig. 4.41. The horizontal distortion lines seen in Fig. 4.41 are due to the coarse number of data acquisition point (40 data points for a distance of $y = 0.2c$) used for the contour plots. The results of the streamwise velocity spectra difference between the MF and HF airfoil at the angles of attack $\alpha = 0^\circ$ show an increase in the velocity spectra at low frequencies up to 2 kHz for the entire crosswise distance above the trailing edge $y/c = 0 - 0.2$. For the MF airfoil at the crosswise region $y/c = 0.1 - 0.11$, an increase at the frequency range 2 – 4 kHz is observed relative to the HF airfoil. The line plots in Fig. 4.42 shows an increase of up to 10 dB for the MF airfoil at low frequencies up to 2 kHz for angle of attack $\alpha = 0^\circ$ at the presented wake locations. This increased energy content for the MF airfoil might be due to the delayed flow separation closer to the trailing edge for the MF airfoil compare to the HF airfoil as seen earlier in Section 4.3.2.

The line plots for angle of attack $\alpha = 4^\circ$ in Fig. 4.42 do not show any substantial difference between the HF and MF airfoils. However, the difference in the wake velocity spectra in Fig. 4.41
for the angle of attack $\alpha = 4^\circ$, shows an increase in the energy content of the velocity spectra for the MF airfoil at around $0.2 - 1$ kHz at all the crosswise location and a reduction in the energy content at around $2 - 8$ kHz at crosswise location $y/c = 0.1 - 0.15$. Overall the results clearly show that there is an increase in the energy content of the velocity spectra at low-frequency range up to $2$ kHz for the MF airfoil at both tested angles of attack $\alpha = 0^\circ$ and $4^\circ$. 
4.3. COMPUTATIONAL FLUID DYNAMICS

Figure 4.40: Wake streamwise velocity spectra at location $x/c = 1.01$ close to the trailing edge for Hinged Flap and Morphed Flap airfoils at angles of attack $\alpha = 0^\circ$ and $4^\circ$.

Figure 4.41: Difference in wake streamwise velocity spectra at location $x/c = 1.01$ for the angles of attack $\alpha = 0^\circ$ and $4^\circ$. 

Figure 4.42: Wake streamwise velocity spectra at location $x/c = 1.01$ close to the trailing edge at various crosswise locations for Hinged Flap and Morphed Flap airfoils at angles of attack $\alpha = 0^\circ$ and $4^\circ$. 
4.3.7 Far-field noise

The far-field noise characteristic of the HF and MF airfoils calculated using the LES simulation will be discussed in this section. The airfoil noise calculations were performed using the Curle’s acoustic analogy [109], which is explained in detail in Appendix B. The source terms for the Curle’s analogy are the surface pressure fluctuations acquired at every time step of the LES simulations for the last 10 FTT. The far-field noise calculation method employed in this study has been proven successful in the author’s previous studies [89, 104]. The far-field noise calculations were made at an observer point 1.2 m ($r = 6c$) above the trailing edge of both the HF and MF airfoils. The sound pressure level results with reference to $p_{Ref} = 2 \times 10^{-5}$ Pa are presented in Fig. 4.43.

Figure 4.43: Acoustic prediction using Curle’s analogy, sound pressure level in dB reference to $p_{Ref} = 2 \times 10^{-5}$ Pa, at observer point 1.2 m ($x/c = 1, y/c = 6$) above the trailing edge for Hinged Flap and Morphed Flap.

The results at the angle of attack $\alpha = 0^\circ$ show that the sound pressure level for the MF airfoil is up to 10 dB higher than the HF airfoil at low frequencies ($f < 1$ kHz). The noise levels for both the HF and MF airfoils show an insignificant difference between each other at mid to high-frequency range ($f > 1$ kHz). At the angle of attack $\alpha = 4^\circ$, the difference in sound pressure level between the two airfoils drops to $\approx 5$ dB at low frequency ($f < 800$ Hz), with the MF airfoil having higher noise levels. This is consistent with the increased surface pressure fluctuations in the trailing edge area of the MF airfoils (see Figs. 4.33 and 4.34) and with the increased spanwise
coherence seen earlier in Figs. 4.35 and 4.36. The over all sound pressure level (OASPL) for the HF and MF airfoils were calculated using Eq. 3.9 and the results are presented in Table 4.1. The results show at angle of attack $\alpha = 0^\circ$ the MF airfoil produces 4.6 dB more than that of the HF airfoil and for angle of attack $\alpha = 4^\circ$ the MF airfoil produces 2.9 dB more than that of the HF airfoil.

The results clearly depict that the MF airfoil generates higher noise than that of the HF airfoil for both the tested angles of attack. From the results, it can be seen that the higher aerodynamic performance for the MF airfoil seen earlier also results in higher noise levels compared to the HF airfoil.

Table 4.1: The overall sound pressure level at observer point 1.2 m ($x/c = 1, y/c = 6$) above the trailing edge for Hinged Flap and Morphed Flap.

<table>
<thead>
<tr>
<th></th>
<th>Hinged Flap</th>
<th>Morphed Flap</th>
<th>Difference</th>
</tr>
</thead>
<tbody>
<tr>
<td>$\alpha = 0^\circ$</td>
<td>21.7 dB</td>
<td>26.3 dB</td>
<td>4.6 dB</td>
</tr>
<tr>
<td>$\alpha = 4^\circ$</td>
<td>21.8 dB</td>
<td>24.7 dB</td>
<td>2.9 dB</td>
</tr>
</tbody>
</table>

4.4 Conclusions

The aerodynamic and aeroacoustic performance of a NACA 0012 airfoil fitted with Hinged- and Morphed Flaps were investigated using experimental and numerical techniques. The airfoil was tested for a flow velocity of $U_\infty = 20$ m/s, corresponding to a chord-based $Re_c = 2.6 \times 10^5$. Surface flow visualization, aerodynamic lift and drag measurements and flow velocity measurements using PIV technique were carried out to better understand the flow characteristics of the HF and MF airfoils. The lift and drag measurements showed an increase of up to $C_{L,max} = 14\%$ for the MF airfoil relative to the HF airfoil. The $C_L/C_D$ performance at low angles of attack showed an improvement of up to 6% for MF airfoil. The stall angle was not changed for the MF airfoil. Surface flow visualization results showed delayed separation over the flap suction side for the MF airfoil compared to the HF airfoil at low angles of attack $\alpha > 8^\circ$, after which the separation point over the flap region between the two airfoils were indifferent.

The mean velocity results showed increased wake deficit for the MF airfoil compared to
4.4. CONCLUSIONS

the HF airfoil at the near-wake locations, $x/c = 1.025, 1.05$ and $1.10$, along with increased flow
deflection angle at far-wake locations, $x/c = 1.20, 1.40$ and $1.60$. The results of the turbulent
kinetic energy at the airfoil wake was up to 50% higher for the MF airfoil relative to the HF
airfoil for all the presented angles of attack. The turbulent kinetic energy results for both the HF
and MF airfoil displayed a characteristic double peak behavior at the near- and far-wake regions.
The two distinct regions at the wake with high turbulent kinetic energy was mainly contributed
by the streamwise normal Reynolds stress component ($u'u'$). The peak values of the $u'u'$ can be
observed close to the trailing edge point with the MF airfoil having relatively higher values. The
Reynolds shear stress component ($-u'v'$) for the MF airfoil is larger than that of the HF airfoil
at the near-wake locations $x/c = 1.025, x/c = 1.05$ and $x/c = 1.10$ for all the presented angles of
attack. The MF airfoil shows improved $C_L/C_D$ performance compared to the HF airfoil this was
found. The increased wake deflection angle for the MF airfoil results in the increased lift and the
larger region of attached flow near the trailing edge resulting in the reduction of form drag and
improved $C_L/C_D$ for the MF airfoil.

High fidelity experimentally validated LES simulations were also carried out for a selected
number of cases ($\alpha = 0^\circ$ and $4^\circ$) to further characterise the flow around the airfoils. The surface
pressure root mean squared results from the LES simulations showed increased fluctuations for
the HF airfoil right after the flap hinge point at $x/c = 0.75$ but the region of high fluctuations is
observed at further downstream locations closer to the trailing edge for the MF airfoil at both
the angles of attack. The evolution of wall pressure spectra showed increased energy content for
the MF airfoil at low frequency $0.2 - 2$ kHz at regions closer to the trailing edge. The spanwise
coherence of the surface pressure results showed that the MF airfoil had larger coherence at low
frequency $0.2 - 2$ kHz at the vicinity of the trailing edge. The results of the cross-correlation of
the streamwise surface pressure showed that the pressure fluctuations generated larger flow
structure for the MF airfoil relative to the HF airfoil. The far-field noise measurements were
calculated using Curle’s acoustic analogy and the results show increased noise for the MF airfoil
and this is correlated to the increased pressure fluctuations that are close to the trailing edge for
the MF airfoil. The results clearly shows that even slight modification to the surface camber of
an airfoil flap gives considerable aerodynamic gains at low angles of attack ($\alpha > 8^\circ$), however, the
results also show that it comes at a cost of increased noise.
5.1 Introduction

The impact of aircraft noise on the communities near the airports has been an issue since the entry of turbofan and turbojet engines into civil aviation from the 1960s and 1970s. The widespread global expansion of air travel has made the environmental impact of aircraft noise much more prominent in recent times. This has forced the International Civil Aviation Organisation (ICAO) to set technical standards for civil air transport aircraft and 180 countries have adopted this. With such upcoming regulations to reduce noise impact on communities near the airports, further understanding of aircraft noise has to be achieved. The introduction of high bypass-ratio turbofans engines into civil aircraft have drastically reduced engine jet noise over the last several decades, making the airframe noise the same magnitude as that of the engine noise, especially during the landing phase. One of the main sources of airframe noise is the high-lift devices, namely the slat and flap. In order to reduce these prominent noise sources several passive and active flow control methods have been investigated in the past, such as, morphing structures [64, 66, 67, 100, 101], porous materials [83, 114, 115], surface treatments [116] and serrations [117, 118].
Studies on conventional slat and wing configurations have shown that it comprises both the broadband and tonal noise components. Several studies on slat noise have shown several discrete tones at mid-frequency range \([70, 119–132]\). Apart from these discrete tones a spectral hump at \(St_s > 1\) was observed in several studies \([124, 127, 129–132]\). The source of which has not yet been isolated. The overall slat noise is generated from the unsteady flow within the slat cove region originating from the vortex shedding at the slat cusp and the slat trailing edge. The aeroacoustic mechanism of the slat with such unique flow characteristics is yet to be fully understood. However, recent experimental and numerical evidence is indicative that the tonal peaks are associated with the cavity feedback mechanism and resonance. A possible quadratic interaction between the tonal peaks has also been observed \([125–128]\). The observed tonal peaks decrease with the angle of attack but their amplitude decreases with increasing slat gap and overlap \([126]\). It is, therefore, essential to understand the noise generation mechanism so that the problem can be addressed in the design phase of the slat and wing.

Several experimental and computational studies \([72–74, 133–139]\) were conducted in the past decade to investigate and reduce the broadband noise arising from within the slat cove region by casing the recirculation region. The broadband noise reduced marginally and the tonal peaks were eliminated at all the instances. However, the aerodynamic performance of the slat cove filler (SCF) configuration is yet to be thoroughly documented. The approach of filling the slat cove region to reduce noise is based on eliminating the strong shear layer created after the slat cusp and avoiding the development of complicated flow structure within the slat cove region by using a smoothly contoured profile. In order to eliminate the unsteady recirculation region within the slat cove, Horne et al. from NASA tested a solid SCF on a Boeing 777-200 semi-span model in the NASA Ames 40 by 80 foot Wind Tunnel. The SCF profiles were derived from a CFD analysis in order to maintain attached flow on the slat pressure surface. The experiments used a microphone phased array and the results showed that the SCF was effective in reducing the broadband slat noise up to 4-5 dB \([133]\). However, no aerodynamic measurements were presented in this study. Streett et al. further investigated the noise and basic aerodynamic performance of the SCF setup using trapezoidal wing swept model fitted \([134]\). The results showed noise reduction to be sensitive to the angle of attack and SCF modification. The SCF modification showed a reduction...
of up to 3-5 dB over a wide spectrum. The aerodynamic performance appeared marginally better than the Baseline at angles of attack below 20° and the stall occurred 2 degrees earlier compared to the Baseline. The specific reason for the loss in aerodynamic performance was not pointed out due to the lack of aerodynamic data, such as detailed surface pressure and wake shear layer measurements.

Imamura et al. and Ura et al. from JAXA showed experimentally and computationally [72, 73] that even though noise reduction can be achieved by the use of SCF and that the SCF profile affects the aerodynamic lift characteristics of the three-element airfoil. They tested two SCF profiles that were designed based on the flow field streamlines of angles of attack 0° and 8° on an MDA 30P30N airfoil. Even though the results showed a reduction of up to 5 dB for both the cases, they found that the aerodynamic performances were the same as that of the Baseline only for the SCF profile made from the flow field streamlines at the angle of attack 8°. Whereas, for the SCF profile made from the streamlines of the angle of attack 0° the airfoil stalls prematurely. In a very recent optimization study, Tao and Sun [74] performed several Detached eddy simulations using 44 configurations of the SCF profile designs aimed to produce maximum lift coefficient for fixed design point with an angle of attack of 22° and \( Re_c = 9 \times 10^6 \). The final optimized SCF profile showed a reduction in noise while maintaining the aerodynamic performance.

Even though several studies [72–74, 133–139] were performed on the noise reduction capabilities of the SCF, only basic aerodynamic and noise measurements were presented. This study presents detailed experimental results of high-lift 30P30N airfoil compared with two slat cove filler cases. This chapter is organised as follows: Section 1.2 reports the aerodynamic results, which presents the aerodynamic force such as lift and drag measurements and coefficient of pressure distribution around the high-lift airfoil. Section 1.3 reports the detailed flow field contours around the airfoil, slat wake analysis and Proper orthogonal decomposition analysis of the PIV results. Section 1.4 reports the detailed aeroacoustic results, which presents the far-field spectral levels, near-field spectral levels, spanwise coherence, correlation length scales, continuous wavelet transform of the near-field measurements, higher order spectral analysis of the near-field measurement and persistence spectrum of the near-field measurements. Finally, Section 1.5 presents an overall summary and conclusions of this chapter.
5.2 Aerodynamic results

To gain a better understanding of the aerodynamic performance of the MDA 30P30N airfoil, lift and drag measurements, and surface pressure distribution measurements were carried out. The high-lift airfoil was equipped with 103 pressure taps to accurately capture the surface pressure distribution over all the three components of the 30P30N high-lift device. The high-lift airfoil setup and instrumentation are discussed in detail in Section 3.1.3. The high-lift airfoil was tested for a range of angles of attack from $\alpha = 0^\circ$ to $15^\circ$ at the free-stream velocities of $U_\infty = 20, 30, 40$ and $47$ m/s. The tested configurations were the Baseline, Half-slat cove-filler (H-SCF), Slat cove-filler (SCF) and Slat cove-filler along with Flap cove-filler (SCF-FCF), as shown in Fig. 3.13.

Figure 5.1: The MDA 30P30N Baseline airfoil fitted with half-slat cove filler (H-SCF), slat cove filler (SCF) and flap cove filler (FCF).
5.2. AERODYNAMIC RESULTS

5.2.1 Aerodynamic force measurements

The aerodynamic lift and drag measurement results for the MDA 30P30N airfoil with different cove fillers at the free-stream velocity of $U_\infty = 40$ m/s, corresponding to a chord-based Reynolds number of $Re_c = 9.3 \times 10^5$ are presented in Fig. 5.2. The tests were carried out for the angles of attack ranging from $\alpha = 0^\circ$ to $18^\circ$ with an increment of $2^\circ$. The lift and drag coefficients ($C_L$ and $C_D$) for the Baseline, H-SCF, SCF, and the SCF-FCF configurations are presented in Fig. 5.2. The $C_L$ results show an insignificant difference between the cases for all the presented angles of attack. The $C_D$ results, on the other hand, show that the SCF-FCF configuration produced the highest $C_D$ compared to the Baseline and all the other configurations at all the angles of attack. The H-SCF configuration has the least $C_D$ relative to the Baseline and the other configurations for all the tested angles of attack. The $C_D$ values for the SCF case is the same as that of the Baseline. Therefore, it can be inferred that the use of slat cove fillers have a more pronounced effect on the drag than the lift generated by the airfoil.

![Figure 5.2: Lift and drag coefficients for the 30P30N airfoil with various cove fillers at chord-based Reynolds number $Re_c = 9.3 \times 10^5$.](image)

The results for the lift-to-drag ratio and the drag polar curves for the Baseline, H-SCF, SCF, and the SCF-FCF configurations are presented in Fig. 5.3. The lift-to-drag ratio ($C_L/C_D$) results show a significant difference between the different configurations. The H-SCF produces a higher $C_L/C_D$ relative to the Baseline and the SCF-FCF configuration. For the SCF configuration with a large cove filler, the $C_L/C_D$ values remain the same as that of the best performing H-SCF between
$\alpha = 0^\circ$ to $8^\circ$ and $16^\circ$ to $18^\circ$. The highest change in $C_L/C_D$ value was observed for the H-SCF case relative to the Baseline between $\alpha = 8^\circ$ and $14^\circ$ with an average increase of $\Delta C_L/C_D \approx 0.214$ (≈ 5.4%). The largest difference in $C_L/C_D$ is found at $\alpha = 6^\circ$ between the H-SCF and SCF-FCF cases, with a $\Delta C_L/C_D = 0.358$. The drag polar curve results in Fig. 5.3b show the $C_D$ in the abscissa and $C_L$ in the ordinate for increasing angles of attack. The results clearly show that the H-SCF has the least drag and highest lift for all the presented angles of attack, while the SCF-FCF has the highest drag and least lift compared to the other configurations.
5.2.2 Pressure coefficient distribution

The pressure coefficient ($C_p$) distribution, calculated from the static pressure measurements acquired along the mid-span of the high-lift device, for various chord-based Reynolds numbers, $Re_c = 4.9 \times 10^5, 7.0 \times 10^5, 9.3 \times 10^5$ and $1.1 \times 10^6$, at the angle of attack $\alpha = 12^\circ$ is presented in Fig. 5.4. The results show that the changes in $C_p$ distribution over the slat and main-element are insignificant for the tested Reynolds numbers except for the flap suction peak. The suction peak ($C_p$) of the main-element showed an increase of only 1.5% for $Re_c = 9.3 \times 10^5$ relative to $Re_c = 4.9 \times 10^5$, whereas the changes on the suction peak of the flap were up to 15% higher for $Re_c = 9.3 \times 10^5$ and 20% higher for $Re_c = 1.1 \times 10^6$ relative to the $Re_c = 4.9 \times 10^5$. Valarezo [42, 43] showed that the effects of Reynolds number on the lift of multi-element airfoil was very evident for flow conditions below $Re_c = 4 \times 10^6$. They also showed a considerable increase in the maximum lift between $Re_c = 2 \times 10^6$ and $9 \times 10^6$ at a Mach number of 0.2. The effects of Reynolds number and its significance on the lift of high-lift airfoil was also shown by Chin et al. [44] and they also discussed the increased effect of Reynolds number on the suction peak of the main-element and flap. As seen in the previous studies the effect of Reynolds number on the suction peaks can also be seen in the present study but only with a marginal magnitude.

![Figure 5.4: Pressure coefficient distribution over 30P30N Baseline airfoil for various chord-based Reynolds numbers at angle of attack $\alpha = 12^\circ$.](image)

The pressure coefficient distribution for the Baseline at the angles of attack $\alpha = 8^\circ, 12^\circ, 14^\circ, 16^\circ$ and $18^\circ$ compared with the experimental measurements by Li et al. [129] at $\alpha = 8^\circ$ are
presented in Fig. 5.5. The results validate well with the existing experimental measurements. The increase in the $C_p$ distribution on the suction side of the main-element as the angle of attack is increased is evident. The loading on the slat and main-element increases with the increase in the angle of attack. The increased suction peak on the main-element at higher angles of attack is due to the higher velocity from the increased mass flow through the slat gap as the angle of attack is increased. The suction peak on the upper surface of the main-element increases up to $\approx 1\%$, $6.8\%$ and $14\%$ for $\alpha = 14^\circ$, $16^\circ$ and $18^\circ$, respectively, relative to $\alpha = 12^\circ$. The suction peak on the upper surface of the flap increases up to $\approx 5.6\%$ for $\alpha = 18^\circ$ relative to $\alpha = 12^\circ$. 

Figure 5.5: Pressure coefficient distribution over 30P30N Baseline airfoil for various angles of attack at a free-stream velocity of $U_\infty = 30$ m/s, $Re_c = 7.0 \times 10^5$ [129].

Figure 5.6: Pressure coefficient distribution over 30P30N Baseline airfoil around the slat and flap region for a free-stream velocity of $U_\infty = 30$ m/s, $Re_c = 7.0 \times 10^5$. 

(a) Slat close up  
(b) Flap close up
5.2. AERODYNAMIC RESULTS

Figures 5.7 and 5.8 show the pressure coefficient $C_p$ results for the Baseline, H-SCF, SCF and SCF-FCF configurations at the angles of attack $\alpha = 12^\circ, 14^\circ, 16^\circ$ and $18^\circ$ for a free-stream velocity of $U_\infty = 30$ m/s. The results in Fig. 5.7 show that the modifications on the slat, such as the application of H-SCF and SCF, affect the suction peak on the main-element of the airfoil. The Baseline case has the highest suction peak for all the presented angles of attack. The $C_p$ suction peak on the main-element at location $x/c = 0.043$ was reduced by approximately 12% at $\alpha = 12^\circ$ and approximately 15% at $\alpha > 12^\circ$ for the H-SCF, SCF and the SCF-FCF cases relative to the Baseline case. The results for the slat in Fig. 5.8 show that the $C_p$ on the pressure side changes quite significantly for the H-SCF and SCF configurations as the angle of attack is increased. The suction peak near the slat cusp is decreased for the H-SCF and SCF configuration relative to the Baseline by up to 40% for the angle of attack $\alpha = 12^\circ$. This is due to the absence of the sudden

Figure 5.7: Pressure coefficient distribution over 30P30N airfoil with slat modifications, at various angles of attack for a free-stream velocity of $U_\infty = 30$ m/s, $Re_c = 7.0 \times 10^5$. 

(a) $\alpha = 12^\circ$
(b) $\alpha = 14^\circ$
(c) $\alpha = 16^\circ$
(d) $\alpha = 18^\circ$
pressure gradient and the increased velocity due to the streamlined profile of the cove fillers. The $C_p$ measurements over the flap for the presented angles of attack remains unchanged for the Baseline, H-SCF and SCF configurations. The results for the SCF-FCF configuration show an increase of up to $\approx 20\%$ in the $C_p$ on the pressure side of the main-element at the location of the flap cove filler between $x/c = 0.6$ and 0.8. The SCF-FCF results over the flap at the suction peak location $x/c = 0.90$ is $\approx 5\%$ higher for the SCF-FCF configuration relative to the Baseline.
5.3 Flow field analysis

In order to thoroughly investigate the flow field within the slat cove region for the 30P30N airfoil with and without slat cove fillers, Particle image velocimetry studies were carried out. The Particle image velocimetry setup for the high-lift airfoil is described in detail in Section 3.2.3. The measurements were carried out for several windows on the suction and pressure side of the 30P30N airfoil as shown in Fig. 3.10. However, the results are presented for selective windows for the purpose of brevity. The contour plots of the mean streamwise and crosswise velocity, turbulent kinetic energy and the Reynolds stress tensors for all tested three configurations in Section 5.3.1. For better comparison between the tested airfoil configurations, the flow field component profiles extracted at strategic locations at the slat wake and over the main-element are presented and discussed in Section 5.3.2. In order to further understand the energy contained within the slat cove region Proper Orthogonal decomposition has been carried out and it is discussed in Section 5.3.3.

5.3.1 Flow field visualization

Detailed Particle Image Velocimetry (PIV) studies were performed in and around the slat and flap region for the Baseline, H-SCF, SCF and SCF-FCF configurations at the angles of attack $\alpha = 6^\circ$, $8^\circ$, $10^\circ$ and $12^\circ$ with a free-stream velocity of $U_\infty = 30$ m/s ($Re_c = 7.0 \times 10^5$) and the results are presented in Figs. 5.9 to 5.12. The figures presented in a table format with the columns showing the different slat configuration and the rows showing the various increasing angles of attack. The contours of the non-dimensional mean streamwise ($U/U_\infty$) and crosswise ($V/V_\infty$) velocity distribution around the slat region with streamlines showing the flow direction are shown in Figs. 5.9 and 5.10, respectively. For the Baseline airfoil, the streamlines show that the shape and structure of the vortices present within the slat cove region is largely influenced by the angle of attack (see Figs. 5.9). The magnitude of the negative streamwise velocity that arises right after the flow impingement ($x/c \approx 0.05$) on the main-element appears to be influencing the trajectory of the slat shear layer leaving the slap cusp. The trajectory of the slat shear layer is also influenced by the angle of attack. At $\alpha = 6^\circ$, the vortices within the slat cavity appears to be the largest
with the longest slat shear layer trajectory, which impinges much closer to the slat trailing-edge and most of the flow after impingement moves toward the slat trailing-edge and mixes into the free-stream. The impingement point of the slat shear layer on the slat lower surface moves away from the slat trailing-edge toward the slat mid-chord location as the angle of attack is increased. This slat shear layer trajectory with a much shorter path before the impingement restricts the recirculation area at increased angles of attack. This decreases the recirculation area resulting in the higher vortex velocity and it also increases the crosswise velocity inside the slat cove region. The increased flow through the slat gap along with the higher negative streamwise velocity at the main-element impingement region appears to be the key factors influencing the movement of the slat shear layer trajectory with the change in angle of attack. The contours show negative velocity inside the slat cove region, which can be associated with the vortices. The highest negative streamwise velocity on the slat lower surface at \( \alpha = 12^\circ \) implies highest vortex velocity amongst the tested angles of attack. The highest streamwise velocity on the upper side can be seen for \( \alpha = 12^\circ \) over the main-element right after the slat gap where the velocity reaches up to twice as much as that of the inlet velocity. The highest velocity on the lower side occurs near the slat cusp where the slat shear layer originates. For the Baseline case, the maximum value of the crosswise velocity occurs at the slat gap region with increased velocity seen at \( \alpha = 12^\circ \) compared to all the other angles of attack. The maximum crosswise velocity lies between the free slat shear layer and the main-element of the 30P30N airfoil for all the presented angles of attack.

The effects of H-SCF and SCF on the flow structure within the slat cove region are minimal as the shape and trajectory of the slat shear layer follows the same trend as that of the Baseline for all the tested angles of attack. However, the size of the vortical structures inside the slat cove region is reduced noticeably. The use of the cove filler inside the slat leads to the elimination of the large vortices within the slat cove region as the available area for recirculation is occupied by the cove fillers. However, closer to the slat trailing-edge, on the lower surface of the SCF, smaller vortices have emerged. Similar to the Baseline airfoil, the size and magnitude of these vortices are clearly influenced by the angle of attack. These vortices also arise right after the impingement of the slat shear layer onto the slat lower surface, as previously seen in the case of the Baseline airfoil. Olson et al. [140] showed that the favorable pressure gradient between
the slat upper and lower surface at the slat cusp accelerates and energizes the flow, which also influences the strength and trajectory of the slat shear layer. The cove filler configurations have completely eliminated this favorable pressure gradient at the slat cusp, thus reducing the energy of the existing limited shear layer. The existing smaller vortices can be completely prevented by having an SCF profile that follows the same profile as that of the slat shear layer trajectory. However, this could prove difficult for practical operation as this slat shear layer trajectory is not only dependent on the angle of attack but also on the operating Reynolds and Mach number. If the SCF profile is larger than the slat shear layer profile then the flow at the slat gap gets restricted, which consequently affects the suction peak and aerodynamic performance of the main-element. Nevertheless, an SCF profile that eliminates the large vortices in the slat cove region, while maintaining the aerodynamic performance at the same time is highly favourable as they are viable sources of noise reduction, as shown by Imamura et al. [72, 73], Tao [74] and also in the current experimental study (see Section 5.2).

The non-dimensional streamwise ($u'\overline{u'}/U_\infty^2$) and crosswise ($v'\overline{v'}/U_\infty^2$) Reynolds normal stress tensors around the slat region for the Baseline, H-SCF and SCF configurations for all the tested angles of attack are presented in Figs. 5.11 and 5.12, respectively. The presented results of the normal Reynolds stress components ($u'\overline{u'}$ and $v'\overline{v'}$) show that the crosswise Reynolds normal stress components ($v'\overline{v'}$) are higher than that of the streamwise Reynolds normal stress components ($u'\overline{u'}$) for all the presented configurations and angles of attack. The maximum value of the $u'\overline{u'}$ components for all the configurations can be found at the originating location of the slat shear layer adjacent to the slat cusp and also at the vicinity of the slat trailing edge. The maximum value of the $v'\overline{v'}$ components can be observed at the slat gap region close to the suction side of the main-element for both the Baseline and the H-SCF configurations. However, the $v'\overline{v'}$ components for the SCF configuration is slightly reduced at the slat gap region but increased values of it can be observed on the lower surface of the SCF itself. Similar behavior can be observed for all the presented angles of attack. The results also show that the shear stress distribution for both the normal eddy stress components ($u'\overline{u'}$ and $v'\overline{v'}$) reduces as the angle of attack is increased for all the three configurations.
Figure 5.9: The normalised mean streamwise velocity ($U/U_\infty$) contours around the slat region for various angles of attack with a free-stream velocity of $U_\infty = 30$ m/s, $Re_c = 7.0 \times 10^5$. 

Baseline

H-SCF

SCF
Figure 5.10: The normalised mean crosswise velocity ($V/U_\infty$) contours around the slat region for various angles of attack with a free-stream velocity of $U_\infty = 30$ m/s, $Re_c = 7.0 \times 10^5$. 
Figure 5.11: The normalised streamwise Reynolds normal stress ($\overline{u'u'}/U_\infty^2$) contours around the slat region for various angles of attack with a free-stream velocity of $U_\infty = 30$ m/s, $Re_c = 7.0 \times 10^5$. 

Baseline  |  H-SCF  |  SCF  
---|---|---
(a) $\alpha = 6^\circ$ | (b) $\alpha = 6^\circ$ | (c) $\alpha = 6^\circ$
(d) $\alpha = 8^\circ$ | (e) $\alpha = 8^\circ$ | (f) $\alpha = 8^\circ$
(g) $\alpha = 10^\circ$ | (h) $\alpha = 10^\circ$ | (i) $\alpha = 10^\circ$
(j) $\alpha = 12^\circ$ | (k) $\alpha = 12^\circ$ | (l) $\alpha = 12^\circ$
Figure 5.12: The normalised crosswise Reynnolds normal stress ($\overline{v'v'}/U_\infty^2$) contours around the slat region for various angles of attack with a free-stream velocity of $U_\infty = 30$ m/s, $Re_c = 7.0 \times 10^5$. 
5.3.2 Slat wake development

Figure 5.13: Boundary layer measurement locations for the MDA 30P-30N airfoil.

Table 5.1: Slat trailing-edge wake measurement locations.

<table>
<thead>
<tr>
<th>No.</th>
<th>x (mm)</th>
</tr>
</thead>
<tbody>
<tr>
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<td>147.0</td>
</tr>
<tr>
<td>$S_{w-10}$</td>
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</tbody>
</table>

The mean velocity profiles extracted from the PIV data at various streamwise locations (see Fig. 5.13) over the main-element of the MDA 30P30N airfoil is presented in Fig. 5.14. The results show that the slat wake deficit peak at the locations $x/c = 0.0575$ and 0.1057 is the highest for the Baseline compared to both the H-SCF and SCF at all the presented angles of attack. This is due to the unrestricted airflow through the slat gap, which is energized by the free shear impingement for the Baseline case. The results for the angle of attack $\alpha = 6^\circ$ show that the slat wake deficit for both the H-SCF and SCF is much lower than that of the Baseline at the location $x/c = 0.0575$ and 0.1057 compared to the angle of attack $\alpha = 12^\circ$. At locations, $x/c = 0.2285, 0.6000, 0.8933$ and 1.0160 the mean velocity results show that the slat modifications have insignificant effects on the flow behavior over the multi-element MDA 30P30N airfoil for the tested configurations and angles of attack.
5.3. FLOW FIELD ANALYSIS

Figure 5.14: Mean velocity profiles over the MDA 30P30N airfoil at various streamwise locations for the free-stream velocity $U_\infty = 30$ m/s, for Baseline —, H-SCF ––– and SCF –·–·.

The slat wake development at various near and far slat wake locations (see Table. 5.1) for the Baseline, H-SCF, and SCF configurations are presented for angles of attack $\alpha = 6^\circ$ to $12^\circ$ in Figs. 5.16 to 5.22 respectively. The results are presented for eight streamwise locations at the near-wake location of the slat with the slat trailing-edge as the datum point. The first six slat near-wake locations $x = 15.0, 19.9, 26.5, 35.3, 46.9$ and $62.4$ mm are located between the slat trailing-edge and the main-element. The last four slat far-wake locations $x = 82.9, 110.0, 147.0$ and $195.0$ mm are located at the slat wake just above the main-element. The results of the non-dimensional streamwise velocity ($U/U_\infty$) at angle of attack $\alpha = 6^\circ$ in Figs. 5.15a and 5.15b show a noticeable change for the slat wake profile of the H-SCF (–––) configuration toward the lower side of the slat-trailing edge with increased wake deficit compared to the Baseline and
SCF configurations. This increased wake deficit on the lower side for the H-SCF also affects the boundary-layer flow over the main-element as seen in Fig. 5.15b. The results for the SCF (- - -) streamwise velocity profile follows similar trend to that of the Baseline (—) with negligible dissimilarities. The H-SCF slat wake also affects the flow above the main-element as seen at location \( x = 82.9 \text{ mm} \) (Fig. 5.15b). The strength of the slat wake above the main-element reduces at further downstream locations \( (x > 147.0 \text{ mm}) \). The results of the TKE at slat wake in Figs. 5.15c and 5.15d clearly show increased TKE for the H-SCF compared to the Baseline and the SCF configurations at all the presented slat wake locations especially in region below the slat trailing-edge and above the main-element, whereas, the results for the Baseline and the SCF case are very similar to each other. The use of H-SCF has evidently affected the shear layer and flow through the slat gap by reducing the flow velocity and increasing the TKE compared to the Baseline and SCF configurations at low angles of attack.

The results of the non-dimensional Reynolds stress components at the angle of attack \( \alpha = 6^\circ \) also show significant differences for the H-SCF configuration with an increased magnitude relative to the Baseline and the SCF configurations. Even though the streamwise normal \((u'\overline{u'})\) Reynolds stress profile for the SCF shows similar trends to that of the Baseline case at \( \alpha = 6^\circ \), the peak magnitude of the crosswise normal Reynolds stress component \((v'\overline{u'})\) at slat wake for the SCF shows increased magnitude relative to the Baseline. At the angle of attack \( \alpha = 8^\circ \), the mean streamwise velocity component show velocity increase for the H-SCF and SCF configuration at the slat wake region \((x = 15.0 – 62.4)\) in Fig. 5.17a. The results also show increased TKE for the H-SCF and SCF configurations compared to the Baseline airfoil. The non-dimensional streamwise velocity shows insignificant differences between the three configurations with slight increase in velocity for the H-SCF and SCF configuration. The crosswise velocity for the SCF case has higher magnitude at all the slat wake and main-element boundary layer locations relative to the Baseline and H-SCF configuration.

The non-dimensional velocity components, turbulent kinetic energy, and Reynolds stresses for the angle of attack \( \alpha = 10^\circ \) and \( 12^\circ \) are shown in Figs. 5.21 to 5.22. The results show insignificant differences in the non-dimensional streamwise velocity between the three different configurations at the angle of attack \( \alpha = 10^\circ \) and \( 12^\circ \). The results show higher velocity at near wake locations.
(x = 15.0, 19.9 and 26.5) for $\alpha = 12^\circ$ relative to the $\alpha = 6^\circ$. The results of the non-dimensional streamwise velocity on the upper side of the slat wake reach up to $U/U_\infty = 1.3$ and on the lower side it reaches up to $U/U_\infty = 1$. This shows that the flow has accelerated on the upper and lower side of the slat trailing-edge at $\alpha = 12^\circ$ relative to the $\alpha = 6^\circ$. At the angle of attack $\alpha = 10^\circ$, the non-dimensional crosswise velocity components between the Baseline and H-SCF configuration show no difference at the slat wake location but for the SCF configuration, it shows increased magnitude compared to both the Baseline and H-SCF configuration. The crosswise velocity at $\alpha = 12^\circ$ is indifferent between each other for all the configurations at the slat wake and main-element boundary layer locations. The Reynolds stresses results show increased magnitude at the location of the trailing edge in the slat wake. The three different configurations follow the same trend for the presented results at $\alpha = 12^\circ$. The crosswise normal Reynolds stress component ($\overline{v'v'}$) shows decreased magnitude at the peak location for both the H-SCF and SCF relative to the Baseline. The results in the far-wake locations ($x > 62.4$ mm) show a double peak for the $\overline{v'v'}$ component with one peak at the boundary layer of the main-element and the other peak at the slat wake above the main-element. At the increased angle of attack $\alpha = 10^\circ$, and $12^\circ$, the results of the mean velocity field and the turbulent kinetic energy profiles show no change or insignificant change between the Baseline, H-SCF, and SCF configurations. The results here show that the use of H-SCF and SCF does not necessarily affect the flow at the slat wake and the boundary layer over the main-element. The slightly increased mean velocity components observed at the slat near-wake region for the H-SCF and SCF configuration also corresponds to the increased aerodynamic performance seen in Section 5.2.
Figure 5.15: Mean velocity and turbulent kinetic energy profiles at the slat wake for $\alpha = 6^\circ$ at the free-stream velocity $U_\infty = 30$ m/s, for Baseline —, H-SCF —— and SCF —·—.
5.3. FLOW FIELD ANALYSIS

Figure 5.16: Reynolds stress tensor profiles at the slat wake for $\alpha = 6^\circ$ at the free-stream velocity $U_{\infty} = 30$ m/s, for Baseline —, H-SCF – – – and SCF – – –.
Figure 5.17: Mean velocity and turbulent kinetic energy profiles at the slat wake for $\alpha = 8^\circ$ at the free-stream velocity $U_\infty = 30$ m/s, for Baseline —, H-SCF ––– and SCF ––.
5.3. FLOW FIELD ANALYSIS

Figure 5.18: Reynolds stress tensor profiles at the slat wake for $\alpha = 8^\circ$ at the free-stream velocity $U_\infty = 30$ m/s, for Baseline —, H-SCF —— and SCF —.—.
Figure 5.19: Mean velocity and turbulent kinetic energy profiles at the slat wake for $\alpha = 10^\circ$ at the free-stream velocity $U_\infty = 30$ m/s, for Baseline —, H-SCF –– – and SCF –– –.
Figure 5.20: Reynolds stress tensor profiles at the slat wake for $\alpha = 10^\circ$ at the free-stream velocity $U_\infty = 30$ m/s, for Baseline —, H-SCF —— and SCF ---.
Figure 5.21: Mean velocity and turbulent kinetic energy profiles at the slat wake for $\alpha = 12^\circ$ at the free-stream velocity $U_\infty = 30$ m/s, for Baseline —, H-SCF —– and SCF —. 
5.3. FLOW FIELD ANALYSIS

Figure 5.22: Reynolds stress tensor profiles at the slat wake for $\alpha = 12^\circ$ at the free-stream velocity $U_\infty = 30$ m/s, for Baseline —, H-SCF —— and SCF ---.
5.3.3 Proper Orthogonal Decomposition

Proper Orthogonal Decomposition (POD) has shown to be an effective method for identifying dominant flow features, such as large coherent structures in a turbulent flow. The small turbulent flow motions often compile to make the large-scale turbulent structures, which can be identified using POD. POD analysis is best suited for problems that involve regular vortex shedding, such as the slat cove flow discussed in the present study. This method has been used by researchers for a variety of flow problems such as airfoils, cavities, bluff-bodies, and jets to isolate (dominant) periodic flow phenomena [128, 144–148]. Previous studies [144–146] used POD to analyze the slat cove dynamics at several angles of attack at a chord-based Reynolds number of \( Re_c = 6.5 \times 10^5 – 1.3 \times 10^6 \), similar to that of the present study. The results showed the existence of smaller structures within the shear layer. It was also suggested that the presence of these smaller features within the shear layer itself and their movement past the slat and main-element would likely result in the generation of high frequency noise levels.

In the present study, snapshot POD [149] is used on the dense vector fields acquired from the PIV measurements. This method is adopted as it uses the PIV snapshots for calculation making it computationally inexpensive. The instantaneous flow filed from the PIV measurement is considered as the PIV snapshot. The POD is calculated for 2400 PIV snapshots for each of the presented case. The vectors in the PIV shadow region were masked in order to eliminate any discrepancies caused by inaccurate vector fields in the shadow region. In the current study, the POD modes are calculated, not only based on velocity data, but also for the vorticity of the flow field for all the tested configurations for angles of attack \( \alpha = 6^\circ, 8^\circ, 10^\circ \) and \( 12^\circ \) at a chord-based Reynolds number of \( Re_c = 7.5 \times 10^5 \). In the current work, vorticity was based on a second-order least squares fit of the form \( ax^2 + by^2 + cxy + dx + ey + f \) to the data pertaining to each of the velocity components. The latter subsequently allowed a straightforward evaluation of the spatial gradients and hence, vorticity.
At first, the calculated mean velocity and vorticity fields are considered as the zeroth mode of the POD. All the fluctuating flow field components are used for rest of the analysis are arranged in a matrix $U$ as

$$U = [u^1 u^2 \ldots u^N] = 
\begin{bmatrix}
  u^1_{1 \ldots N} & u^2_{1 \ldots N} & \ldots & u^N_{1 \ldots N} \\
  v^1_{1 \ldots N} & v^2_{1 \ldots N} & \ldots & v^N_{1 \ldots N} \\
  w^1_{1 \ldots N} & w^2_{1 \ldots N} & \ldots & w^N_{1 \ldots N}
\end{bmatrix},$$

(5.1)

where $u, v$ and $w$ are the three fluctuating velocity components and $N$ is the number of snapshots.

The autocovariance matrix is created as

$$\tilde{C} = U^T U,$$

(5.2)

and the corresponding eigenvalue problem can be solved by,

$$\tilde{C} A^i = \lambda^i A^i.$$

(5.3)

The solutions of which are arranged by the size of the eigenvalues,

$$\lambda^1 > \lambda^2 > \lambda^3 > \ldots > \lambda^N = 0.$$

(5.4)

The resultant eigenvectors from Eq. 5.3 are used to construct the the POD modes $\Phi^i$,

$$\Phi^i = \frac{\sum_{n=1}^{N} A^i_n u^n}{\left\| \sum_{n=1}^{N} A^i_n u^n \right\|}, \quad i = 1, 2 \ldots N,$$

(5.5)

where $A^i$ is the $n^{th}$ component of the non-dimensional eigenvector corresponding to $\lambda^i$ eigenvalue from Eq. 5.3.

The original snapshots of the flow field are expanded in a series of POD modes with expansion coefficients $a^i$ for each POD mode $i$. The expansion of the fluctuating part of a snapshot $n$ is as follows,

$$u^n = \sum_{n=1}^{N} A a^i_n \Phi^i = \Psi^T a^n,$$

(5.6)
where $\Psi = [\Phi^1 \Phi^2 \Phi^3 \ldots \Phi^N]$. The POD coefficients can be determined by projecting the fluctuating flow fields onto the POD modes,

$$a^n = \Psi^T u^n. \tag{5.7}$$

The energy of a fluctuating flow field in a snapshot for a given POD-mode is proportional to the corresponding eigenvalue. The first mode represents the most energetic structure of the flow and it is usually associated with the large scale flow structures. The first POD mode is the most important in terms of energy as ensured by the arrangement of the eigenvalues and eigenvectors in Eq. 5.4. Therefore, the first few modes are sufficient enough to investigate the dominant flow features [150, 151].

Table 5.2: The number of resolved POD modes of the vorticity that contains 90% of the systems energy for each configuration.

<table>
<thead>
<tr>
<th>$\alpha$ (deg.)</th>
<th>Baseline</th>
<th>H-SCF</th>
<th>SCF</th>
</tr>
</thead>
<tbody>
<tr>
<td>6</td>
<td>177</td>
<td>288</td>
<td>144</td>
</tr>
<tr>
<td>8</td>
<td>210</td>
<td>197</td>
<td>143</td>
</tr>
<tr>
<td>10</td>
<td>195</td>
<td>180</td>
<td>125</td>
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<tr>
<td>12</td>
<td>196</td>
<td>172</td>
<td>109</td>
</tr>
</tbody>
</table>

<table>
<thead>
<tr>
<th>$\alpha$ (deg.)</th>
<th>Baseline</th>
<th>H-SCF</th>
<th>SCF</th>
</tr>
</thead>
<tbody>
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<td>233</td>
<td>221</td>
</tr>
<tr>
<td>8</td>
<td>205</td>
<td>249</td>
<td>235</td>
</tr>
<tr>
<td>10</td>
<td>204</td>
<td>246</td>
<td>238</td>
</tr>
<tr>
<td>12</td>
<td>222</td>
<td>219</td>
<td>236</td>
</tr>
</tbody>
</table>

The number of resolved modes that contains 90% of the system’s total energy for each of the configurations for all the tested four angles of attack are presented in Table 5.2. The normalized eigenvalues for each of the first 12 POD modes within the slat cove region and at the slat wake for angles of attack $\alpha = 6^\circ, 8^\circ, 10^\circ$ and $12^\circ$ for all the tested configurations are presented in Fig. 5.23. The normalization was achieved by dividing each eigenvalue by the sum of all the eigenvalues. The eigenvalues provide an estimation of the coherent energy embedded within each of the vorticity POD modes [152]. For all the tested configurations the results show that a significant amount of energy is contained within the first two modes and the remaining energy is distributed on a wide range of modes portraying an exponential decay, this is indicative of a periodic phenomenon [153].

At the angle of attack $\alpha = 6^\circ$, the first two modes of the Baseline case’s slat cove are clearly dominant. Modifying the slat cove with the filler reduces this dominance, yet both the first two modes remain of equal importance for all the presented cases. For the first mode with the highest
energy, the SCF has eigenvalues lower than that of H-SCF configuration but in the case of the second mode, the H-SCF has lower eigenvalue than that of the SCF at $\alpha = 6^\circ$. The results also show that as the angle of attack is increased the flow becomes more turbulent and hence it requires more modes to be described accurately, which is reflected in the increased importance of the higher mode numbers. This suggests the flow to become less temporally coherent and more turbulent requiring more modes to describe each individual flow field. The eigenvalue mode coefficient for the first two POD modes are shown in Fig. 5.25 at the slat cove region for the angle of attack $\alpha = 6^\circ$. The circular pattern exhibited in the Baseline case clearly indicates the presence of regular vortex shedding. As the angle of attack is increased this regularity in the the eigenvalue mode coefficient indicating regular vortex shedding is no longer present (see Figs. 5.27, 5.29 and 5.31).
Figure 5.23: The normalised eigenvalue distribution of the first 12 POD modes of the vorticity within the slat cove region and at the slat wake for angles of attack $\alpha = 6^\circ$ (a,b), $\alpha = 8^\circ$ (c,d), $\alpha = 10^\circ$ (e,f) and $12^\circ$ (g,h).
5.3. FLOW FIELD ANALYSIS

Figure 5.24: The vorticity component of the first 4 POD eigemodes within the slat cove region for \( \alpha = 6^\circ \) with a free-stream velocity of \( U_{\infty} = 30 \text{ m/s}, R e_c = 7.0 \times 10^5 \).

Figure 5.25: The eigenvalue mode coefficient of the first 2 POD mode within the slat cove region for angles of attack \( \alpha = 6^\circ \).
CHAPTER 5. SLAT COVE FILLER

Figure 5.26: The vorticity component of the first 4 POD eigemodes within the slat cove region for \( \alpha = 8^\circ \) with a free-stream velocity of \( U_\infty = 30 \text{ m/s}, Re_c = 7.0 \times 10^5 \).

Figure 5.27: The eigenvalue mode coefficient of the first 2 POD mode within the slat cove region for angles of attack \( \alpha = 8^\circ \).
5.3. FLOW FIELD ANALYSIS

Figure 5.28: The vorticity component of the first 4 POD eigemodes within the slat cove region for $\alpha = 10^\circ$ with a free-stream velocity of $U_\infty = 30$ m/s, $Re_c = 7.0 \times 10^5$.

Figure 5.29: The eigenvalue mode coefficient of the first 2 POD mode within the slat cove region for angles of attack $\alpha = 10^\circ$. 

161
Figure 5.30: The vorticity component of the first 4 POD eigemodes within the slat cove region for $\alpha = 12^\circ$ with a free-stream velocity of $U_\infty = 30 \text{ m/s}, Re_c = 7.0 \times 10^5$.

Figure 5.31: The eigenvalue mode coefficient of the first 2 POD mode within the slat cove region for angles of attack $\alpha = 12^\circ$. 

162
5.3. FLOW FIELD ANALYSIS

(a) Mode-1 (b) Mode-1 (c) Mode-1
(d) Mode-2 (e) Mode-2 (f) Mode-2
(g) Mode-3 (h) Mode-3 (i) Mode-3
(j) Mode-4 (k) Mode-4 (l) Mode-4

Figure 5.32: The vorticity component of the first 4 POD eigemodes at the slat wake region for \( \alpha = 6^\circ \) with a free-stream velocity of \( U_\infty = 30 \text{ m/s}, Re_c = 7.0 \times 10^5 \).

Figure 5.33: The eigenvalue mode coefficient of the first 2 POD mode within the slat wake region for angles of attack \( \alpha = 6^\circ \).
Figure 5.34: The vorticity component of the first 4 POD eigemodes at the slat wake region for \( \alpha = 8^\circ \) with a free-stream velocity of \( U_\infty = 30 \text{ m/s, } Re_c = 7.0 \times 10^6 \).

Figure 5.35: The eigenvalue mode coefficient of the first 2 POD mode within the slat cove region for angles of attack \( \alpha = 8^\circ \).
Figure 5.36: The vorticity component of the first 4 POD eigemodes at the slat wake region for $\alpha = 10^\circ$ with a free-stream velocity of $U_\infty = 30$ m/s, $Re_c = 7.0 \times 10^5$.

Figure 5.37: The eigenvalue mode coefficient of the first 2 POD mode within the slat cove region for angles of attack $\alpha = 10^\circ$. 

165
CHAPTER 5. SLAT COVE FILLER

Figure 5.38: The vorticity component of the first 4 POD eigemodes at the slat wake region for $\alpha = 12^\circ$ with a free-stream velocity of $U_\infty = 30$ m/s, $Re_c = 7.0 \times 10^5$.

Figure 5.39: The eigenvalue mode coefficient of the first 2 POD mode within the slat cove region for angles of attack $\alpha = 12^\circ$.
The results of the first 4 POD modes of the vorticity fields within the slat cove region are presented for the tested angles of attack $\alpha = 6^\circ$, $8^\circ$, $10^\circ$ and $12^\circ$ in Figs. 5.24, 5.26, 5.28 and 5.30, respectively. From the contour plots, it is evident that the mode 1 and 2 have higher energy compared to mode 3 and 4 for each of the configuration for all the presented angles of attack. For mode 1 at $\alpha = 6^\circ$, the alternating pattern of the large coherent structures for the Baseline indicates the presence of regular vortex shedding along the shear layer, which is absent at mode 1 for both the H-SCF and SCF configuration. This vortex shedding can also be confirmed from the circular pattern observed in the eigenvalue mode coefficient results seen in Fig. 5.25. The Baseline case results clearly show that the first two modes with the higher energy contain the larger coherent structures from the vortex shedding, whereas the mode 3 and 4 with relatively lower energy contains the smaller structure vortices. For H-SCF at $\alpha = 6^\circ$, mode 1 clearly shows that the high energy shear layer does not move toward the slat trailing edge and into the slat gap unlike the Baseline and the SCF. The first two POD modes for the H-SCF and SCF configuration at $\alpha = 6^\circ$ shows that the shear layer itself contains the high energy in the absence of the vortex shedding.

For the Baseline case at angles of attack $\alpha = 8^\circ$, $10^\circ$ and $12^\circ$ at mode 1 alternating pattern in the vorticity contours indicating regular vortex shedding is seen, however, the indication of regular vortex shedding is not seen in the eigenvalue coefficient in Fig. 5.27, 5.29 and 5.31. Nevertheless, the structures of the mode propagation evidently show that the dominant flow feature is the slat cusp shear layer. The energy of the shear layer at mode 1 and 2 can be clearly seen to reduce for the H-SCF and SCF relative to the Baseline for all the presented angles of attack. The H-SCF case at angles of attack $\alpha > 6^\circ$ show vortex shedding with smaller periodic structures with much-reduced energy compared to the Baseline case. The elimination of the sudden pressure difference at the slat cusp by the use of a small slat cove filler (H-SCF) evidently reduces the energy contained within the slat shear layer. For the SCF case, the energy and the periodicity in the vortex shedding process are clearly eliminated due to its interaction with the lower surface of the SCF.

The vorticity contours for the modes 1 to 4 at the slat wake region above the main-element for all the tested configurations at angles of attack $\alpha = 6^\circ$, $8^\circ$, $10^\circ$ and $12^\circ$ are presented in
Figs. 5.32, 5.34, 5.36 and 5.38, respectively. The slat wake region is of interest as the slat shea
layer impinges on the slat trailing edge and mixes into the slat wake. The eigenvalue mode
coefficient at the slat wake for the first two modes are presented in Figs. 5.33, 5.35, 5.37 and
5.39 and it shows no sign of vortex shedding. The results clearly show that for mode 1 and 2 the
Baseline case has higher energy in the slat wake compared to H-SCF and SCF configuration for
all the presented angles of attack. The mode 1 results for all the configuration shows alternating
energy with increased energy from the flow through the slat gap, whereas the flow from the slat
upper side contributes lesser relative to the slat gap. For the presented angles of attack, the
results show larger structures at mode 1 and 2 for the Baseline case but for the SCF case the
structures of the vorticity field are comparatively much smaller at the slat wake. The difference
in vorticity structures for the POD mode 1 and 2 are distinctly seen at angles of attack $\alpha = 10^\circ$
and $12^\circ$ in Figs. 5.36 and 5.38. For $\alpha = 12^\circ$, for all the three configurations the mode 1 and 2
show alternating pattern at the slat trailing edge with the unsteady shear layer from the slat
cusp evidently contributing to the high energy mode created from the slat gap especially for the
Baseline case.

The results show that for mode 1 the energy at the slat wake is lowest for the SCF compared
to the Baseline and H-SCF for all the presented angles of attack. The results from the POD has
shown that the unsteady vortex shedding in the shear layer is the dominant feature of the slat
cove flow as it is predominant in mode 1 and 2. The results also show that the application of
the slat cove fillers changes the path of the unsteady vortex shedding in the case of low angles
of attack and reduces the energy of the unsteady vortex shedding in the case of high angles of
attack.
5.4 Aeroacoustic results

High-lift devices are one of the dominant components of the airframe noise. The slat noise is much of interest to researchers due to its complex tonal and broadband noise generation mechanisms [125–132]. The noise generated by high-lift devices largely varies with respect to the geometry in terms of tonal noise frequency range and tonal peak location. Emphasizing on the similarities in the tonal noise seen in rectangular cavities, the slat cove region can be considered as an open cavity. Previous studies [120, 124, 128, 140, 141] have shown that the feedback mechanism between the unsteady vortices emanating from the slat cusp and the trailing edge acts as a resonator. The noise generated in cavities are due to the flow induced cavity oscillations and the multiple acoustic scales arise due to the vortical disturbances driving the oscillations. Rossiter [142] showed that the discrete frequencies in rectangular cavities are due to the oscillations influenced by the acoustic feedback from the shear layer impingement region. The study also proposed an empirical formula to predict the tonal frequencies in rectangular cavities. Kolb et al. [120] applied an improved version of the Rossiter equation [142, 143] on the slat noise mechanism and showed that the analytical Rossiter frequencies agreed well with the measured experimental tonal peaks. Terracol et al. [124] further simplified the Rossiter equations [142, 143] and matched the slat cove tonal frequencies for the FNG Airbus geometry F16. Pascioni and Cattafesta [128] applied the simplified equation by Terracol et al. [124] on a 30P30N geometry and showed that the tones and their harmonics can be accurately predicted. These studies have shown the robustness of the Rossiter equations to predict the tonal noise generated by a slat.

The discrete tonal frequencies due to the flow interaction with the slat, based on Terracol’s study [124], can be predicted from

\[ f_n = \frac{U_\infty}{L_a} \left( \frac{1}{M + \eta_l / \kappa_v} \right). \]  \hspace{1cm} (5.8)

A simplified schematic of the parameters used in Eq. 5.8 by Terracol et al. [124] are shown in Fig. 5.40 and the parameters used are listed in Table 5.3. The equation is found to be highly sensitive to the shear layer \((L_v)\) and acoustic path length \((L_a)\). The flow field, i.e. local flow velocity, data required for this prediction model were acquired from the PIV measurements by from the authors' previous studies [100, 101].
Figure 5.40: Simplified schematic of the tonal frequency prediction model by Terracol et al. [124].

Table 5.3: Parameters used for tonal peak frequency prediction.

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<tr>
<th>Parameter</th>
<th>Value</th>
<th>Units</th>
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<td>18°</td>
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170
5.4. AEROACOUSTIC RESULTS

5.4.1 Far-field spectral levels

Far-field noise measurements were carried out in order to assess the noise generated from the Baseline, H-SCF, and SCF configurations. The sound pressure level measured from a far-field microphone at 90° above the slat trailing edge for the angle of attack $\alpha = 14^\circ$ and $18^\circ$ at the free-stream velocity of $U_\infty = 30$ m/s is shown in Fig. 5.41. The results show that the background noise of the aeroacoustic facility is well below the high-lift airfoil noise levels. While the results for the baseline 30P30N airfoil show the discrete narrowband peaks, typical of the noise signature from such high-lift devices, the far-field noise results of the H-SCF and SCF configuration do not demonstrate such tonal behavior. The SCF configuration at the angle of attack $\alpha = 18^\circ$ clearly shows a reduction in the noise levels over the entire frequency range. Significant noise reduction of up to 8 dB at low to mid-frequency range ($St_s < 5$) are observed for the SCF configuration compared to the Baseline and H-SCF cases. The generation of the tonal peaks and the mechanisms driving them will be discussed in detail in the following sections.

Figure 5.41: Far-field noise spectra for microphone at 90° and 1.75 m above the slat trailing edge for Baseline —, H-SCF – – –, SCF – · – · and Background noise · · · · ·. The resonance modes are listed in Table 5.5.

The Far-field directivity plots from the pressure side elevation angles for the Baseline, H-SCF, and SCF configurations at different slat based Strouhal number ($St_s = f \cdot c_s / U_\infty$) are presented in Fig. 5.42. The results are plotted for the angles of attack $\alpha = 14^\circ$ and $18^\circ$, for a free-stream velocity of $U_\infty = 30$ m/s, corresponding to a chord-based Reynolds number of $Re_c = 7.0 \times 10^5$. The results are shown for the selective Strouhal numbers based on the narrowband peaks observed at the far-field spectral levels in Fig. 5.41. At first glance, it is evident that the application of
the H-SCF and SCF does not influence the overall directivity shape for the presented range of Strouhal numbers compared to the Baseline case. For $\alpha = 14^\circ$, at $St_1$, the acoustic amplitude of the directivity results remains unchanged for the H-SCF and SCF configurations compared to the Baseline case but a reduction of up to 10 dB is observed at $\alpha = 18^\circ$ for the SCF configuration over the whole polar angles. The reduction in the spectral levels for the cove filler configurations for the modes $St_{2&4}$ are substantial, with a reduction of up to 20 dB at both the presented angles of attack. The noise level results show a significant reduction for the H-SCF and SCF cases relative to that of the Baseline case at $\alpha = 18^\circ$, at all frequencies.

The overall sound pressure levels (OASPL) for the different configurations are shown in Fig. 5.43. The overall sound pressure level was resolved for a frequency range from $f = 100$ Hz to 32 kHz. The results show that the applications of the H-SCF and SCF reduces the overall noise level by about 2-3 dB at $\alpha = 14^\circ$ and a significant reduction of up to 10 dB at the higher angles of attack ($\alpha = 18^\circ$) compared to that of the Baseline case, particularly at locations upstream of the slat trailing edge.
5.4. AEROACOUSTIC RESULTS

Figure 5.42: Directivity for the different configurations at different slat based Strouhal number, for Baseline □ H-SCF △ and SCF ○.

Figure 5.43: Overall sound pressure level calculated from the far-field microphones, for Baseline □ H-SCF △ and SCF ○.
5.4.2 Near-field spectral levels

Near-field unsteady pressure measurements were performed to gain an insight into the noise generation mechanism of the slat. The unsteady surface pressure measurements were acquired at various spanwise locations on the surface of the main-element of the high-lift airfoil. The measurements were carried out using 5 surface-mounted pressure transducers, which are detailed in Table 5.4. The data were acquired for 16 s and sampled at 40 kHz. Even though the measurements were carried out for the angles of attack $\alpha = 12^\circ, 14^\circ, 16^\circ$ and $18^\circ$, the results here are presented only for the angles of attack $\alpha = 14^\circ$ and $18^\circ$, for the purpose of brevity. The sound pressure levels are presented in terms of the slat based Strouhal number ($St_s = f \times c_s/U_\infty$).

From the aeroacoustic study carried out by Murayama et al. [70], it was seen that the surface mounted pressure transducers on the main-element can be used to accurately predict the slat tones and can also provide some useful information about the broadband energy content of the flow structures within the slat cove. The results from the unsteady surface pressure measurements from the transducer M1 at the leading edge of the main-element are shown in Fig. 5.44. The tonal characteristics of the wall pressure spectra indicate the presence of cavity oscillations.

The wall pressure fluctuation spectra results for the Baseline in Fig. 5.44 show multiple distinct narrowband peaks for all the tested angles of attack with varying intensities, characterizing cavity oscillations. Some of the tonal peaks were also observed in the far-field noise measurements. The tonal peaks are numbered in Fig. 5.44 and are listed in Table 5.5.

<table>
<thead>
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<th>$z$ (mm)</th>
</tr>
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</tr>
<tr>
<td></td>
<td>M2</td>
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</tr>
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<td></td>
<td>M3</td>
<td>22.414</td>
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<td></td>
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<td>22.414</td>
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</tr>
<tr>
<td></td>
<td>M5</td>
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<td>319.6</td>
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</table>

As discussed earlier, the tonal peaks observed are due to the flow-acoustic coupling which leads to resonance as all of them could be accurately predicted by the simplified Rossiter mode equation (Eq. 5.8). At the angle of attack $\alpha = 14^\circ$, the pressure spectra results show three distinct
5.4. AEROACOUSTIC RESULTS

Figure 5.44: Near-field noise spectra for the surface transducer M1 \(x = 22.414\) mm for Baseline — , H-SCF —— — —, SCF ——— and Background ..... The associated modes \(S_t\) are listed in Table 5.5.

Table 5.5: The narrow-band frequencies observed for the Baseline case in the near-field and far-field measurements at angles of attack \(\alpha = 14^\circ\) and \(18^\circ\) and the labels in Figs. 5.41 and 5.44 are detailed.

<table>
<thead>
<tr>
<th>(\alpha = 14^\circ)</th>
<th>(\alpha = 18^\circ)</th>
</tr>
</thead>
<tbody>
<tr>
<td>(\text{No.} )</td>
<td>(S_t)</td>
</tr>
<tr>
<td>1</td>
<td>(S_t_1)</td>
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</table>

peaks \(S_t_{1,2,3} = 0.885, 1.596\) and 3.203. These are the first three Rossiter modes and the fourth mode, \(S_t_4 = 3.203\) was also observed but with low intensity. The first mode \((S_t_1)\) predicted by the analytical formula is not distinctly seen in the experiments as it lies within the spectral hump seen at low-frequency \(0.5 < S_t < 1\). For the angle of attack \(\alpha = 18^\circ\), at the first glance, the results of the Baseline case appear chaotic with multiple peaks but the observed discrete tonal peaks are the first 14 Rossiter modes, as predicted by Eq. 5.8 (see Table 5.5). The multiple peaks seen here also depict harmonic behavior and possess an algebraic relationship amongst themselves. A cascading effect in the modes is seen through the entire mid to high frequency
range i.e. $St_4 = 2St_2$, $St_5 = St_3 + St_2$, $St_6 = 3St_2$ and $St_7 = St_5 + St_2$. The presence of such an algebraic relationship between the modes suggests the occurrence of nonlinear quadratic interaction between the Rossiter modes, which will be discussed in detail in Section 5.4.5. The magnitude of the tonal peaks also varies relatively for the Baseline case, especially with the peaks of the even modes having higher magnitude compared to the consecutive odd modes. It is notable that the first Rossiter mode is concealed within the spectral hump $0.5 < St_s < 1$. The results are indicative that the source of the spectral hump might not be solely due to the Rossiter modes. The Rossiter modes are due to the flow-induced oscillations and acoustic feedback mechanism since the modes 2-14 observed have a distinct narrowband peak. The source of this spectral hump was suggested to be due to the low-frequency cove oscillation, as evidenced by phase-locked PIV by Pascioni and Cattafesta [128]. The broadband nature of the noise could be attributed to the flow impingement in the slat and main-element. It could also be a consequence of the distorted shear layer exiting through the slat gap. The trailing edge scattering would also add to the broadband component.

The results of the H-SCF and SCF configurations for both the presented angles of attack do not show any indication of the above discussed tonal peaks. The wall pressure spectra at the angle of attack $\alpha = 14^\circ$, for the H-SCF, show an increase of about 5 dB at low-frequency range $St_s > 0.8$ compared to the Baseline and SCF configurations. The presence of the semi-cavity in the case of H-SCF gives rise to two spectral humps at $St_s = 0.6$ and 2 at the angle of attack $\alpha = 18^\circ$. Even though the H-SCF configuration has a semi-cavity the tonal peaks are not observed in the near and far-field measurements. The two spectral humps at $St_s = 0.6$ and 2 at the angle of attack $\alpha = 18^\circ$ might be due to the semi-cavity as they are not observed in the SCF case. The SCF configuration shows a reduction at low-frequency range for the angle of attack $\alpha = 14^\circ$ and increased levels for the same range of frequencies at the angle of attack $\alpha = 18^\circ$. The tonal peaks are also absent for the SCF configuration in the near-field measurements. The increased spectral levels seen in the near-field surface pressure measurements for the SCF are absent in the far-field measurements in Fig.5.41, which implies that the increased spectra in the near-field are due to the non-propagating hydrodynamic energy field within the slat and main-element. The results for the H-SCF case show a spectral hump at $St_1$, the same as that of the Baseline irrespective
of the reduced slat cove size. This again confirms that the $St_1$ and the broadband hump in this region might be different in nature compared to the dominant even-numbered modes seen in the Baseline case. Some of the discrete narrowband peaks seen at high-frequency were not seen in the far-field measurements as they fall below the broadband content of the noise radiated to the far-field observer.

### 5.4.3 Spanwise coherence

The spatial coherence scales are used to determine the extent of the acoustic wave interference and this can help us to better understand the hydrodynamic field and radiated noise. The spanwise coherence calculates the phase correlation between two different spanwise pressure transducers averaged over time. The spanwise coherence between the surface pressure transducers is obtained using the following equation,

$$
\gamma_{p_i,p_j}^2(f, \Delta z) = \frac{|\Phi_{p_i,p_j}(f)|^2}{\Phi_{p_i,p_i}(f)\Phi_{p_j,p_j}(f)} \text{ for } p_i = M1 \text{ and } p_j = M1, M2, \ldots, M5, \quad (5.9)
$$

where M1-M5 are the unsteady pressure transducers mounted on the leading-edge of the main-element and their locations are provided in Table 5.4. The spanwise coherence for the three configurations between the reference transducer M1 and the other spanwise located surface transducers M2, M3 and M5 are presented in Fig. 5.45. For the smallest lateral spacing $\Delta z/c_s = 0.07$ (between M1 and M2) the results show high coherence for all the cases at both the angles of attack for all the frequencies. The coherence for the Baseline and H-SCF (Fig. 5.45 (c)) at the spanwise spacing $\Delta z/c_s = 0.22$ (between M1 and M3) shows coherence reduction in the high-frequency range but high coherence levels for the tonal peaks. At the largest separation distance, $\Delta z/c_s = 0.81$ (between M1 and M5) the coherence for all the frequencies is almost zero, except for the tonal peaks observed in the surface pressure spectra for both the angles of attack. The results at the angle of attack $\alpha = 18^\circ$ show that except for the tonal peaks, the three configurations have similar spanwise coherence for the broadband aspect indicating that they all have similar three-dimensional flow structures. The most notable aspect of the coherence results is that the spectral hump at $St_1$ shows a high level of coherence for all the presented separation distances for all the configurations. It is interesting that high level of coherence for the broadband spectral
hump at low frequency is seen for both the cove filler H-SCF and SCF (see Fig. 5.45 (c)). Even for the largest $\Delta z/c_s = 0.81$, a remnant of the phase correlation of the acoustic waves are seen at about $St_1$, with values of up to $\gamma^2_{p_i,p_j} \approx 0.2$ for the H-SCF and SCF cases. This is indicative of the fact that the source of the spectral hump is not quite as that of the Rossetier modes.

In order to estimate the noise generated through the interaction of advecting pressure in the vicinity of the airfoil, the spanwise correlation length of the wall pressure fluctuations were calculated. The spanwise correlation length of the flow structure and the local hydrodynamic field can be calculated using the following equation,

$$\Lambda_{\gamma}(f) = \int_{0}^{\infty} \gamma_{p_i,p_j}(f, \Delta z) d\Delta z.$$  \hspace{1cm} (5.10)

The length scale ($\Lambda_{\gamma}$) as a function of frequency is calculated using the spanwise coherence results ($\gamma$) between the surface pressure transducers and are presented in Fig. 5.46. For the angle of attack $\alpha = 14^\circ$ the results show two distinct spectral humps for the length scales for all the configurations. The Baseline and the H-SCF results follow the same trend with similar length scales but with the absence of the tonal peaks for the H-SCF configuration. The SCF case shows slightly increased length scales relative to the Baseline and the H-SCF. At the angle of attack, $\alpha = 18^\circ$, the three configurations possess the same broadband trend at mid to high-frequencies ($St_s > 1$). The length scales show the spectral hump for only the Baseline case at low-frequencies ($St_s < 1$). This shows that for the H-SCF airfoil even though the size of the cavity is reduced and the acoustic feedback mechanism that gives rise to the tonal behavior has been eliminated, the spanwise correlation length remains the same as that of the Baseline airfoil.

The auto-correlation were calculated using the unsteady surface pressure, it is defined as,

$$R_{p_i,p_i}(\tau) = \frac{p_i(t+\tau)p_i(t)}{p_{i,RMS}^2},$$  \hspace{1cm} (5.11)

where $p_i$ is the surface pressure, $p_{i,RMS}$ is the surface pressure root mean squared, $\tau$ is the time delay and the time average is represented by the overbar. The results of the auto-correlation of the surface pressure at the transducer location M1 at the angle of attack $\alpha = 14^\circ$ and $18^\circ$ are presented in Fig. 5.47, as a function of the normalized time delay $\tau^* = \tau U_\infty/c_s$. For the Baseline, at the angle of attack $\alpha = 14^\circ$ the results exhibit a fast decaying periodic behavior. At the angle of
5.4. AEROACOUSTIC RESULTS

Figure 5.45: Coherence between the reference transducer M1 and the other spanwise transducers M2-M5 (see Table 5.4), for Baseline —, H-SCF —— and SCF ——.

At the angle of attack $\alpha = 18^\circ$ the results exhibit a slow decaying periodic behavior with a Gaussian shape with a low decay rate, which is suggestive of a strong vortex shedding. At the angle of attack $\alpha = 18^\circ$, the distance between the two peaks in $R_{p_i,p_j}(\tau)$ for the Baseline case corresponds to the vortex
Figure 5.46: Spanwise coherence length scales based on the unsteady surface pressure measurement for Baseline ——, H-SCF ——– and SCF ——.

sheddng frequency ($\tau_{vs}$). The calculated time delay $\tau_{vs} = 0.5711$ corresponds to $St_{vs} = 1.75 \approx St_2$, which is the primary peak seen in power spectral plots (see Fig. 5.44) with the highest magnitude. The vortex shedding for the angle of attack $\alpha = 14^\circ$ is not seen distinctly compared to $\alpha = 18^\circ$. This is due to the lower energy of the vortex shedding frequency (see $St_2$ in Fig. 5.44) at angle of attack $\alpha = 14^\circ$ (10 dB) compared to $\alpha = 18^\circ$ (25 dB). The results of the H-SCF and SCF cases show a very weak periodic shape that decays instantaneously, indicating the absence of a strong vortex shedding for both the presented angles of attack. Even though the H-SCF configuration has half a cavity slat, its behavior is more similar to that of the SCF than the Baseline case.

To further understand the intensity of the noise radiated to the far-field and isolate the non-propagating hydrodynamic field, coherence between the surface pressure transducer $M_1$ and the far-field microphone placed at $90^\circ$ above the slat trailing-edge were carried out. The coherence was calculated using the following equation,

$$\gamma_{p,p_j}^2(f) = \frac{|\Phi_{p,p_j}(f)|^2}{\Phi_{p,p}(f)\Phi_{p_j,p}(f)} \quad \text{for} \quad p_i = M_1 \text{ and } p_j = M_90^\circ,$$

where $M_1$ is the reference surface pressure transducer and $M_90^\circ$ is the far-field microphone at $90^\circ$ above the slat trailing edge. The near- to the far-field coherence results are presented in Fig. 5.48. The results show high coherence at all tonal peaks that arise due to the Rossiter modes. The H-SCF and SCF cases show low coherence over the entire frequency range. However, a noticeable spectral hump at $1 < St_s < 4$ with maximum coherence values up to $\gamma_{p,p_90^\circ}^2 = 0.6$ for the H-SCF and SCF cases are seen. This increased coherence shows a feature that was not seen in both the
5.4. AEROACOUSTIC RESULTS

Figure 5.47: Auto-correlation of the surface pressure fluctuations at the near-field transducer location M1 for Baseline —, H-SCF – – – and SCF – – –.

near-field and far-field sound pressure level measurements.

Figure 5.48: The coherence between the reference near-field surface pressure transducer M1 and the far-field microphone 90° above the trailing edge for Baseline —, H-SCF – – – and SCF – – –.
5.4.4 Continuous Wavelet Transform

The physical mechanism of the multiple distinct tones generated by the high-lift airfoils are suggested to be due to an amplitude modulation mechanism and was successfully shown recently using continuous wavelet transform by Li et al. [130–132]. The wavelet transform technique adds time resolution to the frequency, enabling us to see the temporal characteristics of the signals and their associated frequency. The continuous wavelet transform (CWT) [154] breaks down a given signal into a time-scale space and its squared magnitude. This technique overcomes the shortcomings of the Fourier analysis by adding the time resolution. The continuous wavelet transform (CWT) method employed in the current study is defined as,

\[ W_x(a, \tau) = \int_{-\infty}^{+\infty} x(t) \psi_{a,\tau}^*(t) dt, \]  

(5.13)

where \( W_x(a, \tau) \) is the continuous wavelet transform of function \( x(t) \), \( a > 0 \) is the scale variable, \( \tau \) is the time delay, \( \psi_{a,\tau}(t) \) is the wavelet function, and the symbol \( * \) denotes the complex conjugate.

The continuous translation and dilation of the mother wavelet \( \psi(t) \) are obtained by

\[ \psi_{a,\tau}(t) = \frac{1}{\sqrt{a}} \psi\left(\frac{t-\tau}{a}\right). \]  

(5.14)

The Morlet wavelet [155] was chosen as the wavelet function for the analysis, given by,

\[ \psi(t) = \Pi^{-1/4} e^{i\omega_0 t} e^{-\left(t^2/2\right)}, \]  

(5.15)

where \( \omega_0 \) is the non-dimensional frequency and is chosen to be 6.0 to satisfy the wavelet admissibility condition [156].

The contour plots of the wavelet coefficient magnitude for the pressure signal collected by the near-field pressure transducer M1 for the three tested configurations at the angle of attack \( \alpha = 14^\circ \) and \( 18^\circ \) are presented in Fig. 5.49. Even though the measurements were carried out for 16 seconds, the results are presented only for 0.6 seconds with a higher temporal resolution for better visualization. The frequency variation with time is evidently visible in the presented results for all the configurations. For the Baseline airfoil, the Rossiter modes are distinctly evident. In Fig. 5.49 (a) and (b), the high temporal resolution of the results clearly show that the
modes are amplitudes modulated in time. However, only the first four modes are clearly seen with the mode two ($St_2$) possessing the highest level of energy. The results for the angle of attack $\alpha = 14^\circ$ distinctly display this behavior, whereas at the angle of attack $18^\circ$ the results have higher amplitude with increased occurrences, making it harder to spot the mode amplitude modulation. The absence of the Rossetier modes for the H-SCF and SCF configuration are also clearly seen in the results shown in Fig. 5.49. The behavior of the modes switching in time was previously shown by Kergerise et al. [157] for cavity flow and by Li et al. [131, 132] for a 30P30N high-lift airfoil. These studies showed that at a given time period the cavity resonates at a given mode or modes but the modes modulate and interact amongst themselves [131, 132, 157]. This is expected as the mode number of the cavity oscillation is related to the spacing between the vortices [157].

In order to further understand the multiple tone generation mechanism seen in the current study, power spectral density of the time signal and the wavelet coefficient magnitude ($|W^2|$) at selected frequencies $St_{2-10}$ (see Table 5.5) are presented in Fig. 5.50. The results in Fig. 5.50 shows the amplitude of the wavelet coefficient magnitude ($E(St_s)$) in terms of slat chord-based Stouhal number ($St_s$). The presented results show that at the angle of attack $\alpha = 14^\circ$, for modes 3 and 4, the amplitude is modulated by a frequency $\Delta St_{3,4} = 1.603 (\approx St_2)$ and at the angle of attack $18^\circ$, for mode 3, 4 and 5, the amplitude is modulated by a frequency $\Delta St_{3,4,5} = 1.729 (\approx St_2)$. The amplitude modulation frequency found for various Morlet scales ($a$), in Eq. 5.15, are the primary acoustic energy concentrated at mode 2, which is also referred to as $St_2$ (see Table 5.5). The modulation phenomenon states that the secondary frequencies can be predicted by the following [131, 132],

$$St_n = St_{n,\text{max}} \pm a \cdot \Delta St_s,$$

(5.16)

where $a$ is the positive integer, $St_{n,\text{max}}$ is the frequency of the primary tone, $\Delta St_s$ is the modulation frequency. By applying the calculated modulation frequency in Eq. 5.16, for the angle of attack $\alpha = 18^\circ$, several secondary frequencies can be predicted as $St_4 = St_2 + \Delta St_s = 3.462$, $St_5 = St_3 + \Delta St_s = 4.210$ and $St_6 = St_4 + \Delta St_s = 5.197$, etc. This equation can also be used to predict all the weaker tones observed in Fig. 5.44. The results show that the relationship between the tonal peaks are not only confined to the amplitude modulation phenomenon but they also
have a harmonic and a non-linear relationship between themselves as shown in Table 5.5 and will be further discussed in the following sections.

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</table>

Figure 5.49: The contours of the wavelet coefficient magnitude ($|W_x|$) for the near-field pressure transducer M1 calculated using Morlet wavelet function.
Figure 5.50: Power spectral density of time signal and wavelet coefficient at selected resonance frequencies (St\(_n\)) from Table 5.5 for the Baseline case.
5.4.5 Higher order spectral analysis

The turbulent cascade phenomena of fluids can be well characterized by identifying the non-linear exchange of energy from one frequency to another. In order to identify and interpret the non-linear energy transfer between the frequencies, the higher order spectral analysis, namely auto-bispectrum was carried out. This method was successfully used to show the non-linear interactions between the Rossiter modes in cavity flows [157]. In a recent study, Pascioni and Cattafesta [128] showed the mode interaction phenomenon in slat cavity flow. It is well known that the phase information is suppressed by the power spectral density, making higher order spectral methods as an essential tool to quantify the quadratic phase coupling between frequencies as they retain the phase information. If several tonal peaks are present in the power spectral density the number of independent sources cannot be identified but auto-bicoherence allows one to discover if a tonal peak has been created by the quadratic nonlinear interaction. The auto-bispectrum \( B_{ppp}(f_i, f_j) \) is used to determine the quadratic coupling and algebraic sum between the frequencies \( f_i \) and \( f_j \) and it is calculated from,

\[
B_{ppp}(f_i, f_j) = \lim_{T \to \infty} \frac{1}{T} \text{EV}[P(f_i)P(f_j)P^*(f_i + f_j)],
\]

(5.17)

where \( P(f) \) is the Fourier Transform of \( p(t) \), \( T \) is the time length, \( \text{EV}[\cdot] \) is the expected value and \( ^* \) denotes the complex conjugate. The auto-bispectrum can also be normalized by the corresponding power spectrum elements, known as the auto-bicoherence, as follows,

\[
b_{ppp}^2(f_i, f_j) = \frac{|B_{ppp}(f_i, f_j)|^2}{\Phi_{pp}(f_i + f_j)\Phi_{pp}(f_i)\Phi_{pp}(f_j)}.
\]

(5.18)

The auto-bicoherence between the three waves measures their phase coupling. If the frequencies of the wave at \( f_i, f_j \) and \( f_i + f_j \) are characterised by statistically independent phase relationship, then \( b_{ppp}^2 = 0 \). If the frequency component at \( f_i + f_j \) exhibits any phase relationship with \( f_i \) and \( f_j \), then the corresponding auto-bicoherence will have a value, as \( 0 < b_{ppp}^2 < 1 \). If the waves are perfectly quadratically coupled, then \( b_{ppp}^2 = 1 \).

The contour plots of the auto-bicoherence for the unsteady surface pressure transducer signal at M1 at the angle of attack \( \alpha = 18^\circ \) for all the three configurations are presented in Fig. 5.51. The sum of the frequencies is shown only up to the region of interest (\( St_{12}, St_{12} \)). For the Baseline
5.4. AEROACOUSTIC RESULTS

In Fig. 5.51a, it is evident from the results that the multiple peaks have quadratic coupled modes. The slat cavity modes self interact \((St_1, St_1), (St_2, St_2), (St_3, St_3)\), etc. and also generate harmonics. The results clearly show that the even modes \((St_{2,4,...12})\) have a stronger bicoherence compared to the odd modes \((St_{1,3,...11})\). As mentioned above, if a tonal peak is created by non-linear interaction, the bicoherence value would be \(b^2_{ppp} = 1\). For the frequencies \(St_4, St_6\), the bicoherence value is \(b^2_{ppp} > 0.85\), indicating that these harmonics \(St_4 = 2St_2\) and \(St_6 = 3St_2\) are possibly generated by quadratic coupling. Moreover, a large degree of phase coupling is seen for all the even modes. To further analyze, let us first consider only the odd modes. The results show that there is an interaction between \((St_1, St_1)\) and a mild interaction between \((St_1, St_3)\), but then no phase coupling for \(St_1\) with any other mode. When considering the third mode \(St_3\), it shows coupling only with \(St_4\) and \(St_6\). The only other odd mode to show phase coupling behavior is \((St_5, St_6)\). Therefore, it is clear that the observed odd modes are not in phase with themselves but are occasionally phase coupled with \(St_4\) and \(St_6\). The even modes show a very large degree of phase coupling with most of the observed modes. For the mode \(St_2\), results show phase coupling with all the other modes, including the odd modes, except for the \(St_3\) and \(St_4\) modes. All the other even modes \((St_{2,4,...12})\) show phase coupling with all the other odd and even modes to some degree, but their degree of phase coupling with the even modes are much higher. The results show a high level of quadratic coupling with some of the modes and no coupling between some modes. This shows that the modes reinforce each other in a way at times or exist on their own in some instances. The strong self-interaction of the first mode \(St_1\) with no cross interaction yet again suggests that the first mode might be of a different nature.

The bicoherence results for the H-SCF and SCF configurations are presented in Figs. 5.51b and 5.51c. The results for the H-SCF and SCF configuration show self-interaction of the broadband hump \(0.5 < St_s < 4\) observed in both the cases. This broadband hump was the most dominant feature seen in the near-field unsteady surface pressure results and spanwise coherence results in Fig. 5.44 and 5.45, respectively for the H-SCF and SCF configurations. Even though the bicoherence of the broadband hump is not higher than \(b^2_{ppp} \approx 0.05\), they are still statistically significant due to a large number of averages used for the bicoherence calculations. The results show that the self-interaction occurs at \(St_2\) for both the H-SCF and SCF configurations. The
results here show strong self-interaction of the tones for the Baseline case. The use of the H-SCF and SCF configuration does not only eliminate the tone but also the constructive self-interaction that arise from it. The results are not presented for the angle of attack $\alpha = 14^\circ$ as they showed insignificant self and cross coupling between the tones due to their weaker tones and weaker vortex shedding as seen in Figs. 5.44 and 5.47.

Figure 5.51: The auto-bicoherence contour for the near-field pressure transducer M1 on the main-element for the angle of attack $\alpha = 18^\circ$ labelled with the associated modes ($St_n$) for the Baseline case detailed in Table 5.5.
5.4.6 Persistence spectrum

The interference between two acoustic waves can be constructive or destructive depending on their phase difference. The phase coupling and interference of the signals can be visualized using the persistence spectrum. To further understand the nature of the observed peaks in the present study, the unsteady surface transducer signals from the near-field pressure transducer M1 on the main-element is used to plot the persistence spectrum. The persistence spectrum is a histogram in power-frequency space that shows the percentage of the time that a given frequency is present in a given signal. The time percentage shown as density contours has a higher value in the results if a particular frequency persists in a signal for a longer period of time [158]. The persistence spectrum was calculated for the entire measured time signal of 16 seconds. The short-time Fourier transform for the persistence spectrum was carried out for a time resolution of 0.04 seconds and a frequency resolution of $St_s = 0.45$. The results for all the three tested configurations at the angle of attack $\alpha = 18^\circ$ are presented in Fig. 5.52. The results for the Baseline configuration clearly show that the $St_2 = 1.733$ i.e., the vortex shedding frequency, holds the primary acoustic energy as it is present through the entire time period. The harmonics of the second mode $St_{4,6,8}$ hold the next highest energy over the time period. All the odd modes, which do not have any phase relation (see Fig. 5.51) with the even modes clearly have lesser magnitude and their energy is distributed over time. These results for the Baseline are yet again suggestive that the odd modes might have a different source compared to the even modes. The results for the H-SCF and SCF configurations clearly show that their noise is of broadband nature and is spread over the entire time period. The spectral hump at $St_1$ is not dominated over time, rather the energy of the broadband spectra is evenly distributed in time over the entire frequency range.
Figure 5.52: The persistence spectrum contour for the near-field pressure transducer M1 on the main-element at angle of attack $\alpha = 18^\circ$.

## 5.5 Conclusions

The aerodynamic and aeroacoustic characteristics of 30P30N airfoil with and without slat cove fillers were investigated. As part of the noise reduction study of the MDA 30P30N airfoil, a half slat cove filler (H-SCF) and a slat cove-filler (SCF) configuration along with a Baseline configuration were considered. Results have shown that the aerodynamic lift and drag measurements exhibit an insignificant difference between the tested configurations. The H-SCF exhibits the best aerodynamic performance relative to the Baseline and the SCF configuration in terms of the lift-to-drag ratio and the drag-polar plots. The coefficient of pressure distribution results show that the application of the slat cove fillers decreased the suction peak by up to 15% over the main-element of the airfoil, which does not appear to influence the aerodynamic performance. The flow field contours showed that the unsteady vortex within the slat cove region is eliminated with the application of the H-SCF and SCF. The contours normal Reynolds stress components showed high magnitude in the slat shear layer for all the three configurations. The use of slat cove
fillers do not influence the turbulence level within the slat cove or slat shear layer. The mean velocity profiles of the boundary layer over the main-element show that the slat wake deficit for both the H-SCF and SCF is much lower than that of the Baseline at all the measurement stations over the airfoil. The slat wake profiles in the near-wake region showed insignificant difference in the mean wake velocity and turbulence kinetic energy between the airfoil configurations at high angles of attack, which also corresponds to the insignificant aerodynamic changes observed between the tested configurations.

The POD results showed large coherent structures for high energy POD modes, which arises from that slat cusp of the Baseline case and this large structures are broken down in to smaller structures by the application of the slat cove fillers. Moreover the POD modes also showed that the energy of the vortex shedding is suppressed with H-SCF and the vortex shedding is almost eliminated in the case of the SCF. The near-field surface pressure measurements show increased noise at low-frequency for the H-SCF and SCF configurations, which is due to the non-propagating hydrodynamic of the flow field within the slat and main-element as this noise increase is not observed in the far-field measurements. The results also clearly show that the far-field noise measurements and the overall sound pressure level can be significantly reduced by up to 5 – 9 dB with the application of the slat cove fillers. The lateral coherence studies have shown that a high level of coherence is present for all the configurations, particularly with a distinct broadband spectral humps at low-frequency for cases with the slat cove fillers. The contour plots of the wavelet coefficient show that the Rossiter modes for the Baseline case are amplitudes modulated in time, however, these modes are absent for the configuration with slat cove fillers. The results of the higher spectral order analysis show that the Rossiter modes that arise from the slat cavity of the Baseline case display quadratic interaction amongst themselves. This experimental study shows that there is a need for more fundamental research on the low-frequency broadband hump that arises in the high-lift device slat noise. This chapter also shows that the application of the slat cove fillers eliminate the Rossiter modes generated by the slat cavity and reduces broadband noise without compromising the aerodynamic performance of the high-lift device.
CONCLUSIONS AND FUTURE WORK

This chapter provides a brief summary of the main conclusions from the present study. The aim of the present study was to investigate the aerodynamic and aeroacoustic performance of morphing technologies on different types of airfoils. The use of morphing structures on aircraft has been sought after for over half-century. The advent of high strength pliable composite materials has made this a possible reality in recent years. Even though the material and structural aspect of the morphing technologies were studied intensely in the past decade, their aerodynamic and aeroacoustic performance were not investigated in detail. This study considers the application of morphed structures on the trailing edge of a simple symmetric NACA 0012 airfoil and morphed slat cove filler on a slat of MDA 30P30N high-lift airfoil. These two airfoil configurations cover a wide range of possible applications on an aircraft wing, as well as in a wind turbine. The NACA 0012 airfoil was studied with several morphed flap with varying surface profile camber in comparison with a hinged flap configuration. The 30P30N high-lift airfoil was investigated in order to further the understanding to suppress slat noise by the use of slat cove fillers.

In order to investigate the aerodynamic and aeroacoustic behavior of the NACA 0012 and MDA 30P30N airfoil configurations, several experimental and computational studies were carried out. The NACA 0012 airfoil was tested for two configurations Hinged Flap (HF) and Morphed Flap (MF) airfoil. The 30P30N high-lift airfoil was tested for three configurations the Baseline,
Half Slat Cove Filler (H-SCF) and Slat Cove Filler (SCF). The aerodynamic studies were carried using lift and drag, and surface pressure distribution measurements. The aerodynamic force measurements are further analysed with the help of surface flow visualization and Particle Image Velocimetry measurements. The unsteady flow field and aeroacoustic characteristics of the NACA 0012 airfoil was studied computationally with the help of Large Eddy Simulation. The aeroacoustic and unsteady flow characteristics of the MDA 30P30N airfoil were investigated using far-field microphones and unsteady surface pressure transducers. Below, a short summary of the each study is provided.

(a) The summary from the NACA 0012 experimental and computational study are given below,

[i] The MF airfoil showed superior aerodynamic performance compared to the HF airfoil. The lift and drag measurements showed an increase of up to $C_{L,max} = 14\%$ for the MF airfoil relative to the HF airfoil. The $C_L/C_D$ performance at low angles of attack showed an improvement of up to 6% for MF airfoil. The MF airfoil shows improved lift characteristics at low angles of attack $\alpha > 8^\circ$ compared to the HF airfoil. The stall characteristics were not altered by the MF airfoil.

[ii] To understand the improved aerodynamic performance of the MF airfoil, surface flow visualization was carried out for both the suction and pressure side over a wide range of angles of attack. The results showed delayed separation over the flap suction for the MF airfoil compared to the HF airfoil at low angles of attack $\alpha > 8^\circ$. The separation point over the airfoil suction at angles of attack $\alpha < 8^\circ$ did not show any difference between the HF and MF airfoil. The surface flow visualization on the pressure side did not portray any separation or mentionable flow features for both the HF and MF airfoil for all the tested angles of attack.

[iii] The mean velocity results showed increased wake deficit for the MF airfoil compared to the HF airfoil at the near-wake locations, along with increased flow deflection angle at far-wake locations. The MF airfoil shows up to 50% increased turbulent kinetic energy relative to the HF airfoil for all the presented angles of attack at the near-wake region. The turbulent kinetic energy results at the airfoil wake displayed a characteristic double
peak behavior, which was mainly contributed by the streamwise normal Reynolds stress component ($u'u'$). The MF airfoil showed higher peak values of the $u'u'$ at the trailing edge vicinity compared to the HF airfoil. Higher Reynolds shear stress component ($-u'v'$) at the near-wake locations is observed for the MF airfoil at all the presented angles of attack. The increased wake deflection angle for the MF airfoil results in the increased lift and the larger region of attached flow near the trailing edge resulting in the reduction of form drag and improved $C_L/C_D$ for the MF airfoil.

Large eddy simulations were performed to further investigate the unsteady characteristics of the HF and MF airfoil. The results from the LES simulations validate well with the mean wake velocity profiles from the PIV measurement. The HF airfoil showed increased pressure fluctuations right after the flap hinge point, whereas in the case of MF airfoil the increased fluctuations were located at the further downstream location closer to the trailing edge. The results of surface pressure spectra contours showed much higher energy content for the MF airfoil at low frequency $0.2–2$ kHz at the vicinity of the trailing edge relative to the HF airfoil. The MF airfoil also showed higher spanwise coherence at frequency $0.2–2$ kHz closer to the trailing edge. The streamwise cross-correlation of the surface pressure probes showed a larger flow structure for the MF airfoil. The MF airfoil with increased aerodynamic performance also showed increased noise from the far-field noise measurements. The results of the increased noise also correspond to the increased surface pressure fluctuations over the flap region for the MF airfoil. From this study, it can be concluded that even slight modification to the surface camber of an airfoil flap gives considerable aerodynamic gains at low angles of attack. From the present study, it can be incurred that even though the MF airfoil showed increased aerodynamic performance it comes at the cost of increased noise.

The summary from the MDA 30P30N high-lift airfoil experimental study are given below,

The results of the aerodynamic lift and drag measurements do not show much difference between the three tested configurations for the tested range of angles of attack and Reynolds number. The H-SCF had the best aerodynamic performance with superior lift-to-
drag ratio and the drag-polar plots compared to the Baseline and SCF configurations. The surface pressure distribution results showed that the use of the H-SCF and SCF leads to the decrease of the suction peak by up to 15% over the main-element of the airfoil. However, this does not appear to influence the aerodynamic performance of the high-lift airfoil.

[iii] The mean velocity contours from the PIV measurements showed large vorticity within the slat cove region for the Baseline configuration. The use of the H-SCF reduces the size of the vorticity substantially and the use of SCF completely eliminates the large vorticity. The contours of streamwise normal Reynolds stress components at the slat cove region showed high magnitude in the slat shear layer for all the three configurations at all the presented angles of attack. The use of the slat cove fillers does not essentially influence the turbulence levels in the slat cusp shear layer. The mean velocity profiles of the boundary layer over the main-element showed reduced slat wake deficit for both the H-SCF and SCF compared to the Baseline at all the measurement stations over the airfoil. The slat wake profiles in the near-wake region showed an insignificant difference in the mean wake velocity and turbulence kinetic energy between the three configurations at high angles of attack. This also corresponds to the insignificant aerodynamic changes observed between the Baseline, H-SCF and SCF configurations. However, only at low angles of attack ($\alpha = 6^\circ$) the results showed higher turbulence kinetic energy and increased mean velocity for the H-SCF compared to the Baseline and SCF configurations.

[iv] The POD modes with high energy showed the presence of large coherent structures, which arises from the slat cusp of the Baseline case and this large structures were broken down into smaller structures by the use of the slat cove fillers. Moreover, the POD modes also showed the reduction in the slat vortex shedding energy with the use of the H-SCF and also the capability to completely eliminate the vortex shedding with the use of SCF.

The near-field surface pressure measurements showed tonal narrow band peaks for the Baseline case at $St_s > 1$. The tonal peaks were eliminated by the use of H-SCF and SCF but showed increased noise at low-frequencies $St_s < 1$ compared to the Baseline. This was confirmed to be due to the non-propagating hydrodynamic of the flow field within the slat.
cavity and main-element as this low-frequency increase is not observed in the far-field noise measurements. The use of the slat cove fillers has shown to significantly reduce the noise up to $5 - 9$ dB compared to the Baseline in the far-field noise measurements and the overall sound pressure levels.

[v] The results of the near to far-field coherence studies have shown that a high level of coherence is present for all the configurations, particularly with a distinct broadband spectral humps at low-frequencies for the cases with the slat cove fillers. The contour plots of the wavelet coefficient showed that the Rossiter modes for the Baseline case are amplitudes modulated in time, however, these modes are absent for the configuration with slat cove fillers. The results of the higher spectral order analysis show that the Rossiter modes that arise from the slat cavity of the Baseline case display quadratic interaction amongst themselves. This experimental study shows that there is a need for more fundamental research on the low-frequency broadband hump ($St_s < 1$) that arises in the high-lift device slat noise. This study also shows that the use of the slat cove fillers eliminate the tonal modes generated by the slat cavity and reduces broadband noise without compromising the aerodynamic performance of the high-lift device.

6.1 Research contribution

The current literature available for morphing structures is more inclined toward, shape optimization, solid mechanics and fluid-structure interaction. Even the very few available studies on the aero aspect are only concerned with the mean aerodynamics. First aspect of the presented work is focused on expanding the understanding of the aerodynamic and aeroacoustic performance of airfoils with morphed trailing edges. This study provides a large data set from the experimental measurements, which would further aid computational modeling. The investigation of the aerodynamic performance and mean flow fields revealed that the delayed flow separation resulted in the improved aerodynamic performance for the morphed trailing edge at low angles of attack. The unsteady flow field and aeroacoustic characteristics of the morphed airfoils have given new insights with the morphing trailing edge configurations generating more noise compared to the
conventional hinged trailing edge.

The second aspect of the presented works is on the use of morphing structures on a slat. A wide varied of experimental and computational study are available on 30P30N high-lift airfoil. However, only few a experimental studies are available on noise reduction capabilities with slat modifications. This study has shown with the use of slat cove filler aerodynamic performance are not lost while reducing a considerable levels of slat noise. This study focused on further understanding the slat noise mechanism by eliminating the slat cavity to see the resultant slat noise in a state of the art aeroacoustic facility at the University of Bristol. The study fulfils the need for the near-field unsteady surface pressure measurements and near to far-field coherence studies for slat cove fillers in comparison with the Baseline. Moreover, the broadband hump found at low-frequencies $St_s < 1$ for the Baseline has always been thought to be due to the low-frequency slat cove oscillation. But, the present study has shown that the source of the broadband hump was not related to the slat cavity as it was also seen in the coherence results for the slat cove filler configurations. Further fundamental studies would be required to identify its source.

### 6.2 Future work

The results from the first half of the presented study showed the superior aerodynamic characteristics of morphing structure. The study also showed that the airfoil trailing edge camber plays a major role in the noise generation mechanism. The second half of the study showed the possibility to suppress slat noise without compromising the aerodynamic performance for high-lift airfoil. The results also shed light in to the slat noise phenomenon with the identification of the low-frequency bump, which showed no relation to the slat tones. The suggestions and recommendations for the future work are listed below.

[i] In order to further understand the aeroacoustic behavior of the HF and MF airfoil, experimental study using the near- and far-field microphones, directivity measurements and beam forming should be carried out.

[ii] As a next step to further improve our understanding of morphed trailing edges, aerodynamic and aeroacoustic performance of spanwise morphing trailing edge should be carried
out.

[iii] When considering flap noise, the flap side edge noise is a major contributor. Therefore, aeroacoustic investigation into the side edge noise reduction possibilities using morphing structures should be carried out.

[iv] Further fundamental research is required to identify the source of low-frequency broadband hump \((St_s < 1)\) using time resolved Particle Image Velocimetry for all the three configurations.

[v] Investigations using beamforming method for the 30P30N airfoil with and without the slat cove filler to isolate the noise sources.

[vi] The nature of the non-propagating hydrodynamic field for all the slat cove filler configurations should be further investigated.

[vii] The narrow band features are often not observed in studies at high Reynolds number conditions. Therefore, the effects of the slat cove filler and low-frequency broadband hump and the non-propagating hydrodynamic field needs to be further investigated at real flight conditions.

[viii] Other passive methods such as slat cove cover and slat extensions should be investigated to identify the low-frequency broadband hump.
BIBLIOGRAPHY


The slat cove filler profiles used in the experiments were extracted from the slat shear layer path, obtained from the RANS computations performed on the 30P-30N high-lift airfoil, as also explained in Section 3.1.4. The preliminary steady-state RANS simulations were carried out using the OpenFOAM opensource code using the $k-\omega$ SST turbulence model. Three different angles of attack, $\alpha = 3^\circ$, 5.5$^\circ$ and 8.5$^\circ$ were validated with existing experimental data sets to track the shear layer path.

### A.1 Computational setup

The multi-block structured two-dimensional grid for the 30P-30N high-lift airfoil was created using the ICEM CFD software. The computational setup had a domain size of 20c in the streamwise ($x$-axis) and crosswise ($y$-axis) directions, as shown in Fig A.3. The grid was intended to be orthogonal to the airfoil surface, as shown in the close-up picture of the grid in Figs. A.1 and A.2. Each element of the multi-element airfoil was treated as an individual airfoil since the flow around all the elements were of high interest, thus such a densely meshed. Previous studies have shown the solution for a two-dimensional 30P30N RANS simulation is grid independent at approximately 500,000 elements. The grid for the current study consisted of a total number of
APPENDIX A. SLAT COVE FILLER DESIGN

Figure A.1: Dense grid around 30P30N airfoil.

Figure A.2: Dense grid around slat (left) and flap (right).

630,000 elements. Initially, the entire multi-element airfoil was set with a $y^+ = 30$ and solved with wall-functions but it resulted in poor results especially within the slat cove due to overestimated recirculation, so after further testing only the slat cove was set to $y^+ = 1$ and used without wall-functions whereas the rest of the airfoil had a $y^+ = 30$ and used with wall-functions.
A.1. COMPUTATIONAL SETUP

The simulations were carried out for the 30P-30N airfoil with a retracted chord length of $c = 0.457m$ and with an inlet velocity of $U_\infty = 58 \text{ m/s}$, corresponding to a chord-based Reynolds number of $Re_c = 1.71 \times 10^6$. The simulations were carried out for the angles of attack $\alpha = 3^\circ, 5.5^\circ$ and $8.5^\circ$ and validated with experimental results from Murayama et al. [70]. The results for the $C_p$ distribution and flow field contours for all the tested three angles of attack are presented in Figs. A.5-A.16 and are discussed below.

Figure A.3: Full domain size used for gridding and simulation.
APPENDIX A. SLAT COVE FILLER DESIGN

A.1.1 Angle of attack of 3 degrees

![Graph showing mean surface pressure distribution around 30P30N airfoil with angle of attack, \( \alpha = 3^\circ \) and \( Re_c = 1.7 \times 10^6 \) compared to experiments by Murayama et al. [70].]

Figure A.4: Mean surface pressure distribution around 30P30N airfoil with angle of attack, \( \alpha = 3^\circ \) and \( Re_c = 1.7 \times 10^6 \) compared to experiments by Murayama et al. [70].

![Graph showing slat (left) and flap (right) mean surface pressure distribution for 30P30N airfoil with angle of attack, \( \alpha = 3^\circ \) and \( Re_c = 1.7 \times 10^6 \) compared to experiments by Murayama et al. [70].]

Figure A.5: Slat (left) and flap (right) mean surface pressure distribution for 30P30N airfoil with angle of attack, \( \alpha = 3^\circ \) and \( Re_c = 1.7 \times 10^6 \) compared to experiments by Murayama et al. [70].

The results for the mean surface pressure (\( C_p \)) distribution at the angle of attack \( \alpha = 3^\circ \) at chord-based Reynolds number are presented in Figs. A.4 and A.5. The overall pressure distribution over the entire 30P30N airfoil validate well with the experimental measurements by Murayama et al. [70]. The \( k - \omega \) SST model slightly overpredicts the suction peak over the main element at chord location \( x/c \approx 0.7 \). The \( C_p \) results over the slat and flap presented in Fig. A.5 validate very well with the experimental measurements.
The results for the velocity contours at angle of attack $\alpha = 3^\circ$ for the 30P30N airfoil are presented in Figs. A.6 and A.7. From the close-up view of the flow field around slat in Fig. A.7 it can be observed that the flow separating from the slat cusp scantly reattaches to the trailing edge of the slat just a before mixing with the slat wake and free-stream flow. From the close-up view of the flow field around the flap it can be seen that the slat wake and the main-element wake are mixing and losing momentum, thus interfering with the flow over the flap on the suction surface. The separation on the flap only occurs at around $x/c \approx 1.07$ at the very aft of the airfoil.
A.1.2 Angle of attack of 5.5 degrees

Figure A.8: Mean surface pressure distribution around 30P30N airfoil for angle of attack, $\alpha = 5.5^\circ$ and $Re_c = 1.7 \times 10^6$ compared to experiments by Murayama et al. [70].

Figure A.9: Slat (left) and Flap (right) mean surface pressure distribution for angle of attack, $\alpha = 5.5^\circ$ and $Re_c = 1.7 \times 10^6$ compared to experiments by Murayama et al. [70].

At the angle of attack $\alpha = 5.5^\circ$ (see Figs A.8 and A.9) the results for the mean surface pressure ($C_p$) distribution over the entire 30P30N airfoil validate well with the experimental measurements by Murayama et al. [70]. The $k-\omega$ SST model slightly overpredicts the suction peak over the main element at chord location $x/c \approx 0.7$, as seen at the low angle of attack $\alpha = 3^\circ$. The $C_p$ results over the slat and flap presented in Fig. A.9 validate very well with the experimental measurements. Over the suction side of the slat, the $k-\omega$ SST model marginally overpredicts the surface pressure at location $x/c \approx -0.07$. 222
Figure A.10: Mean velocity contour around 30P30N airfoil with angle of attack, $\alpha = 5.5^\circ$.

Figure A.11: Mean velocity distribution (left) and turbulent kinetic energy (right) for 30P30N airfoil slat with angle of attack, $\alpha = 5.5^\circ$ and $Re_c = 1.7 \times 10^6$.

The flow field and turbulent kinetic energy (TKE) around the slat, within the slat cove and slat trailing edge of the 30P-30N airfoil for the angle of attack, $\alpha = 5.5^\circ$ are shown in Fig. A.11. The flow leaving the slat cusp for $\alpha = 5.5^\circ$ reattaches to the slat trailing edge just a little before the reattachment location to that of $\alpha = 3^\circ$ which follows the trend of the results that were observed and discussed from previous PIV experimental results by Pascal [48], Jenkins [49] and Khorrami [93], where some of the flow after deflecting the slat trailing edge joins with the recirculation within the slat cove and some of the flow leaves through the slat gap. This flow behavior is more evident in the TKE contour plots where the TKE is much greater at the slat
lower surface at the shear layer impingement region and inside the slat cove implying on the recirculating flow.

Figure A.12: Mean velocity distribution (left) and turbulent kinetic energy (right) for 30P30N airfoil flap for angle of attack, $\alpha = 5.5^\circ$ and $Re_c = 1.7 \times 10^6$.

The flow field and turbulent kinetic energy around the flap for 30P-30N airfoil for the angle of attack $\alpha = 5.5^\circ$ is presented in Fig. A.12. The results show flow separation on the suction side of the flap for $\alpha = 5.5^\circ$ at around $x/c \approx 1.05$, which is slightly earlier than that of the previous angle of attack $\alpha = 3^\circ$. Even though the flow appears converged, the TKE contour plots appear to have not converged mostly due to the unsteady nature of the flow at the vicinity of the flap trailing edge implying further unsteady simulations are necessary in order to carry forward with 30P-30N airfoil detailed flow study.
A.1.3 Angle of attack of 8.5 degrees

Figure A.13: Mean surface pressure distribution around 30P30N airfoil with angle of attack, $\alpha = 8.5^\circ$ and $Re_c = 1.7 \times 10^6$ compared to experiments by Murayama et al. [70].

Figure A.14: Slat (left) and flap (right) mean surface pressure distribution for 30P30N airfoil with angle of attack, $\alpha = 8.5^\circ$ and $Re_c = 1.7 \times 10^6$ compared to experiments by Murayama et al. [70].

The results for the mean surface pressure ($C_p$) distribution at the angle of attack $\alpha = 8.5^\circ$ at chord-based Reynolds number are presented in Figs. A.13 and A.14. The overall pressure distribution over the entire 30P30N airfoil validate well with the experimental measurements by Murayama et al. [70]. The $k-\omega$ SST model accurately predicts the suction peak over the main-element at chord location $x/c \approx 0.7$. The slight over-prediction of the suction peak on the main-element seen earlier at angles of attack 3$^\circ$ and 5.5$^\circ$ is not observed here. The $C_p$ results over the slat and flap presented in Fig. A.5 validate very well with the experimental measurements.
Figure A.15: Mean velocity contour around 30P30N airfoil with angle of attack, $\alpha = 8.5^\circ$.

Figure A.16: Slat (left) and flap (right) mean velocity distribution for 30P30N airfoil with angle of attack, $\alpha = 8.5^\circ$ and $Re_c = 1.7 \times 10^6$.

The velocity contours over the 30P30N airfoil at the angle of attack $\alpha = 8.5^\circ$ for chord-based Reynolds number $Re_c = 1.7 \times 10^6$ is presented in Fig. A.16. From Fig. A.16 it can be observed that unlike the flow behavior at lower angles of attack, at $\alpha = 8.5^\circ$ the wake of the slat and the main-element are more prominent and do not mix until further downstream locations at $x/c \approx 1.5$.

The velocity contours showing the flow field around the slat and flap for 30P-30N airfoil at the angle of attack $\alpha = 8.5^\circ$ are presented in Fig. A.15. The flow leaving the slat cusp reattaches with the slat trailing edge much closer to the slat mid-chord than the previously discussed lower angles of attack showing that as the angle of attack increases the reattachment impingement
A.1. COMPUTATIONAL SETUP

point is moved further inwards, which would add to the momentum of the recirculating flow within the slat. The close-up view of the flow around the flap shows the two distinct wakes from the slat and main-element and its less interference with the flow on the flap suction surface.

The flow field contour plots presented in Figs. A.7, A.11 and A.16 clearly show that the slat cusp shear layer path length is sensitive to angle of attack. This path length reduces as the angle of attack is increased. For the current study as suggested by Imamura et al. [72, 73] the case with the smallest shear layer path is selected ($\alpha = 8.5^\circ$) and used to design the slat cove filler profile shown in Fig. A.17. Further discussion on the available literature on slat cove filler can be found in Section 3.1.4.

(a) TKE RANS, $\alpha = 8.5^\circ$

(b) SCF Insert

Figure A.17: (a) Shematic of the manually extracted shear layer path using contours of turbulent kinetic energy around 30P30N airfoil slat for an angle of attack, $\alpha = 8.5^\circ$ at $Re_c = 1.7 \times 10^6$ and (b) The 3D printed SCF fitted on the 30P30N airfoil in the low turbulence wind tunnel
In the field of aeroacoustic it is of utmost importance to be able to accurately predict the noise generated by an airfoil or a jet at a far-field location. A common way of predicting the far-field noise is to use computational fluid dynamics to simulate the flow field and use the resultant accurate flow field as an input into acoustic analogies to predict the noise. The very first acoustic analogy was formulated by Lighthill in 1952 [108]. Adopting and improving Lighthill's method several other acoustic analogies were developed over the years such as Curle [109], Ffowcs Williams and Hawking [110], Amiet [111] and Howe [112]. The analogy is a simple representation of the Navier-Stokes equation, considering the wave operators on the left hand side and source terms on the right hand side. Lighthill's equation due to its limitation has been modified to be suitable for many other general problems and its further explained in this section. In what follows, we will provide a short description of the Lighthill's acoustic analogy and Curle's acoustic analogy.

### B.1 Lighthill's acoustic analogy

Lighthill's acoustic analogy can be derived from the fluid dynamics governing equation, namely the *continuum and momentum equation*. They are
The Cartesian form of the continuity equation is shown in Eq. B.1 and the Reynolds form of the
momentum equation is shown in Eq. B.2, where, \( p \), \( u_i \), \( \rho \), and \( \tau_{ij} \) are the fluid pressure, fluid
velocity, fluid density and the viscous stress, respectively. Differentiate Eq. B.1 with respect to
time, and Eq. B.1 with respect to space, the subtract the latter from the differentiated Eq. B.1 to
obtain,

\[
\frac{\partial^2 \rho u_i}{\partial t^2} - \frac{\partial^2 \rho u_i u_j}{\partial x_i \partial x_j} = \frac{\partial^2 p}{\partial x_i} - \frac{\partial^2 \tau_{ij}}{\partial x_j}.
\]  

(B.3)

The fluctuating fluid properties, such as the density fluctuation \( \rho' = \rho - \rho_0 \) and the pressure
fluctuation \( p' = p - p_0 \) are now introduced, where \( \rho_0 \) and \( p_0 \) are the atmospheric density and
pressure, respectively. The Lighthill’s inhomogenous wave equation can now be derived by
subtracting \( c_0^2 \frac{\partial^2 \rho}{\partial x_i^2} \) from both the left and right hand side of the Eq. B.3, which results in,

\[
\frac{\partial^2 \rho'}{\partial t^2} - c_0^2 \frac{\partial^2 \rho'}{\partial x_i^2} = \frac{\partial^2 T_{ij}}{\partial x_i \partial x_j}.
\]  

(B.4)

The term \( T_{ij} \) is the Lighthill’s sterss tensor in the Lighthill’s inhomogenous wave equation in
Eq. B.4 and is defined as,

\[
T_{ij} = \rho u_i u_j + \delta_{ij} (p' - c_0^2 \rho') - \tau_{ij},
\]  

(B.5)

where \( c_0 \) is the speed of sound \( \delta_{ij} \) is the Kronecker delta.

The exact terms containing the physics of the sound propagation is contained in Eq. B.4 as
it does not contain any assumptions. In Lighthill’s analogy, no rigid objects or surfaces were
allowed as it was mainly developed for free jet streams. This made the direct use of Lighthill’s
analogy impossible for many applications involving solid surfaces. Therefore, this equation can
be viewed as an inhomogeneous wave equation with assumptions, such as the right hand side

of the equation is known independent of the left hand side, meaning that sound propagation is
separated from its source.

**B.2 Curle’s acoustic analogy**

As mentioned earlier Lighthill’s analogy covered only the application of theory without any solid
boundaries. Since it was very evident that solid boundaries played an important role in noise
generation Curle provided an extension for Lighthill’s general theory of aerodynamic sound that
incorporated the influence of the solid boundaries [109]. Curle [109] showed the part played by
solid boundaries in detail, thus increasing the number of global sound generation. They can be
explained by the following two points.

1) The sound generated from the quadrupoles of Lighthill’s acoustic analogy will be further
calculated taking into consideration of the reflection and diffraction due to the presence of solid
boundaries.

2) Dipole sources generated due to the interaction between the fluid and the solid boundary.

Curle contributed to Lighthill’s analogy by an additional term \( \frac{\partial f_i}{\partial x_i} \) to the right-hand side of
Eq. B.4. The dipole sources are much more efficient noise mechanism at low Mach numbers than
the quadrupole sources. The wave characteristics will be changed by as the radiated sound will
be reflected and diffracted by the solid boundaries. In most cases, the quadrupole sources are
neglected and only the dipole sources are often used for evaluating the far-field acoustics. The
most general form of the Lighthill’s inhomogeneous wave equation solution is as follows,

\[
\rho' = \frac{1}{4\pi c_0^2} \int_V \frac{\partial^2}{\partial y_i y_j} \cdot \frac{T_{ij}}{r} dV(y) + \frac{1}{4\pi} \int_S \left\{ \frac{1}{r} \frac{\partial p}{\partial n} + \frac{1}{r^2} \frac{\partial r}{\partial n} \rho + \frac{1}{c_0 r} \frac{\partial r}{\partial t} \frac{\partial \rho}{\partial n} \right\} dS(y),
\]

(B.6)

where \( x \) and \( y \) are the location of the sound and observer \( r = |x - y| \), \( S \) is surface of the solid
boundaries, \( V \) is the volume outside to the solid boundaries and \( n \) is outward normal to the fluid.

In the above Eq. B.6 \( \frac{\partial^2 T_{ij}}{\partial y_i y_j}, \frac{\partial \rho}{\partial t}, \) and \( \frac{\partial \rho}{\partial n} \) are calculated at lagging times \( t - \frac{r}{c_0} \).

Curle introduced the free-space Green’s function and further simplified the Eq. B.6. The
simplified equation is as follows,
For flows at low Mach number the first integral in Eq. B.7 can be neglected, thus not considering the quadrupole sources. Several assumptions were made in order to formulate Eq. B.7, and the readers are referred to the paper by Curle [109] for detail description. More details on Lighthill’s acoustic analogy, Curle’s acoustic analogy, and FW-H acoustic analogy, the derivation, assumption and their application can be found in the paper by Goldstein [113].