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RPPR Final Report

as of 11-Feb-2022

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Major Goals: The original goals of this program were to study the effects of compressibility on the unsteady loading of an airfoil in a surging flow. The impetus for this study was a preliminary data set that showed significant differences between measured loading and the anticipated (theoretical) result. Our initial hypothesis was that compressibility effects were the primary reason for the discrepancy, and our goal was to elucidate the specific compressible flow features that led to the departure from theoretical predictions. The initial phase of the proposed project involved a low-Mach number validation in order to establish a baseline condition which we presumed would show agreement between experiment and theory. However, in this initial set of data, we were unable to obtain a data set that showed agreement between theory and the experimental results. (Note: our experimental results also disagreed with recent low-M, low-Re experimental results from another research group). Thus, we carefully considered all experimental factors and confounding error sources, but there remained significant deviations between our results and the established theory. At this point, we reevaluated the scope of the investigation and framed a new hypothesis to explain the discrepancy. This hypothesis was that the classical (steady) Kutta condition was violated in this unsteady surging flow. Thus, we framed a new set of objectives for this research program, in order to rule out any extenuating or confounding factors and to assess the validity of this new hypothesis. These goals are to:

- 1) Carefully characterize the unsteady characteristics of the wind tunnel, through experiments, computations, and wave propagation theory.
- 2) Evaluate unsteady velocity measurement techniques.
- 3) Develop the background-oriented schlieren (BOS) measurement technique in the unsteady tunnel for visualizing flow properties in the trailing-edge stagnation region.
- 4) Directly measure the trailing edge stagnation streakline as a function of time (phase angle of oscillations), compared with loads inferred from pressure data, to assess the validity of the hypothesis that the steady Kutta condition does not hold.
- 5) Evaluate the impact of various flow physics on the trailing edge flow, and the resulting loading on the airfoil. Effects such as boundary layer state, shape factor, and momentum thickness are likely to have a significant effect. Boundary layer properties will be tailored by controlling Re, angle of attack, airfoil shape, and differential BL trip.

Accomplishments: Please see attached report.

RPPR Final Report as of 11-Feb-2022

Training Opportunities: The doctoral student appointed to this project was successfully mentored to degree completion in December 2021. This is the same student who was appointed to the grant throughout the project, so this funding directly contributed to his successful graduation with a doctorate in aerospace engineering. In addition to the primary doctoral student, three M.S. students, three B.S. students, and two high school students were supported at various times. All students had the opportunity to receive training on how to pose and answer research questions, specific training on measurement techniques and wind tunnel operations, data analysis methods, and how to report results.

Results Dissemination: Four papers have been published to date on the findings from this grant (1 archival journal publication and three abstract-reviewed AIAA conference papers). In addition, members of our group have given 8 presentations on the work in various venues, including invited seminar presentations at universities (5), invited presentations at the AIAA SciTech conference (2), and a regular presentation at the APS fluid dynamics meeting (1). The doctoral student working on this grant, Wenbo Zhu, also completed his doctoral dissertation in December 2021, shortly after grant completion. In the coming months, we intend to submit key portions of his doctoral dissertation for publication in peer-reviewed archival journals.

Following is a citation list:

Journal Papers:

Zhu, W, McCrink, MH, Bons, JP, and Gregory, JW, 2020, "The unsteady Kutta condition on an airfoil in a surging flow," *Journal of Fluid Mechanics*, v. 893, n. R2. doi: 10.1017/jfm.2020.254.

Thesis:

Zhu, W, 2021, "Experimental Investigation of the Lift Frequency Response of a Surging Airfoil and its Trailing-Edge Flow Physics at High Reynolds Number," Ph.D. dissertation, Department of Mechanical and Aerospace Engineering, The Ohio State University.

Conference Papers:

Zhu, W, Harter, B, Gregory, JW, and Bons, JP, 2018, "Characterizing Wave Propagation in an Unsteady Transonic Wind Tunnel," 2018 AIAA Aviation Forum (AIAA 2018-3568), Atlanta, GA, June 25-29, 2018. doi: 10.2514/6.2018-3568.

Zhu, W, Bons, JP, and Gregory, JW, 2019, "Reynolds Scaling Effects on Dynamic Stall of VR-7 and VR-12 Airfoils," 2019 SciTech (AIAA 2019-0304), San Diego, CA, January 7-11, 2019. doi: 10.2514/6.2019-0304.

Zhu, W, McCrink, MH, Bons, JP, and Gregory, JW, 2020, "Aerodynamic Performance and Trailing Edge Flow Physics on an Airfoil in an Oscillating Freestream," 2020 AIAA SciTech Forum (AIAA 2020-1758), Orlando, FL, January 6-10, 2020. doi: 10.2514/6.2020-1758.

Presentations:

Gregory, JW, Auburn University, Department of Aerospace Engineering, Auburn, Alabama, "Unsteady Compressible Aerodynamics in a Time-Varying Freestream," October 19, 2018.

Gregory, JW, University of California – Irvine, Department of Mechanical and Aerospace Engineering, "Unsteady Compressible Aerodynamics in a Time-Varying Freestream," January 11, 2019.

Gregory, JW, Purdue University, School of Aeronautics and Astronautics, West Lafayette, Indiana, "Limitations of Unsteady Airfoil Theory for Surging Flows," March 25, 2019.

Gregory, JW, University of Illinois Urbana-Champaign, Department of Aerospace Engineering, Urbana, IL, "Limitations of Unsteady Airfoil Theory for Surging Flows," November 14, 2019.

Gregory, JW, University of Texas, Department of Aerospace Engineering, Austin, TX, "Limitations of Unsteady Airfoil Theory for Surging Flows," February 6, 2020.

Gregory, JW, Zhu, W, and Bons, JP, 2018, "Unsteady Compressibility Effects on a Surging Airfoil," invited presentation for a special session on surging airfoils, AIAA SciTech, January 8, 2018.

Gregory, JW, Zhu, W, Altamirano, G, Plank, J, and Bons, JP, 2018, "Unsteady Lift on a Fixed-AoA Airfoil in Unsteady Freestream," BAPS.2018.DFD.A16.00005, 71st Annual Meeting of the APS Division of Fluid Dynamics, November 18-20, 2018, Atlanta, GA.

Gregory, JW, Zhu, W, Altamirano, G, Harter, B, McCrink, M, and Bons, JP, 2019, "Limitations of Unsteady Airfoil Theory for Surging Flows," invited presentation for a special session on surging airfoils, AIAA SciTech, January 7, 2019.

RPPR Final Report as of 11-Feb-2022

Honors and Awards: Mr. Braxton Harter, student who worked on some on tasks associated with the grant, was selected as an NDSEG fellow.

James Gregory (PI) was selected as a Fellow of the Royal Aeronautical Society (2021) and the American Institute of Aeronautics and Astronautics (2022).

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Participant Type: PD/PI

Participant: James Gregory

Person Months Worked: 3.00

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Participant: Jeffrey Bons

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Participant Type: Staff Scientist (doctoral level)

Participant: Matthew McCrink

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Participant: Wenbo Zhu

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Participant Type: Graduate Student (research assistant)

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Participant: Braxton Harter

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Participant Type: Undergraduate Student

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Person Months Worked: 2.00

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Abstract: This work presents an experimental validation study of Isaacs' incompressible unsteady-airfoil theory at Reynolds numbers above 10^6 , and explores the validity of the classical Kutta condition applied to surging flows. Harmonic variation of the free-stream velocity was produced by rotating choke vanes in an unsteady transonic wind tunnel, with time-resolved lift coefficients determined from surface pressure measurements on a NACA 0018 airfoil. Unsteady lift results demonstrate the same trends with reduced frequency and velocity amplitude ratio that are predicted by Isaacs' theory. However, significant deviations of the lift magnitude and phase angle are observed. In order to understand the cause of these deviations, the background-oriented schlieren technique was used to visualize density gradients in the immediate vicinity of the airfoil trailing edge. The time-resolved background-oriented schlieren displacement field indicates oscillatory behaviour of the trailing-edge stagnation...

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Partners

,

I certify that the information in the report is complete and accurate:

Signature: James W. Gregory

Signature Date: 2/11/22 11:33AM

Final Report
“Unsteady Aerodynamics - Unsteady Compressibility Effects for Modern Rotorcraft”
Proposal Number 69177EG

James W. Gregory (PI)

Jeffrey P. Bons (co-PI)

Report co-authored with Dr. Wenbo Zhu (Ph.D. student funded on the grant)

February 13, 2017 – August 31, 2021

Note: During the scope of the work, the original hypothesis on compressibility effects was quickly found to be incorrect. We pivoted the objectives of the grant, in coordination with the program manager, leading to work that culminates in this report which is more appropriately

titled,

“Effects of the Unsteady Kutta Condition on Unsteady Lift Generation
on an Airfoil in Surging Flow”

Abstract

With the growing interest in faster helicopters, commercial unmanned vertical lift vehicles, and vertical-axis-wind-turbines, it becomes critical to understand the unsteady loading on the rotor blade so that more efficient designs can be produced. The classic unsteady airfoil theories are tools useful to predict loads on two-dimensional rotor blade sections for moderate surging, pitching, and plunging motions. Specifically, Isaacs' surging airfoil theory assumes a two-dimensional thin flat plate at small angles of attack under a sinusoidally-oscillating freestream in an incompressible inviscid flow. The unsteady lift coefficient is then solved by the potential flow theory with the Kutta condition at the trailing-edge. However, Isaacs' surging airfoil theory has not been fully validated experimentally. As a result, this work is intended to investigate this surging airfoil theory experimentally and evaluate the key parameters in such unsteady motion, the flow physics, and the validity of the various assumptions. Firstly, the wind tunnel at the Aerospace Research Center, the Ohio State University, is modified to generate a sinusoidally-oscillating freestream in the test section with the design of rotating elliptical choke-vanes downstream. The modified unsteady wind tunnel is characterized to understand its unsteady response for a surging airfoil experiment, with the help of various intrusive measurements and analytical and numerical tools. Secondly, a NACA 0018 and a NACA 63-015A airfoil were tested at moderate surging flow conditions in terms of Reynolds number, Mach number,

velocity amplitude, and reduced frequency. Surface pressure measurements of the lift frequency response indicated a significant disagreement with Isaacs' theory in terms of the lift overshoot and undershoot, where optical measurements revealed signs of an oscillatory trailing-edge stagnation condition for the surging airfoil which violates the classic Kutta condition in the unsteady potential flow theory. Finally, a hypothesis was formulated about the momentum balance of shear layers at the trailing-edge, which is intended to develop a viscous and unsteady correction to the classic Kutta condition and connect the unsteady stagnation condition with the resulting lift frequency response. A near-wake velocity survey was conducted on a NACA 63-015A airfoil with various flow transition behaviors. Results supported the hypothesis that a dynamic shear layer momentum balance exists near the trailing-edge and its dynamics can be correlated to the measured lift undershoot and overshoot. Overall, the experimental study in this work offers additional insight into the unsteady trailing-edge stagnation condition, and highlights the need to modify the Kutta condition with unsteady and viscous effects for a surging airfoil.

Table of Contents

	Page
Abstract	1
List of Tables	6
List of Figures	7
1. Introduction	1
1.1 Unsteady Aerodynamics of Rotorcraft Blades	1
1.2 Classic Unsteady Airfoil Theory	4
1.3 Kutta Condition and the Controversy	12
1.4 Motivations	17
2. Methodology for Surging Airfoil Investigations	19
2.1 Wind Tunnel Facility	19
2.2 Test Articles	23
2.3 Optical Measurements	26
2.3.1 Background-Oriented Schlieren Technique	26
2.3.2 Infrared Thermography	28
2.4 Traversing Wake Rake	29
2.5 Computational Model	35
3. Characterization of the Unsteady Wind Tunnel	38
3.1 Generation of Surging Freestream	38
3.2 Motivations and Objectives	41
3.3 Methods	44
3.3.1 Experimental Instrumentation	44
3.3.2 In-Situ Calibration of the Hot-Film Probe	49

3.3.3	Numerical Model	55
3.4	Results and Discussion	57
3.4.1	Effects of the Forcing Frequency	57
3.4.2	Evaluation of the Time-Varying Freestream Velocity	64
3.4.3	Turbulence Intensity	74
3.5	Summary	78
4.	Utilization of Background-Oriented Schlieren Method for Boundary Layer and Wake Measurements in A Blow-Down Type Wind Tunnel	80
4.1	Density Gradients in the Flow Field	80
4.2	Characteristics of the Steady Boundary Layer	87
4.3	Characteristics of the Steady Wake	94
4.4	Summary	96
5.	Validation of the Classic Surging Airfoil Theory	99
5.1	Test Matrix and Uncertainty Analysis	99
5.2	Characteristics of Steady Lift Coefficients	102
5.3	Quasi-Steady Lift Coefficient	111
5.4	Unsteady Frequency Response	114
5.5	Discussion	132
5.5.1	Thin Airfoil	132
5.5.2	Low Angles of Attack	134
5.5.3	Incompressible Flow	134
5.5.4	Zero Streamwise Pressure Gradient	134
5.5.5	Planar Wake	135
5.5.6	Shed Vorticity Convects at Local Freestream Velocity	136
5.5.7	Inviscid Potential Flow	137
5.5.8	Kutta Condition	137
5.6	Summary	139
6.	Experimental Investigation of the Unsteady Viscous Kutta Condition for a Surging Airfoil	141
6.1	Modification of Shear Layer Momentum	142
6.2	Characteristics of the Steady Airfoil	144
6.3	Characteristics of the Surging Airfoil	158
6.4	Summary	170
7.	Conclusions	173

Appendices	176
A. Location of the Surface Pressure Tap	176
B. MATLAB Code for the Rotating Choke-Vane Design	178
C. Characterization of the Frequency Response of the Pitot-Static Probe in an Acoustic Resonance Cavity	183
D. Analytic Model for the Unsteady Response of the Wind Tunnel	186
Bibliography	189

List of Tables

Table	Page
5.1 Test matrix of the surging airfoil experiment.	100
5.2 Uncertainty of the experimental measurements.	101
5.3 Predicted and measured flow transition on the upper surface of the NACA 63-015A airfoil.	110
A.1 Pressure tap location on NACA 0018.	176
A.2 Pressure tap location on NACA 63-015A.	177
C.1 Dimension of the pitot-static probe and the pneumatic line.	185

List of Figures

Figure		Page
1.1	Illustration of Mach number across the rotor disk for a helicopter under forward flight (from Wang & Zhao 2016).	2
1.2	Illustration of various unsteady phenomena experienced by the helicopter rotor blade (Helicopter Flying Handbook, FAA).	4
1.3	Lift frequency response predicted by Isaacs' theory, a) effect of advance ratio σ , b) effect of reduced frequency k	7
1.4	Schematic of the oscillating freestream, the shed wake vorticity, and the induced normal velocities (from Strangfeld <i>et al.</i> 2016).	8
1.5	Schematic of possible trailing-edge streamlines under oscillatory freestream (from Poling & Telionis 1986), a) at decelerating phases, b) at accelerating phases.	15
1.6	Free vortex model by Xia & Mohseni 2017, a) schematic of the free vortex sheet for a finite-angle trailing-edge, b) the angle of the trailing-edge stagnation condition as the ratio of boundary edge velocities. . .	16
2.1	Schematic of the blow-down type unsteady wind tunnel.	22
2.2	Theoretical operating range of the wind tunnel evaluated at the room temperature and a characteristic length of 15.24cm.	23
2.3	Model of the NACA 0018 airfoil, a) isometric view, b) surface pressure tap distribution.	24
2.4	Model of the NACA 63-015A airfoil, a) isometric view, b) surface pressure tap distribution.	25

2.5	Background-oriented schlieren technique, a) schematic showing each component in a BOS experiment, b) setup of the current BOS wind tunnel experiment.	28
2.6	Setup of the infrared camera for airfoil surface temperature measurements during the wind tunnel experiment.	30
2.7	Schematic of the traversing wake rake during wind tunnel experiments.	31
2.8	Illustration of the machined wake rake assembly.	32
2.9	Traversing wake rake installed in the test section for wind tunnel experiments.	33
2.10	Comparison of wake survey strategies from a steady wind tunnel operation.	35
2.11	Computational model, a) the computational domain for NACA 0018 airfoil, b) the grid independence study based on distribution of the pressure coefficient C_p	36
3.1	Dimension of the elliptical choke vane from the inverse design process.	40
3.2	Error analysis of the sinusoidal freestream Mach number based on the choke vane design.	41
3.3	Schematic of instruments installed in the wind tunnel test-section. . .	44
3.4	Schematic of the pitot-static probe made by United Sensor, dimensions are in inches.	46
3.5	Schematic of the fast-response temperature probe designed by Winkler 1945.	47
3.6	Schematic of the Dantec hot-film CTA system.	49
3.7	Probe array at the test-section of the unsteady wind tunnel.	50
3.8	Time history of flow parameters during a typical 15-second steady blow-down operation.	53

3.9	Time history of pressure measured during a frequency-sweep wind tunnel experiment. Measured pressure at different intervals (7-8s, 15-15.5s, 20-20.5s) and frequencies (1.5Hz, 5Hz, 8Hz) are highlighted.	54
3.10	Calibration curve of the hot-film probe.	55
3.11	Phase-averaged parameters vs. the forcing frequency, subplots are aligned based on the accelerating mean of the Mach number, a) stagnation pressure, b) static pressure, c) Mach number from isentropic relation.	58
3.12	Transfer function of the pressure measuring system from the acoustic excitation experiment and compared with predictions from the B-T model.	60
3.13	Power spectrum of the pressure measurements from the unsteady experiment, a) forcing frequency = 1.5Hz , b) forcing frequency = 3.5Hz, c) forcing frequency = 7Hz, d) forcing frequency = 10.5Hz.	63
3.14	Time history of Re_d and comparison of hot-film and pitot-static probe measurements.	65
3.15	Comparison of the power spectrum of the film Reynolds number Re_d , a) forcing frequency = 1.5Hz , b) forcing frequency = 3.5Hz, c) forcing frequency = 7Hz, d) forcing frequency = 10.5Hz.	67
3.16	Power spectrum of the freestream velocity measured by the hot-film probe, a) forcing frequency = 1.5Hz , b) forcing frequency = 3.5Hz, c) forcing frequency = 7Hz, d) forcing frequency = 10.5Hz.	68
3.17	Time history of the freestream velocity and comparison of hot-film and pitot-static probe measurements.	69
3.18	Comparison of freestream parameters based on hot-film measurements and the isentropic relation, a) film Reynolds number, b) velocity, c) Mach number.	70
3.19	Ratio of velocity amplitude as a function of the forcing frequency. . .	72

3.20	Phase difference of pressure at different streamwise locations as a function of the forcing frequency.	73
3.21	Comparison of freestream velocity based on the choke-vane inverse design, the unsteady Euler code, and the experimental measurements, a) forcing frequency = 1.5Hz, b) forcing frequency = 3.5Hz, c) forcing frequency = 7Hz, d) forcing frequency = 10.5Hz.	75
3.22	Schematic of the wake of the rotating elliptical choke vane.	76
3.23	Turbulence intensity of the sinusoidally-oscillating freestream $M \approx 0.2 \pm 0.05$, $k = 0.025$	78
4.1	Surface temperature of steady NACA 0018 airfoil at $\alpha = 4^\circ$, a) snapshot of the instantaneous temperature field of the airfoil suction surface measured by the IR camera, b) comparison of the instantaneous freestream flow temperature and the airfoil surface temperature (averaged over the chord at mid-span).	83
4.2	Steady pixel displacement field around the airfoil trailing-edge, $\alpha = 4^\circ$, a) from experimental BOS measurements, b) from CFD simulation with highlighted streamlines.	86
4.3	Normalized temperature profiles from the suction surface of the airfoil, $\alpha = 4^\circ$, a) effects of streamwise location, b) thermal effects.	87
4.4	Normalized density profiles from the suction surface of the airfoil, $\alpha = 4^\circ$, a) effects of streamwise location, b) thermal effects.	90
4.5	Density gradient profiles from the suction surface of the airfoil, $\alpha = 4^\circ$, a) effects of streamwise location, b) thermal effects.	91
4.6	Pixel displacement profiles from the suction surface of the airfoil, $\alpha = 4^\circ$, a) comparison of the analytic model, the CFD simulation, and the BOS measurements, b) displacement profiles normalized by local maximum magnitude.	92
4.7	Comparison of the zero-displacement line from CFD simulations and BOS measurements as a function of the angle of attack.	95

4.8	Comparison of the wake profile in terms of the absolute displacement magnitude from CFD simulations and BOS measurements, a) $\alpha = 2^\circ$, b) $\alpha = 4^\circ$, c) $\alpha = 8^\circ$	96
4.9	Comparison of the wake profile at $x/c = 1.05$ in terms of the absolute displacement magnitude from CFD simulations and BOS measurements as a function of the angle of attack.	97
5.1	$C_l - \alpha$ of the NACA 0018 airfoil under steady freestream conditions.	103
5.2	$C_l - \alpha$ of the NACA 63-015A airfoil under steady freestream conditions.	104
5.3	Experimentally-measured C_p distribution of the NACA 0018 airfoil under steady freestream conditions.	105
5.4	Comparison of C_p distribution between analytic prediction and experimental measurements of the NACA 0018 airfoil under steady freestream conditions, a) $\alpha = 2^\circ$, b) $\alpha = 4^\circ$, c) $\alpha = 6^\circ$, d) $\alpha = 8^\circ$	106
5.5	Experimentally-measured C_p distribution of the NACA 63-015A airfoil under steady freestream conditions.	107
5.6	Comparison of C_p distribution between analytic prediction and experimental measurements of the NACA 63-015A airfoil under steady freestream conditions, a) $\alpha = 3^\circ$, b) $\alpha = 4^\circ$, c) $\alpha = 6^\circ$	108
5.7	Flow transition of the NACA63-015A airfoil under steady freestream measured by IR camera, $\alpha = 4^\circ$, a) instantaneous thermography, b) instantaneous surface temperature along the chord at $z/c = 0$	109
5.8	Displacement field of the airfoil under the steady freestream condition, $\alpha = 4^\circ$, a) NACA 0018, b) NACA 63-015A.	111
5.9	Steady trailing-edge stagnation streamline from BOS measurements, a) NACA 0018, b) NACA 63-015A.	112
5.10	Quasi-steady C_p distribution of the NACA 63-015A airfoil a) untripped airfoil experimental data, $\alpha = 1^\circ$, b) prediction from the panel method, $n = 4$, $\alpha = 1^\circ$	115
5.11	Comparison of the quasi-steady C_l of the NACA 63-015A airfoil.	116

5.12	Comparison of normalized unsteady lift coefficients for NACA 0018 at mean $Re_c = 1.5 \times 10^6$, mean $M_\infty = 0.21$, $k = 0.025$, and $\sigma = 0.21$. . .	117
5.13	Comparison of normalized unsteady lift coefficients for NACA 63-015A at mean $Re_c = 1.5 \times 10^6$, mean $M_\infty = 0.21$, $k = 0.025$, and $\sigma = 0.21$. . .	118
5.14	Effects of reduced frequency to normalized unsteady lift coefficients for NACA 0018 at mean $Re_c = 1.5 \times 10^6$, mean $M = 0.2$, $\sigma = 0.21$, and $\alpha = 4^\circ$	119
5.15	Comparison of the dynamics of the phase-averaged freestream flow and the unsteady lift frequency response, NACA 0018, $\alpha = 4^\circ$, $k = 0.05$, a) Mach number b) acceleration $dM/d\phi$	120
5.16	Comparison of the dynamics of the phase-averaged freestream flow and the unsteady lift frequency response, NACA 63-015A, $\alpha = 4^\circ$, $k = 0.025$, a) Mach number b) acceleration $dM/d\phi$	121
5.17	Comparison of the dynamics of the time-resolved freestream flow and the unsteady lift frequency response, NACA 63-015A, $\alpha = 4^\circ$, $k = 0.025$, a) Mach number b) acceleration $dM/d\phi$	122
5.18	Phase-averaged C_p distribution of NACA 0018 at $\phi = 90^\circ$ (maximum freestream velocity).	123
5.19	Phase-averaged C_p distribution of NACA 0018 at $\phi = 270^\circ$ (minimum freestream velocity).	124
5.20	Displacement field around the surging NACA 0018 airfoil trailing-edge at selected phase angles from BOS measurements, $\alpha = 2^\circ$, a) $\phi = 0^\circ$, b) $\phi = 90^\circ$, c) $\phi = 180^\circ$, d) $\phi = 270^\circ$	125
5.21	Displacement field around the surging NACA 63-015A airfoil trailing-edge at selected phase angles from BOS measurements, $\alpha = 3^\circ$, a) $\phi = 0^\circ$, b) $\phi = 90^\circ$, c) $\phi = 180^\circ$, d) $\phi = 270^\circ$	126
5.22	Trailing-edge stagnation streakline of the surging NACA 0018 airfoil at selected phase, a) $\alpha = 2^\circ$, b) $\alpha = 4^\circ$, c) $\alpha = 6^\circ$	128

5.23	Schematic of the effective camber effect due to oscillation of the trailing-edge stagnation point.	130
5.24	Height of the effective Gurney flap of the NACA 0018 airfoil based on trailing-edge stagnation condition measured by the BOS technique, $\alpha = 4^\circ$, $k = 0.025$	131
5.25	Comparison of effective camber effect and Isaacs' theory, a) $\alpha = 2^\circ$, b) $\alpha = 4^\circ$, c) $\alpha = 6^\circ$	133
6.1	Effective trip wire diameter as a function of the angle of attack, NACA 63-015A airfoil, $x_k/c = 0.05$, $Re_c = 1.5 \times 10^6$, $M_\infty = 0.2$	143
6.2	Comparison of the steady lift coefficient of the NACA 63-015A airfoil with different transition conditions.	144
6.3	Experimentally measured C_p distribution of the NACA 63-015A airfoil, a) tripped at $x/c = 0.05$ on the suction surface, b) tripped at $x/c = 0.05$ on the pressure surface.	146
6.4	Comparison of the steady C_p distribution of the NACA 63-015A airfoil with different transition conditions, a) $\alpha = 1^\circ$, b) $\alpha = 2^\circ$, c) $\alpha = 3^\circ$, d) $\alpha = 4^\circ$, e) $\alpha = 6^\circ$, f) trailing-edge C_p , $\alpha = 4^\circ$	148
6.5	Trailing-edge stagnation streamline from BOS measurements, a) untripped airfoil, b) tripped upper surface, c) tripped lower surface, d) comparison of the three transition conditions.	149
6.6	Wake profiles of the clean airfoil under steady freestream, $x/c = 1.05$	150
6.7	Comparison of the displacement profile from BOS measurements and velocity profile from the wake survey, $\alpha = 2^\circ$, $x/c = 1.05$	151
6.8	Steady wake profile, $x/c = 1.05$, a) tripped upper surface, b) tripped lower surface.	153
6.9	Comparison of the steady wake profile for different transition conditions, a) $\alpha = 1^\circ$, b) $\alpha = 2^\circ$, c) $\alpha = 3^\circ$, d) $\alpha = 4^\circ$	154
6.10	Illustration of curve-fitting strategy for the experimentally measured wake data.	155

6.11	Comparison of the steady wake centerline at $x/c = 1.05$	156
6.12	Momentum balance of the shear layer near the steady airfoil trailing-edge.	158
6.13	Comparison of lift frequency response of the NACA 63-015A airfoil with different transition conditions, $\sigma = 0.21, k = 0.025, \alpha = 1^\circ$ a) normalized unsteady lift coefficient, b) unsteady lift coefficient.	160
6.14	Comparison of the normalized unsteady lift coefficient with different transition conditions, $\sigma = 0.21, k = 0.025$, a) $\alpha = 2^\circ$, b) $\alpha = 3^\circ$, c) $\alpha = 4^\circ$, d) $\alpha = 6^\circ$	161
6.15	Comparison of the unsteady lift coefficient with different transition conditions, $\sigma = 0.21, k = 0.025$, a) $\alpha = 2^\circ$, b) $\alpha = 3^\circ$, c) $\alpha = 4^\circ$, d) $\alpha = 6^\circ$	163
6.16	Comparison of the surging airfoil trailing-edge stagnation condition at $\alpha = 4^\circ$, a) untripped airfoil, b) tripped upper surface, c) tripped lower surface.	165
6.17	Wake profile of the NACA 63-015A airfoil during the surging freestream condition, $x/c = 1.05$	166
6.18	Dynamics of the unsteady wake centerline during the surging freestream, $\alpha = 1^\circ, x/c = 1.05$	167
6.19	Momentum balance of the shear layer at the near wake as a function of the angle of attack, a) untripped airfoil, b) tripped upper surface, c) tripped lower surface.	168
6.20	Momentum balance of the shear layer at the near wake as a function of flow transition, a) $\alpha = 1^\circ$, b) $\alpha = 2^\circ$, c) $\alpha = 3^\circ$, d) $\alpha = 4^\circ$	170
6.21	Effect of the unsteady trailing-edge stagnation condition to the unsteady lift response, clean NACA 63-015A airfoil, $\alpha = 4^\circ$	171
C.1	Schematic of the acoustic box experiment.	184

C.2	Transfer function of the pressure measuring system predicted by the B-T model under the tunnel operating condition, $M = 0.2$, $Re_c = 1,500,000$	185
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Chapter 1: Introduction

1.1 Unsteady Aerodynamics of Rotorcraft Blades

Vertical flight has attracted growing interest and attention in the past decade, both in the military and the commercial market. Different from fixed-wing aerodynamics, rotorcraft aerodynamics are highly unsteady and complex at cruising conditions, due to the rotation of the rotor blades. The most representative rotorcraft is the helicopter. When a helicopter is in forward flight, the rotor blade aerodynamics are highly unsteady [1]. In a two-dimensional point of view, one of the unsteadiness experienced by the rotor blade section is that the freestream velocity impinging on it varies in time as the blade section sweeps across the rotor disk, shown in Figure 1.1. The time-varying freestream velocity u_∞ relative to the blade is due to the composite effect of the vehicle forward speed and the rotational speed of the blade. As a result, the freestream velocity felt by the rotor blade section at a radial distance r is sinusoidal, which is the vector sum of the blade rotation Ωr and the forward flight velocity of the vehicle U_∞

$$\vec{u}_\infty(r, t) = \vec{U}_\infty + \vec{\Omega}r = U_\infty \sin \psi + \Omega r \quad (1.1)$$

where ψ is the azimuth angle of the rotor disk.

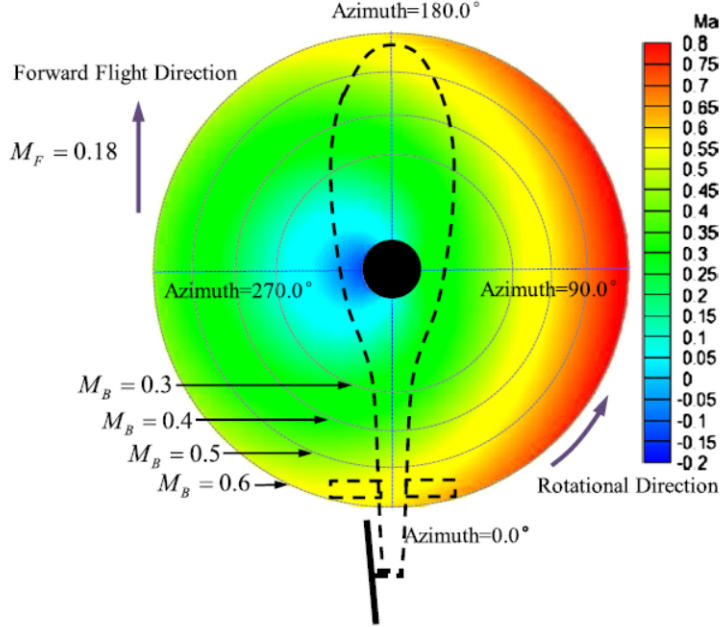


Figure 1.1: Illustration of Mach number across the rotor disk for a helicopter under forward flight (from Wang & Zhao 2016).

The resulting sinusoidal freestream velocity in the two-dimensional blade section view is comprised of a steady component due to the blade rotation and an unsteady component due to the contribution of the forward flight speed, written as

$$u_{\infty}(t) = U_0(1 + \sigma \sin(\omega t)) \quad (1.2)$$

where $U_0 = \Omega r$ is the mean velocity magnitude, σ is the advance ratio which represents the oscillation magnitude [1], ωt is the phase angle of the blade sweeping through the rotor disk. This harmonically varying freestream velocity is referred to as surging flow (or sinusoidal incident velocity [1]).

Another unsteadiness in rotor blade aerodynamics is the continuous pitching motion of the blade. This is designed to balance the lift generated on the rotor blade

across the rotor disk. Because lift is proportional to the freestream dynamic pressure, the oscillating freestream velocity will cause a stronger lift distribution around the advancing side (azimuth angle = 90°) and a weaker lift distribution around the retreating side (azimuth angle = 270°), leading to an undesired rolling moment on the vehicle [1]. The solution is to adapt the blade pitching angle as the blade rotates, where the pitching angle is lower around the advancing side and higher around the retreating side. Thus, a uniform lift can be produced on the whole rotor disk at trim conditions, whereas the cyclic pitch can change the orientation of the rotor thrust vector and is used for controlled rolling moment. In addition, such pitching motion can exceed the static-stall angle of the blade section and causes dynamic stall phenomena [1, 2, 3], which has been widely studied in the unsteady aerodynamics community.

These two unsteadiness are illustrated in Figure 1.2 and more unsteady behaviors are discussed in reference [1] such as plunging motion, transverse gust, lead-lag motion, etc. In addition, the rotor blade aerodynamics in real flight are three-dimensional and affected by rotor wakes and blade tip vortices. Overall, the unsteady aerodynamics of the rotor blade is very complex and various phenomena can interact with each other and not in a linear fashion. Such unsteady flows are not only encountered in helicopter blades, but also prevalent in horizontal and vertical axis wind turbines (HAWTs and VAWTs), and micro aerial vehicles (MAVs). As a result, the flow physics and loading of the blade section can be significantly altered compared to the steady and quasi-steady scenarios.

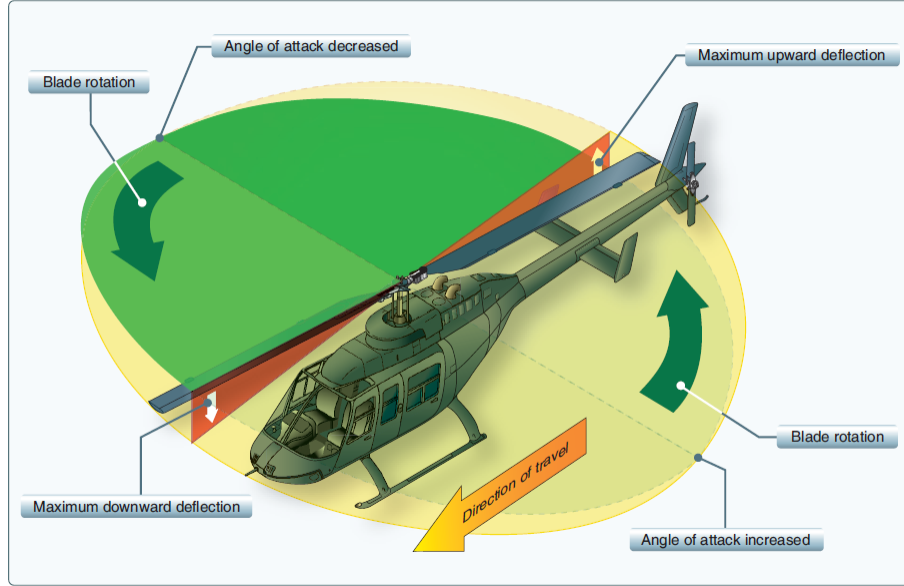


Figure 1.2: Illustration of various unsteady phenomena experienced by the helicopter rotor blade (Helicopter Flying Handbook, FAA).

1.2 Classic Unsteady Airfoil Theory

Because of the complexity of the unsteady aerodynamics of rotor blades in real applications, individual phenomenon is studied to aid in understanding its effects. Historically, reduced-order models have been developed to characterize these unsteady motions, such as surging, pitching, and plunging motions, and predict the resulting aerodynamic performance of the two-dimensional blade sections as early as in the 1930s. These reduced-order models are referred to as the classic unsteady airfoil theory.

In 1935, Theodorsen [2] analyzed the forces and the mechanism of airfoils under flutter (sinusoidal pitching and plunging motions) with potential flow analysis and the adoption of the classic Kutta condition for closure. The solution was derived with

Bessel functions of the first and second kind and of zero and first order. Later, Isaacs [4] extended Theodorsen's theory to airfoils at a constant angle of attack experiencing a sinusoidal freestream velocity, *i.e.*, surging flow. Isaacs' solution of the unsteady lift and pitching moment was in the form of a complete Fourier series and was later extended to include the effects of harmonically oscillating angles of attack [5]. Shortly after that, Greenberg [6] made a high frequency simplification about the shed wake vorticity and he derived a general solution for combined surge, pitch, and plunge motion.

In addition, the classic unsteady airfoil theory [2, 4, 5, 6] share the following assumptions: the flow is two dimensional, inviscid, irrotational, and incompressible; the airfoil is a thin flat plate at low angles of attack and flow is always attached; the airfoil generates a continuous non-deforming wake of trailing vortices that travel downstream with the freestream velocity and the classic Kutta condition is always satisfied. While these assumptions help simplify the problem for analytical derivation, it also limits these theories in the real applications of unsteady flow which are often three dimensional, compressible, turbulent with pressure gradient, and with strong viscous effects (*e.g.*, detached flow, reversed flow, shock boundary layer interaction). However, their solutions are focused on first order effects and should be treated as a basis to develop more realistic predictive models.

In 1990s, van der Wall and Leishman [3] revisited these theories and validated them with a numerical model based on Euler's equations. They concluded that Isaacs' theory is the only exact solution since it does not assume constant freestream velocity across the chord. They also concluded that all theories handled the oscillating freestream as a fore-aft moving airfoil in steady flow since they assumed uniform

convecting velocity of the shed wake vorticity. However, the difference between oscillating frequency and fore-aft motion are negligible at low reduced frequencies. In their review, they generalized Isaacs' theory about a fixed pitch axis at the airfoil quarter-chord for surging, pitching, and plunging motions. In addition, they were the first to express the lift frequency response of a surging airfoil in terms of normalized lift coefficients because the dynamic pressure dominates the unsteady forces and moments and can mask the important flow physics of the surging airfoil. The final form of the lift frequency response of a surging airfoil is expressed as

$$\frac{C_l(t)}{C_{l,qs}} = \frac{1}{(1 + \sigma \sin(\omega t))^2} \left[1 + 0.5\sigma^2 + \sigma(1 + \text{Im}(l_1) + 0.5\sigma^2) \sin(\omega t) \right. \\ \left. + \sigma(\text{Re}(l_1) + 0.5k) \cos(\omega t) + \sigma \sum_{m=2}^{\infty} (\text{Re}(l_m) \cos(m\omega t) + \text{Im}(l_m) \sin(m\omega t)) \right], \quad (1.3)$$

where $C_{l,qs}$ is the quasi-steady lift coefficient that depends on the steady lift and the square of the velocity, $l_m = -m(-i)^m \sum_{n=1}^{\infty} [F_n(J_{n+m} - J_{n-m}) + iG_n(J_{n+m} + J_{n-m})]$ is composed of series summations of the Bessel function and the Theodorsen function [2, 3, 7], σ is the advance ratio which describes the amplitude of the velocity fluctuation, and $k = \omega c/2u_{\infty}$ is the reduced frequency that depicts how fast the vorticity is shed in the wake [1, 3].

The lift frequency response of a surging airfoil predicted by Issacs' theory is illustrated in Figure 1.3, which highlights the effects of advance ratio σ and reduced frequency k . Figure 1.3 shows that there is lift undershoot ($\phi = 0 - 135^\circ$ and $\phi = 330 - 360^\circ$) when the freestream velocity is generally increasing and lift overshoot ($\phi = 135 - 330^\circ$) when the freestream velocity is generally decreasing. The maximum lift undershoot is at approximately $\phi = 20^\circ$ and the maximum lift overshoot is at approximately $\phi = 250^\circ$. As the advance ratio or the reduced frequency increases, the

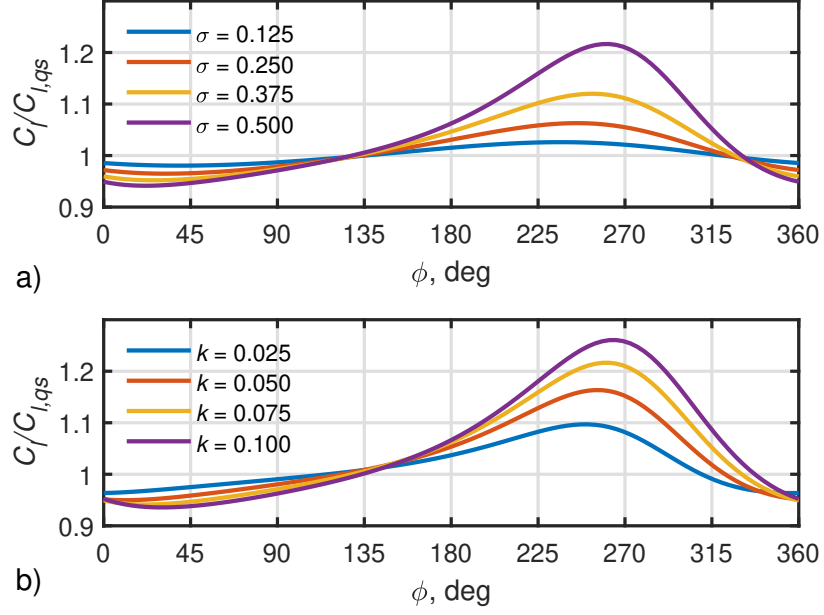


Figure 1.3: Lift frequency response predicted by Isaacs' theory, a) effect of advance ratio σ , b) effect of reduced frequency k .

lift overshoot and undershoot increase in magnitudes and the phases shift to higher ϕ . This is explained as a result of shed vorticities in the wake, see Figure 1.4.

When the freestream velocity increases, the bound circulation of the airfoil is clockwise and increases. For conservation of circulation, a new counterclockwise vorticity is shed into the wake. The newly shed counterclockwise vorticity introduces a downwash to the bound vortex along the airfoil which reduces the effective angle of attack and decreases the lift coefficient. Vice versa, when the freestream velocity decreases, the circulation of the airfoil decreases and a new clockwise vorticity is shed into the wake. This clockwise vorticity introduces an upwash to the bound vortex along the airfoil which increases the effective angle of attack and increases the lift coefficient. In addition, the larger lift overshoot compared to the lift undershoot is

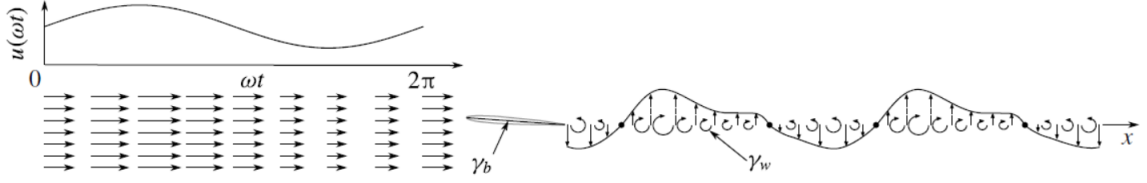


Figure 1.4: Schematic of the oscillating freestream, the shed wake vorticity, and the induced normal velocities (from Strangfeld *et al.* 2016).

due to the slower convective velocity of the shed vorticity at the retreating phases of the freestream velocity, thus the distance between the airfoil and the shed vorticity is shorter and the upwash effect is stronger than the downwash effect (Biot-Savart law [8]). In summary, the location and strength of the shed vorticity in the wake have a first-order effect on the induced velocity field about the airfoil in Isaacs' theory, which directly affects the circulation and lift. Thus, there is a two-way coupling between the lift generation and the shed circulation in the wake.

Since the development of the classic unsteady airfoil theory, numerous experimental and numerical studies have been conducted to investigate these flow phenomena. However, there are only limited experimental investigations of a fixed airfoil under sinusoidal freestream velocity. This is perhaps due to the challenges in simulating surging motion in the laboratory, especially to match rotorcraft flight conditions (*e.g.*, high Reynolds number, Mach number, advance ratio, and reduced frequency). Typical facilities built to produce surging flow conditions include low-speed wind tunnel utilizing a louver system downstream (or upstream) of the test section [9, 10, 11, 12], pressurized blow-down type subsonic wind tunnel with choke vanes downstream of the test section [13, 14], steady freestream water tunnel with traversing system to

oscillate the test article [15], and water towing tank in which the test article is driven in a quiescent water tank [16]. These facilities each have their advantages, disadvantages, and unique challenges [11, 16, 17]. In addition, a large number of parameters is involved in such unsteady motion which can affect the measurements and facility performance. For example, the chord Reynolds number Re_c , freestream velocity u_∞ or Mach number M_∞ , reduced frequency k , and advance ratio σ define the motion kinematics; the airfoil angle of attack α , thickness t/c , camber h/c , aspect ratio AR , and other geometric treatment (*e.g.*, boundary layer trip, leading edge and trailing edge shape) can also affect the flow physics. As a result, each facility has a different operating capability and comparisons of identical test conditions are scarce in the open literature.

Among limited surging airfoil investigations, Saxena *et al.* [9] and Fejer [18] may be the first group to conduct experiments on pure surging flow. Their studies were about harmonically oscillating freestream over a stationary NACA 0012 airfoil at the static-stall angle of attack under low Reynolds number. They found that dynamic stall due to the time-varying freestream can occur in a variety of forms depending on the frequency and amplitude of the oscillation. However, no direct load comparison was made to the classic unsteady airfoil theory. Graham [19] conducted similar experiments as Saxena *et al.* [9] with flow visualization. He observed a periodic vortex shedding near the leading-edge (LEV) convecting at about half the freestream speed which would affect the dynamic load of the surging airfoil. In 1990s, Krause & Schweitzer [20] studied the effects of reduced frequency and velocity amplitude to a NACA 4412 airfoil in surging flow at low Reynolds number. They observed periodic separation and reattachment of the flow and relaminarization of the boundary

layer. A growing interest in surging airfoil started in 2010s. Granlund *et al.* [10] investigated surging airfoil in two different ways, one fixing the airfoil in a wind tunnel that generates oscillating flow and the other oscillating the airfoil in a steady freestream water tunnel. Their study showed that surging flow investigations by these two different approaches are equivalent with appropriate corrections, as previously suggested by van der Wall & Leishman [3]. In addition, they reported the unsteady forces and moments under the velocity cycle. Choi *et al.* [21] also investigated a surging airfoil and discovered synchronization of wake instability and LEV at specific frequencies that augment the forces. Their study was focused on low Reynolds number and large angles of attack where the airfoil had been fully stalled. Strangfeld *et al.* [7] studied surging flow of a two-dimensional NACA 0018 airfoil at low angles of attack and Reynolds number. They may be the first group to compare the experimental data with Isaacs' theory in the form van der Wall & Leishman suggested [3]. Their measurements indicated two different lift frequency responses. One agreed reasonably well with Isaacs' theory in amplitude and phase but trailing-edge flow separation was present and modified the Kutta condition. The other deviated significantly from the theory and a mid-chord separation bubble was found to form and shed along the airfoil. Thus, they concluded that the location and shedding of the bubble is decisive in the lift frequency response. Their work can be considered as partial validation of Isaacs' theory at low Reynolds number. Recently, Gloutak *et al.* [22] measured lift frequency response of a wing section in an unsteady wind tunnel and explored effects of aspect ratio. Some deviations from Isaacs' theory in amplitude and phase over the test matrix were found and laminar separation bubble on the wing and flow three-dimensionality were believed to be the cause. Other experimental investigations

involving surging flow include those combined with pitching or plunging motions [23, 24, 25, 26, 27, 28], with compressibility effects [13, 26, 27, 28], and with active flow control techniques [28].

In addition, there have been many numerical and computational studies of surging airfoils. Jose *et al.* [29] developed an indicial based unsteady airfoil theory for surging and pitching motions in compressible flow. They validated this theory with a CFD solution based on Euler’s equations. Strangfeld *et al.* [30] compared their experimental results with a two-dimensional unsteady Reynolds-averaged Navier-Stokes (URANS) simulation. For pure surging flow at low angles of attack, they found reasonable agreement between Isaacs’ theory and their experimental measurements. However, discrepancies were present due to the inability to resolve wake vorticity by the URANS model. Wang & Zhao [31] conducted a URANS simulation of a SC1095 airfoil under surging and/or pitching motion at moderate subsonic Mach number. In the pure surging airfoil case, they found that the stability of the boundary layer was destroyed due to the effect of varying freestream velocity and a separated vortex was formed even at low angles of attack, low reduced frequencies, and moderate advance ratios. This significantly affected the lift and pitching moment of the airfoil. General agreement with Isaacs’ theory can be deduced, however, significantly higher lift deficit was observed when this vortex was near the trailing-edge. Kocher *et al.* [32] carried out a numerical investigation of surging airfoil utilizing large eddy simulation (LES). They compared results to cases of Granlund *et al.* 2016 [15] and Strangfeld *et al.* 2016 [7]. In conditions similar to reference [15], there was a good agreement in time-varying lift and a distinct LEV was shed on the suction side due to large angles of attack and advance ratios. In conditions similar to reference [7], good agreement

with experimental measurements in lift was observed and no distinct vortex was shed from the airfoil.

Among the existing experimental and computational studies on surging airfoils in the open literature, only a handful of them is about pure surging motion and even fewer investigations are aimed at validating Isaacs' theory in the appropriate form of the aerodynamic loads. In addition, the majority of these works are conducted at low Reynolds numbers and one or multiple assumptions in Isaacs' theory have already been violated (*e.g.*, flow separation and compressibility). As a result, there is still the need to fully validate Isaacs' surging airfoil theory.

1.3 Kutta Condition and the Controversy

Besides the prediction of unsteady loads based on Isaacs' surging airfoil theory, it is also important to understand the flow physics of a surging airfoil, especially the evaluation of the assumptions in the classic unsteady airfoil theory to understand the limitation of the model. The important assumptions have been given above. First, the theory assumes two-dimensional flow. This assumption simplifies the real complex rotor blade dynamics and can be easily simulated in laboratories. Thus, it helps to establish a fundamental analytical model and can be replicated for evaluation. Second, the theory assumes the flow is incompressible. In reality, airflow is always compressible, however, the compressibility effects can be neglected as long as the freestream velocity is low (*i.e.*, the freestream Mach number is lower than 0.3). If higher Mach number is targeted, an evaluation of the compressibility is required to modify the existing theory. Third, the airfoil is treated as a thin flat-plate so that it can be simulated by a vortex sheet. This assumption can be satisfied if a thin airfoil is

tested. However, certain applications, such as turbine blades or rotor sections near the hub, may require a thicker airfoil. In these scenarios, a correction of existing theories may be necessary. In addition, the theory assumes small disturbances where flow is always attached on the airfoil surface. This requires the experimental evaluation to focus on small angles of attack. Such assumptions limit the application of the theory and are often violated in existing studies listed above. Finally, the theory utilizes potential flow in which the flow is inviscid and irrotational, assumes a straight and flat non-deforming wake, and applies classic Kutta condition at the trailing-edge. Potential flow is a powerful analytical tool, however, viscous effects are important in real environments as the Reynolds number can be small, such as helicopter rotor blade in the retreating side near the hub or rotor blades of MAVs. As such, a key component of flow physics is neglected. Some simple viscous corrections may be added, such as including the displacement thickness in a panel code [33]. However, unsteady viscous effects may be more challenging to be accurately included in a low-order model. The straight and flat non-deforming wake is also a viscous effect, and effects of the wake are largely unknown in existing unsteady airfoil investigations. However, it is reasonable to deduce that the wake does not always satisfy this assumption since the wake is a viscous phenomenon and is affected by the airfoil's angle of attack (pitching motion) and the freestream or local velocity (surging motion). At last, the classic Kutta condition is utilized to close the potential flow solution of an airfoil. Thus, it is also neglecting any viscous effects. In addition, the Kutta condition is only a convenient solution based on observation rather than a well-rounded mathematical derivation, and its validity for unsteady flows is not fully understood.

In conclusion, the last three assumptions listed above are all related to viscous effects, which cannot be eliminated in laboratories or real world application. Among them, the adoption of the classic Kutta condition in an unsteady flow is most controversial [34, 35]. As pointed out above, the classic Kutta condition is an empirically based closure mechanism that narrows down an infinite number of potential flow solutions on the airfoil to a single solution that predicts steady airfoil lift extremely well. As a result, the classic Kutta condition forces the rear stagnation point to be at the trailing-edge of an airfoil. In addition, enforcement of the Kutta condition implies that - for a steady flow - there is no pressure jump between the upper and lower surfaces at the trailing-edge (*i.e.*, continuous pressure), the velocity at the trailing-edge is finite or zero, and no vorticity is shed aft of the airfoil [34, 36]. For steady flow, the implication of these conditions is that the trailing-edge stagnation streamline must bisect the airfoil trailing-edge angle or be parallel to a cusped trailing edge.

The classic Kutta condition is fundamental to the development of the classic unsteady airfoil theory which utilizes the potential flow theory. Interestingly, however, lift generation and shedding of vorticity in the wake are inherently viscous phenomena. In addition, the shedding of circulation in the wake originates from the boundary layer, thus, time-varying boundary layer behavior can potentially allow for a pressure discontinuity at the trailing-edge [36, 37, 38]. Thus, Isaacs' simultaneous imposition of the classic Kutta condition and the shedding of vorticity in the wake is an inherent contradiction. However, robust definition of an unsteady Kutta condition is much more ambiguous. Giesing [39] proposed a model of the unsteady Kutta condition that flow streamlines could be tangential to either the upper or the lower surface of the airfoil at the trailing-edge depending on the direction of the shed vorticity,

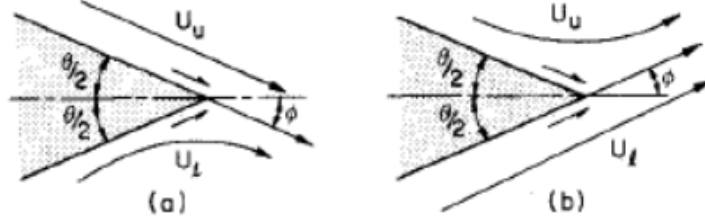


Figure 1.5: Schematic of possible trailing-edge streamlines under oscillatory freestream (from Poling & Telionis 1986), a) at decelerating phases, b) at accelerating phases.

see Figure 1.5. However, the difficulty of this model is that it does not converge on the classic Kutta condition as the unsteady frequency goes to zero. Recently, Xia & Mohseni [37] pointed out that the direction of the flow exiting the trailing-edge depends on the boundary layer momentum balance of the upper and lower surfaces, see Figure 1.6. They developed a free vortex model that models the shedding and convection of vorticity in the wake in a pitching airfoil and showed good comparison with computational simulations. Taha & Razaei [38], on the other hand, adopted the triple-deck theory [40] and relaxed the Kutta condition with viscous correction to resolve deviations of pitching airfoil data from Theodorsen's theory.

Furthermore, there is a building body of experimental evidence that the classic Kutta condition is not applicable in an unsteady flow. McCroskey & Philippe [41] found that the boundary layer shear stress and displacement thickness on a flat plate vary significantly in amplitude and phase in an oscillating freestream compared to the quasi-steady condition. The behavior is found to depend on Reynolds number, reduced frequency, and whether flow is laminar or turbulent. Satyanarayana & Davis

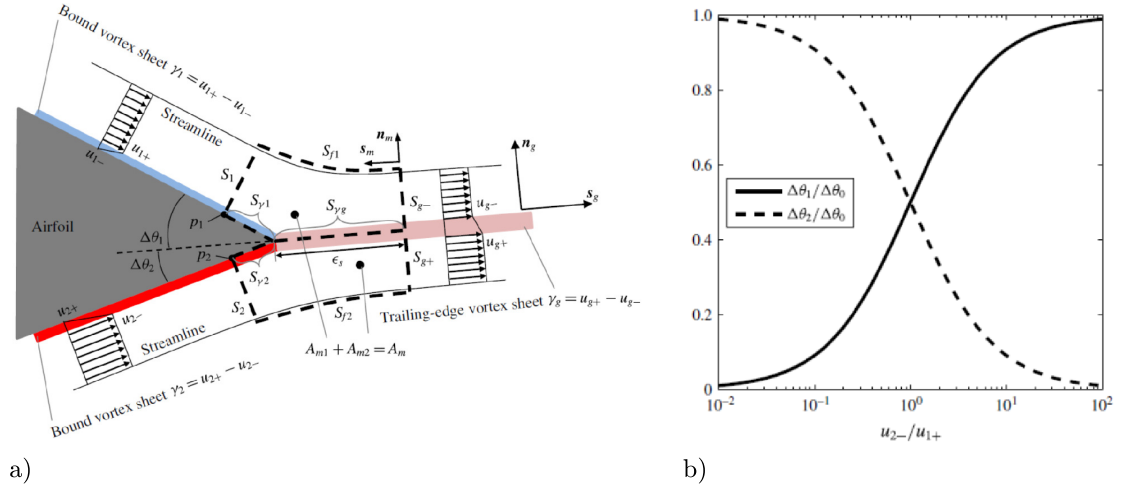


Figure 1.6: Free vortex model by Xia & Mohseni 2017, a) schematic of the free vortex sheet for a finite-angle trailing-edge, b) the angle of the trailing-edge stagnation condition as the ratio of boundary edge velocities.

[42] conducted experiments on a pitching airfoil to study unsteady trailing-edge condition. They found that deviations of pressure differentials and phase angles at the trailing-edge start to increase with reduced frequency above 0.6. Poling & Telionis [36] utilized Laser Doppler Velocimetry (LDV) to measure the trailing-edge stagnation streamline of a fixed-pitch airfoil immersed in the wake of a pitching airfoil. They found that the wake centerline of the fixed-pitch airfoil deflects dynamically in a region that is totally immersed in the viscous layers immediately downstream of the trailing-edge. Liu *et al.* [43] performed hot-wire and LDV measurements on the trailing-edge of a pitching NACA 0012 airfoil and measured deflection of the trailing-edge stagnation streamline in the unsteady flow. More similar works about unsteady trailing-edge stagnation condition can be found in the open literature [44, 45, 46, 47, 48, 49, 50].

Throughout the 1970s and 1980s, there was a fair amount of attention to the controversy regarding the exact nature of the Kutta condition in an unsteady flow, especially for pitching motions. Various studies showed pressure discontinuities at the trailing-edge or stagnation streamlines that did not stay fixed in space and bisect the trailing-edge for airfoils subjected to high-frequency or high-amplitude unsteady flows. However, consensus seemed to form around the idea that the classic Kutta condition was apparently satisfied in low frequency, low amplitude, and high Reynolds number experiments [51]. Yet, how trailing-edge stagnation condition interacts in surging airfoils and affects the aerodynamic response compared to classic unsteady airfoil theory is still largely unknown.

1.4 Motivations

In summary, there is still a lack of experimental validation of Isaacs' surging airfoil theory across the wide range of parameters $(Re_c, M_\infty, k, \sigma)$, specifically the load evaluation in the form of normalized unsteady lift coefficient. In addition, there are controversies of the assumptions in Isaacs' theory where the validity of the classic Kutta condition on a surging airfoil has not been fully understood. In recent experimental investigations, there have been reported cases indicating Isaacs' surging airfoil theory failed to predict the lift frequency response [7, 52]. As a result, this investigation is aimed at experimentally validating or refuting Isaacs' surging airfoil theory at Reynolds number on the order of 10^6 , and experimentally investigating the unsteady flow physics around the airfoil trailing-edge. In order to do this, multiple experimental techniques were utilized to characterize the unsteady pressure distribution, the trailing-edge flow field, and the near-wake characteristics of a two-dimensional

surging airfoil. The following chapters include a description of the experimental and computational methods for the surging flow investigation, the modification to a pressurized blow-down type wind tunnel for oscillating freestream and a thorough characterization of its dynamics, a detailed explanation of the background-oriented schlieren method for local shear layer measurements, a comparison between Isaacs' surging airfoil theory and experimental measurements of a NACA 0018 and a NACA 63-015A airfoil in terms of normalized unsteady lift coefficient, and investigations of a surging NACA 63-015A airfoil trailing-edge flow physics with near-wake surveys to form an explicit trailing-edge stagnation condition that include viscous and unsteady effects.

Chapter 2: Methodology for Surging Airfoil Investigations

This chapter describes the experimental facility, setup, and methods to investigate the surging airfoil. Details include the introduction of the wind tunnel facility, the test article, the optical measuring techniques, the traversing wake rake, and the computational model.

2.1 Wind Tunnel Facility

Unsteady facilities involving oscillating freestream conditions are relatively uncommon due to many practical challenges present in such scenario. In general, two types of strategies are applied in the laboratory to simulate surging flow, either by creating an oscillatory flow impinging on a stationary test article or by dynamically moving a model in a steady flow environment. A few exemplary facilities are found throughout the literature. Greenblatt [11] recently designed an unsteady low-speed wind tunnel suitable for surging flow study. This wind tunnel is equipped with a blowing fan upstream of the plenum and a louver system of flat plates downstream of the test-section. By controlling the fan RPM and louver motion, surging flow can be created in the test-section with various mean speed, oscillating amplitude, and frequency. However, the test condition is limited to incompressible flow and low Reynolds number. A similar wind tunnel is utilized by Gloutak *et al.* [22] where the

louver system is made of NACA 0015 airfoils and the test-section can be switched between open and closed configurations. These are only a couple of examples of wind tunnels designed for oscillatory freestream experiments and a rotating louver system is the common mechanism for such purpose. Water tunnels, on the other hand, have also been adopted for surging flow investigations and higher Reynolds number can be achieved due to higher density of water. However, due to the greater inertia of water, the approach is switched to driving the test article with a fore-aft motion under a constant freestream [15]. An advantage of this type of facility over the wind tunnel is that reversed flow can also be achieved. Unlike the ones mentioned above, Jentzsch *et al.* [16] developed a large water towing tank where the test article can be pulled through the quiescent water at time-varying velocity. This facility has the advantage of omitting typical challenges in a wind tunnel, such as acoustic resonance, cut-off frequency, and pressure losses, but other challenges such as surface waves, rig vibration, cavitation, and long downtime are present [16]. Despite various facilities are utilized for surging flow investigations, they each hold certain advantages and disadvantages over the other, highlighting the practical challenges in the experimental efforts of this area of research.

The facility used in the current investigation is the Ohio State University (OSU) 6-inch $\times$ 22-inch unsteady transonic wind tunnel, located at the university's Aerospace Research Center (ARC). This is a pressurized blow-down type wind tunnel that possesses a unique capability of oscillating the angle of attack of the test article and oscillating the freestream Mach number, either independently or synchronously. Because of the blow-down nature, high Reynolds number and compressible flow can be

achieved in this facility. The schematic of the wind tunnel is given in Figure 2.1. Compressed air is supplied to the tunnel through a 0.2m diameter supply line from two 42.5m³ air storage tanks that can be pressurized up to 17 MPa with in-line air dryers to control condensation. The high-pressure air flow is controlled by three valves. The first valve (closer to the downstream) is a control valve which sets the total pressure downstream by restricting the mass flow rate through a specified percentage opening. As a result, the flow density and Reynolds number are established by the stagnation pressure and temperature in the test section. The second valve is a pneumatically controlled fast-acting valve used to start and stop the tunnel operation. Further upstream there is a hand valve which is not included in the schematic. The wind tunnel was designed as a low-turbulence facility with the primary goal of matching flight conditions in the experiments. To accomplish this objective, the settling chamber is equipped with a perforated plate, a honeycomb section, and eight screens (60 mesh) to lower turbulence to acceptable levels under nominal steady flow conditions (less than 0.5% for $M = 0.4$ [14]). Downstream of the settling chamber, a subsonic nozzle with a contraction ratio of 15:1 further establishes flow uniformity in the 0.15m $\times$ 0.56m (6-inch $\times$ 22-inch) test section. The span-wise floor and ceiling walls in the test section are perforated with 3.2mm straight holes, yielding an effective porosity of 6%. These isolation cavities are open to the flow only downstream of the test section and aid in producing a high-quality flow in the test section by reducing Mach wave reflections when the flow is transonic. In addition, the test section has three circular cutouts for optical access, one on each sidewall and the third one on the ceiling. Circular optical windows made of plexiglass allow optical measurements. The sidewall optical windows also serve as the support for the test article, such as an airfoil. As a

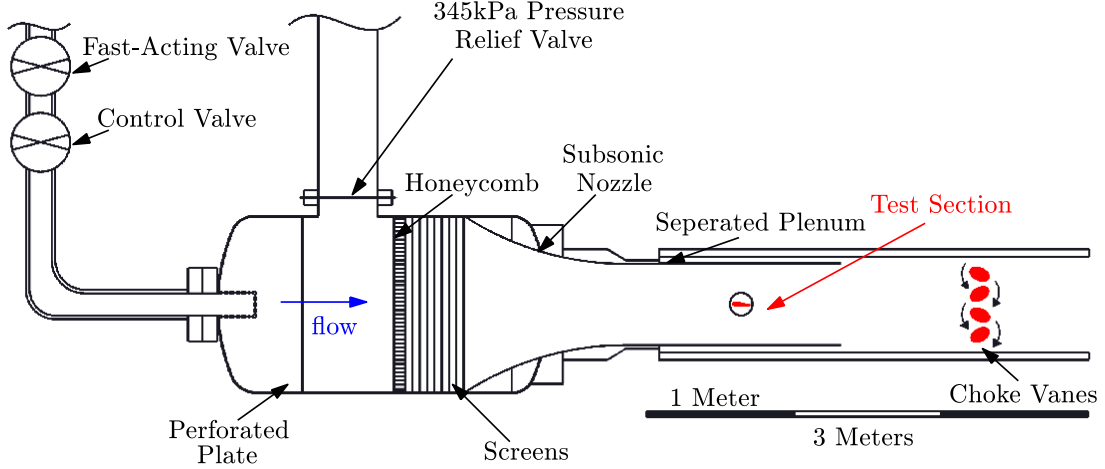


Figure 2.1: Schematic of the blow-down type unsteady wind tunnel.

result, different optical windows can be switched for different test articles or optical measurements.

Further downstream, blockages are added to choke the tunnel. Typical blockages are circular choke bars for steady flow experiments. With flow choked at the throat, the flow Mach number at the test section is determined by the area ratio between the test-section and the throat,

$$\frac{A_{ts}}{A^*} = \left(\frac{\gamma + 1}{2} \right)^{\frac{\gamma+1}{2(\gamma-1)}} \frac{\left(1 + \frac{\gamma-1}{2} M^2 \right)^{\frac{\gamma+1}{2(\gamma-1)}}}{M} \quad (2.1)$$

where A_{ts} is the cross-sectional area of the test section, A^* is the cross-sectional area of the throat, γ is the specific heat ratio of air, and M is the Mach number of the flow in the test section.

Because the freestream Mach number is determined by the area ratio between the test-section and the throat, the chord Reynolds number can be independently varied as long as the tunnel is choked and the operating stagnation pressure is below the

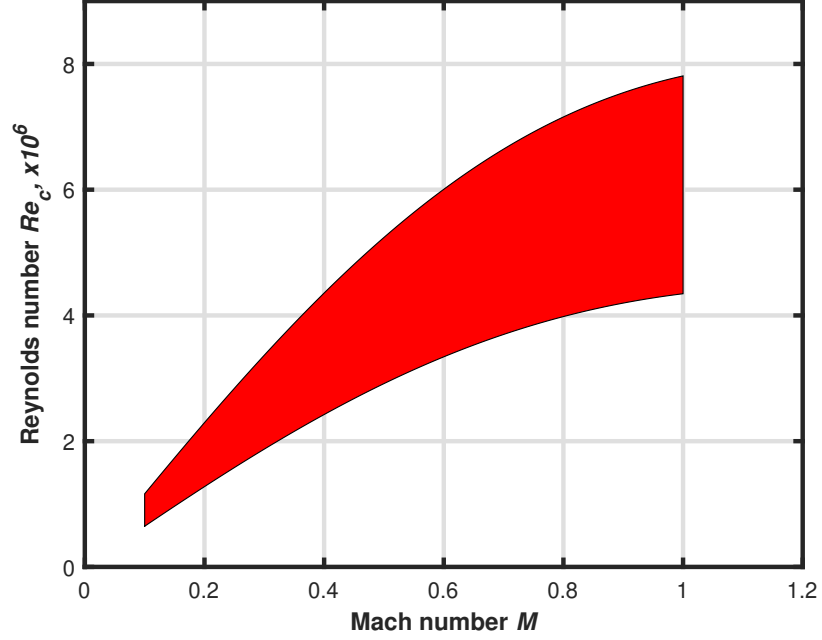


Figure 2.2: Theoretical operating range of the wind tunnel evaluated at the room temperature and a characteristic length of 15.24cm.

safety margin (345kPa or 50psia). A total of 15 circular choke bars with different diameters can be installed to modulate the throat area, and the operating range of the flow condition in terms of Mach number and Reynolds number based on an airfoil chord length of 15cm (6-inch) is shown in Figure 2.2.

2.2 Test Articles

A few different airfoils were tested during the whole experimental campaign, but two airfoils were the primary focus of the detailed surging flow investigation. They are the NACA 0018 airfoil and the NACA 63-015A airfoil. The NACA 0018 airfoil was selected in order to compare with the surging airfoil investigation by Strangfeld *et al* [7], whereas the NACA 63-015A airfoil is a natural-laminar-flow airfoil in order to

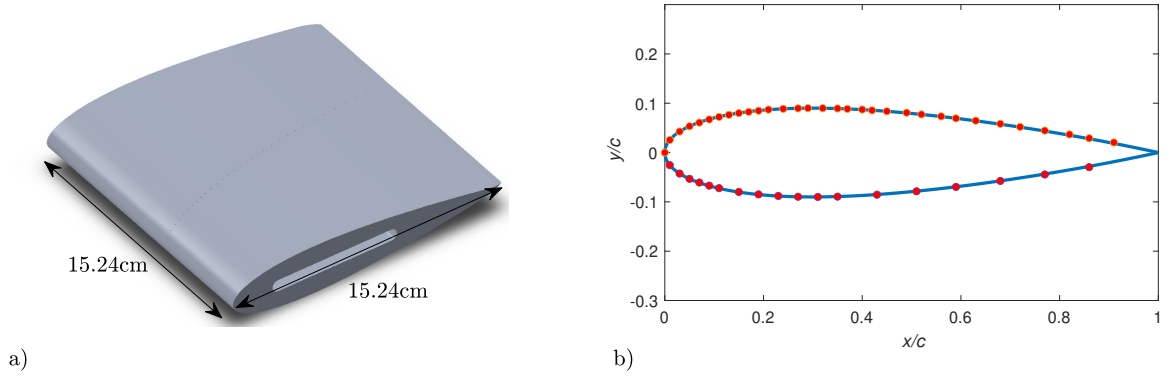


Figure 2.3: Model of the NACA 0018 airfoil, a) isometric view, b) surface pressure tap distribution.

investigate the effect of flow transition. Both airfoils are symmetric, which eliminates the effect of camber that can add to the lift coefficient.

Both airfoils have a span b and a chord c equal to 15.24 cm, resulting in aspect ratio $AR = 1$. The model and surface pressure tap design of the two airfoils are shown in Figure 2.3 and Figure 2.4, respectively. Both airfoils were prototyped by stereolithography (SLA) with an acrylonitrile-butadiene-styrene (ABS) like material. They were made by 3D SYSTEMS in high resolution and the surface roughness was found to be comparable to machined metal airfoils. In addition, these prototyped airfoils were an order of magnitude more economical than traditionally machined models, in terms of cost and lead time.

For the NACA 0018 airfoil, a total of 50 pressure taps was constructed for surface pressure measurements, with 31 on the suction surface, 18 on the pressure surface, and 1 at the leading-edge. Each surface tap has a diameter of 0.4mm. In order to acquire accurate surface pressure, the pressure taps were designed to have a greater spatial resolution towards the leading- and the trailing-edge. However, the physical

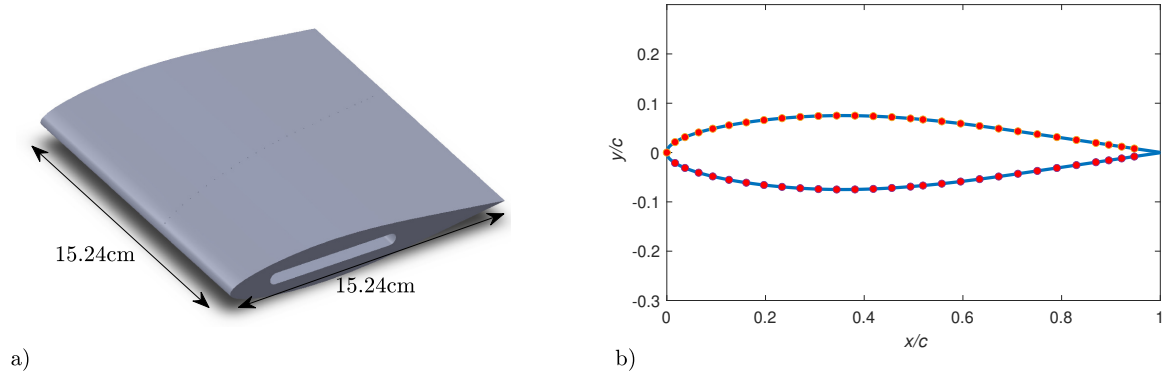


Figure 2.4: Model of the NACA 63-015A airfoil, a) isometric view, b) surface pressure tap distribution.

space available in these regions of the airfoil still limits the number of pressure taps. In addition, pressure taps were staggered by 15° along the chord to minimize interference from taps upstream. The aft-most pressure taps were located at $x/c = 0.90$ and $x/c = 0.86$ on the suction and the pressure surface, respectively, due to the limited space available near the trailing-edge. During experimental measurements, the trailing-edge pressure at $x/c = 1$ was interpolated based on pressures upstream. These designs are shown in Figure 2.3 and more details of the pressure tap locations are given in Appendix A.

For the NACA 63-015A airfoil, a total of 57 pressure taps were symmetrically distributed on the suction and pressure surfaces. The physical trailing-edge surface tap was also not available due to the limited thickness. As a result, the aft-most surface tap was located at $x/c = 0.95$, and the trailing-edge pressure was also interpolated in the same way described above. Figure 2.4 shows the airfoil model and the surface pressure tap distribution. Additional information can also be found in Appendix A.

The airfoil surface pressures were measured by two ESP 32HD pressure scanners connected to the taps via flexible tubing of 1.4mm diameter and approximately 20cm in length. The ESP pressure scanners were connected to a DTC Initium data acquisition unit, with the multiplexed signals sampled at a rate of 1kHz. More details of the pressure measuring system are given in Chapter 3.

2.3 Optical Measurements

2.3.1 Background-Oriented Schlieren Technique

The background-oriented schlieren (BOS) method is an optical density visualization technique that measures the deflection of light rays caused by changes in the local index of refraction [53]. Similar to the traditional schlieren technique, the first spatial derivatives of density are measured by the BOS method. However, the BOS method has advantages over the requirement of fewer optical components and the capability of quantifying the density field using a Poisson solver [54]. The general setup of the BOS method includes a volume of the target flow field, a speckle pattern that is behind the flow field, a light source that illuminates the background speckle pattern, and an imaging system that looks through the flow field and focuses on the background pattern. When density gradients are present in the flow field, light rays are refracted and the background speckle pattern of a wind-on image is displaced relative to that of a wind-off reference image [55]. Thus, the displacement of the speckle pattern in the image plane is an indicator of the density gradient as described by

$$\Delta y = f \frac{Z_D}{Z_D + Z_A - f} \frac{G}{n_0} \int \frac{\partial \rho}{\partial y} dz \quad (2.2)$$

where Δy is the displacement in a given direction in the image plane, f is the focal length of the lens, Z_D is the distance between the center of the fluid volume and the

background pattern, Z_A is the distance between the center of the fluid volume and the lens, G is the Gladstone-Dale constant, n_0 is the index of refraction of the fluid at reference density, and z is the depth through the fluid volume along the light path. Details of the derivation were given by Raffel [54] and Richard & Raffel [55]. Since the development of the BOS method, it has been widely used in various flow field studies ranging from incompressible to supersonic flow regimes, both in the laboratory and flight tests, due to the unique advantages of a convenient setup and the possibility of acquiring high spatial resolution data. A recent review of the BOS developments and applications is given by Raffel [54].

The BOS setup in the current investigation is shown in Figure 2.5. A high-speed Phantom 1210 complementary metal oxide semiconductor (CMOS) camera with a Nikon lens of 200mm focal length was placed on one side of the wind tunnel, 0.47m from the spanwise center of the test section. Frames were sampled at 500Hz, with a spatial resolution of 1280×800 pixels. A speckle pattern with 0.3mm dots and 0.1mm spacing (edge-to-edge distance between dots) was positioned on the opposite side of the test-section and back-illuminated with a 14000 lumen light-emitting diode (LED) array. A wind-off reference image was acquired before each blow-down of the wind tunnel, which was cross-correlated with wind-on images in DaVis 8 (commercial package from LaVision). The cross-correlation was multi-passed with a 16×16 interrogation window with overlap ratio of 0.75. This imaging system was synchronized with the pressure measurements with a TTL pulse train. With this setup, flow fields along the span of the airfoil were investigated, particularly surrounding the trailing-edge of the airfoil in the current study.

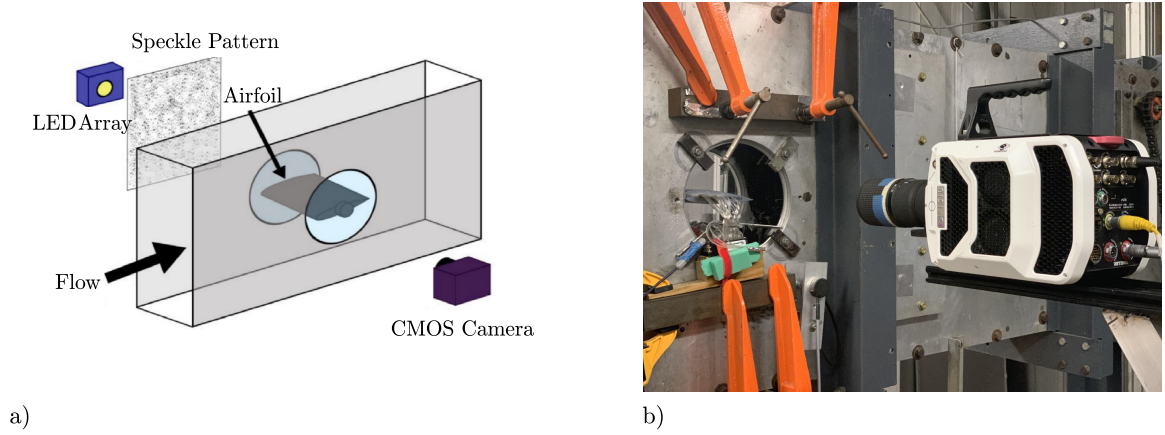


Figure 2.5: Background-oriented schlieren technique, a) schematic showing each component in a BOS experiment, b) setup of the current BOS wind tunnel experiment.

2.3.2 Infrared Thermography

The other optical method utilized in this investigation was thermal imaging. An Electrophysics Silver 420M infrared (IR) camera was mounted on top of the wind tunnel. This camera utilizes a photovoltaic indium antimonide (InSb) semiconductor focal-plane array which requires cryogenic cooling to maximize the signal to noise ratio. InSb has a high quantum efficiency of 0.9, which is an indicator of the number of carrier electrons excited per incident photon. The array is optimized for the frequency spectrum between 3 and 5 micrometers and was configured to record 100 frames per second with a spatial resolution of 320×256 pixels. The InSb sensor returns an intensity level for each pixel during recording. The intensity is converted to an electric unit in the operating software FLIR and can be converted to temperature, whereas temperature is calculated based on calibration equations built into the software. The calibration relation can be altered by changing user-supplied variables such as surface emissivity and ambient temperature. In addition, this camera can accept lenses of

varying focal length of various filters to optimize transmission of the desired frequency spectrum. The array has a sensitivity of 0.02 Kelvin with a bias uncertainty of ± 2 Kelvin. The FLIR software provides dynamic camera control such as the focus of the lens and can export both videos and still frames acquired by the camera. The acquired frames can be post-processed in MATLAB for further analysis.

The physical setup is indicated in Figure 2.6 and is similar to that in reference [56]. The camera was mounted on top of the wind tunnel test section, aligned with the optical access on the tunnel ceiling where an IR-transmitting window was made of sapphire crystal. For this application, a wide optical transmission band was attained with the sapphire crystal window which transmits 95% of infrared radiation. Since relatively short transmission distances were involved, atmospheric absorption, humidity, and pollutant levels were negligible. With this setup, the IR camera measured the temperature field of the airfoil upper surface where it was intended to capture flow transition due to different convective heat transfer effects between laminar and turbulent boundary layer.

2.4 Traversing Wake Rake

In addition to surface pressure and optical measurements, a traversing wake rake was designed and installed in the wind tunnel to investigate the flow near the vicinity of the airfoil trailing-edge. Figure 2.7 shows the schematic of the traversing wake rake installed in the wind tunnel test-section. The whole design consists of three major components which include the rake body inside the wind tunnel, the structure holding the rake and enclosing the test-section, and the traversing system that drives the rake.



Figure 2.6: Setup of the infrared camera for airfoil surface temperature measurements during the wind tunnel experiment.

The rake body was made of several stainless steel pieces and is illustrated in more details in Figure 2.8. Two circular tubes with outer diameter of 1.27cm are connected with a coupling. The coupling is used to compensate the relative streamwise location of the airfoil and the hot-film probes for wake survey, and provides additional degree of freedom in streamwise sampling; whereas the two tubes are designed to minimize blockage and house the cables. In addition, a fairing was added to the tube fully immersed in the test-section to minimize vibrations of vortex shedding over the rake. Four Dantec hot-film probes (model 55R01) can be installed onto the rake. The sensing element is made of quartz fibers with a $0.5\mu m$ nickel coating. In addition, the sensing element has a diameter of $70\mu m$ and is 1.25mm long. These hot-films can be operated at maximum sensor temperature of $300^{\circ}C$ or at an ambient air temperature up to $150^{\circ}C$. The maximum measured air velocity is 350m/s and the maximum frequency limit is 175kHz. Three of these probes are closely staggered

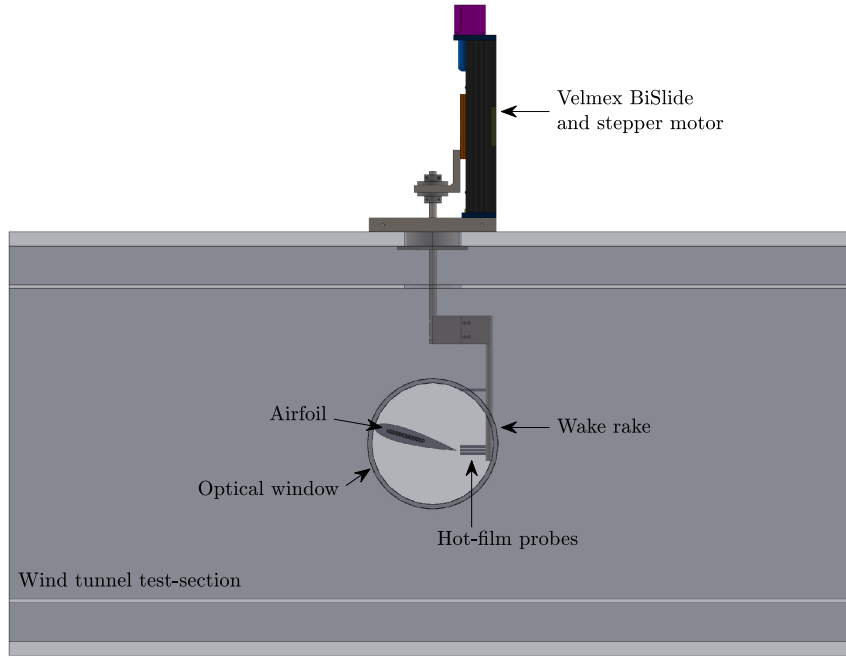


Figure 2.7: Schematic of the traversing wake rake during wind tunnel experiments.

(0.25-inch to each other) to measure the wake flow and the fourth probe is located higher along the rake (4-inch away) to provide reference measurements away from the airfoil's sphere of influence (where the local flow is disturbed because of the presence of the airfoil). These hot-film probes were connected to a Dantec multi-channel CTA. Proper circuit settings were applied with a choice of sensor temperature equal to 250°C (overheat ratio equals to 1) for best sensitivity. A sampling rate of 10kHz was chosen due to the limited computational power (RAM available in the data acquisition computer) when multiple signals were recorded at the same time.

On top of the wind tunnel, several parts were used to seal the wind tunnel and provide support for the traversing wake rake components. A sleeve bearing is used



Figure 2.9: Traversing wake rake installed in the test section for wind tunnel experiments.

the traverse is 0.0015-inch (38.1 microns) and the resolution is 0.00025-inch (6.35 microns).

With the wake rake installed, additional blockage was introduced in the wind tunnel. The maximum blockage ratio when the wake rake is fully extended into the test section is about 5%, doubling the blockage ratio when there is only the airfoil (*e.g.*, NACA 0018 at 10 degrees angle of attack). This blockage ratio is considered acceptable in the current investigation. The installed traversing wake rake system inside the unsteady wind tunnel is shown in Figure 2.9.

The traversing wake rake can be controlled by the stepping motor controller, the COMSOL software, or a LabVIEW VI. For each operation, the wake rake is first moved to a dedicated position (typically the trailing-edge of the airfoil) and zeroed

as the reference starting point. Then, based on dedicated travel distance and speed, it will traverse through the designed test matrix during the wind tunnel operation. The wake survey is designed to measure the near wake or very near wake flows and the hot-film tips are located approximately 5% chord downstream the airfoil trailing-edge. The thickness of the wake was found to be approximately 10% chord and a total of 13 traversing locations (39 probe sampling locations) was traveled for each angle of attack.

In addition, two surveying strategies were evaluated and utilized between the steady and oscillatory freestream experiments. For steady freestream experiments, the rake is traversed through the entire wake from top to bottom over a 30s wind tunnel operation, where it stays at each height for a maximum of 2s. This allows a complete wake survey with one blow-down experiment. On the other hand, during oscillatory freestream experiments, the rake was fixed at a designated position and a total of 13 blow-down runs were conducted for a complete wake survey. This allows sufficient data for phase averaging in each sampling location. However, in either strategy, there is an uncertainty in flow Reynolds number (continuous decline during a steady blow-down experiment and repeatability between unsteady experiments). A maximum of 3% variation was found in either scenario. To mitigate such uncertainty of the measured wake parameters, the wake velocity data are always normalized by the freestream velocity in each experiment. Figure 2.10 shows a comparison of the wake profile in steady freestream experiments utilizing the two strategies. The results show a maximum of 3% difference.

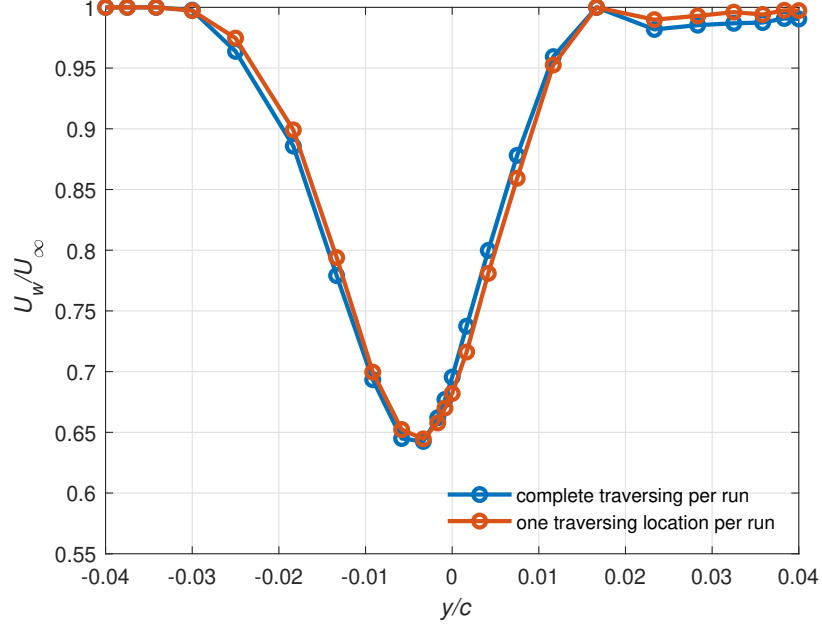


Figure 2.10: Comparison of wake survey strategies from a steady wind tunnel operation.

2.5 Computational Model

Finally, a computational model utilizing RANS equations was constructed to compare with experimental measurements and to provide additional insights of the flow field, specifically, to compare with BOS measurements in terms of density gradient in the flow field. Figure 2.11 shows the two-dimensional computational domain that extends $15c$ upstream and $20c$ downstream of the airfoil. This is a C grid with fully structured mesh. The grid was more refined towards the airfoil surface and around the airfoil wake in order to capture detailed flow structures of the viscous region. The first layer of mesh around the airfoil satisfies the condition of $y^+ = 1$.

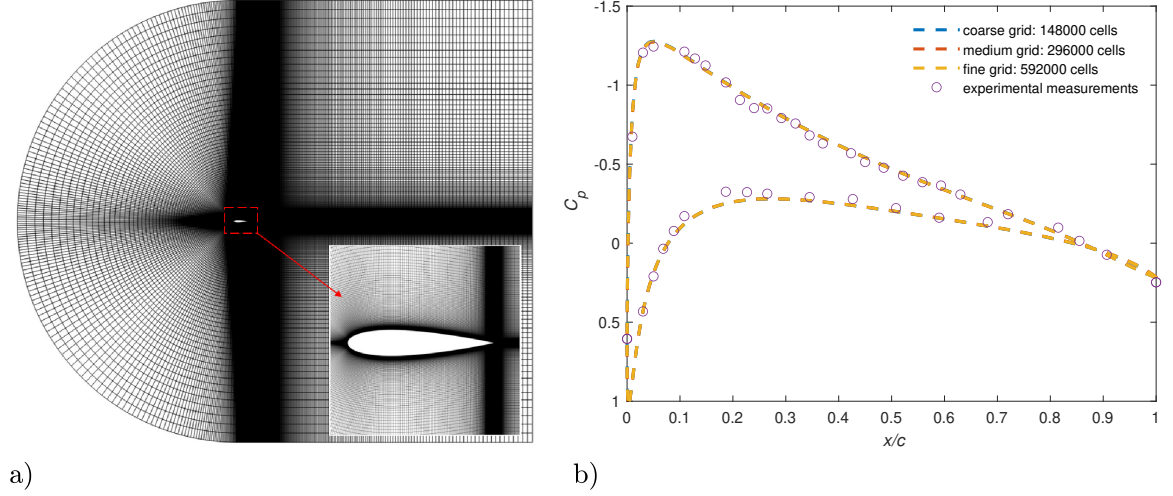


Figure 2.11: Computational model, a) the computational domain for NACA 0018 airfoil, b) the grid independence study based on distribution of the pressure coefficient C_p .

The RANS simulation was performed using Ansys Fluent with pressure-based solver, coupled scheme, double precision, and second order accuracy in spatial discretization. A least square cell based method was utilized for spatial gradients. The energy equation was calculated and the $k - \omega$ SST turbulence model was adopted. The viscous heating term was enabled and will be discussed in later chapters. The ideal gas law was selected for air with constant specific heat and thermal conductivity while Sutherland's equations were used for viscosity. Even though conjugate heat transfer was not included in the current investigation, the density, specific heat, and thermal conductivity of ABS-like material ($\rho = 1080\text{kg/m}^3, c_p = 1423.5\text{J/kgK}, k = 0.1946\text{W/mK}$) were applied to the airfoil surface. A velocity inlet boundary condition was applied at the inlet of the domain and a pressure outlet was assumed as the

boundary condition far downstream of the airfoil. The freestream pressure and velocity boundary conditions were set to match the experimental condition and the inlet turbulence intensity was considered as 1% (see Chapter 3) with a turbulent length scale of 15.24cm. A few different flow temperatures were chosen to create various temperature difference between the freestream flow and the airfoil surface while the airfoil surface satisfies the no slip condition and is at a constant temperature (*i.e.*, an isothermal surface).

A grid independence study is shown in Figure 2.11b. The grid was refined laterally to improve resolution normal to the airfoil surface. It is evident that the C_p distribution stays relatively constant after the medium grid size where the aerodynamic coefficients also reach uniformity. In order to conserve computational cost, the medium grid size simulations were considered for further analysis.

Chapter 3: Characterization of the Unsteady Wind Tunnel

This chapter describes the characterization of the OSU unsteady wind tunnel which is utilized to study airfoil performance under various unsteady motions, specifically, surging flow in this investigation. Understanding the characteristics of the flow generated in this wind tunnel including the operating limit of the surging flow condition, the measurement accuracy, and the flow qualities in the test section are vital for the subsequent investigation of the unsteady airfoil response. This chapter consists of the mechanism utilized to generate surging flow conditions, the experimental and numerical methods applied to characterize the wind tunnel, and results and discussions of the unsteady response of the wind tunnel.

3.1 Generation of Surging Freestream

The generation of unsteady flow in this pressurized blow-down type facility is following the same principle in the steady flow scenario. That is the inviscid area-Mach relation for choked flow. This theoretical basis is already shown as Equation 2.1, but assuming A^* is a function of time. Based on this principle, if the flow is quasi-steady a proper time history of A^* can generate a sinusoidal Mach number (and thus velocity) in the test section. So in order to generate a sinusoidal freestream Mach number, an inverse design process is made towards the time-varying area of blockage.

First, the target sinusoidal Mach number is defined in terms of the mean and the amplitude. Second, the time history of the area ratio is obtained based on Equation 2.1. Finally, if assuming the throat area changes in a uniform speed, the dimension of the throat area and the corresponding geometry of blockage can be defined.

From the inviscid flow analysis, it was found that a choke vane of elliptical shape rotating at constant RPM will generate a sinusoidal-like Mach number in the test section [14]. The superellipse shape was found to output even greater agreement with the target sinusoidal freestream, defined as

$$\left(\frac{x}{a}\right)^m + \left(\frac{y}{b}\right)^n = 1 \quad (3.1)$$

where x and y are Cartesian coordinates of the geometry, a and b are the semi-major and semi-minor axis of the superellipse, respectively, and m and n are positive numbers that define the curvature of the superellipse. When m and n are equal to 2, the geometry becomes an ellipse.

A set of four choke vanes based on this design was made to install in the tunnel throat and they are referred to as elliptical choke vanes in this investigation. The individual choke vane geometry for $M = 0.2 \pm 0.05$ is shown in Figure 3.1. The resulting Mach number and comparison to the desired sinusoidal Mach number is shown in Figure 3.2. The design parameters a , b , m , and n for best sinusoidal Mach number were found with an iteration algorithm written in MATLAB, see Appendix B. Figure 3.2 shows that for one rotation of the elliptical choke vanes two sinusoidal cycles in Mach number are generated. In addition, there are some deviations between the target and the resulting Mach number and the greatest disagreement occurs when the Mach number is near the minimum (which are maximum blockages at the throat and the major axis of the choke vane is perpendicular to the freestream flow). This

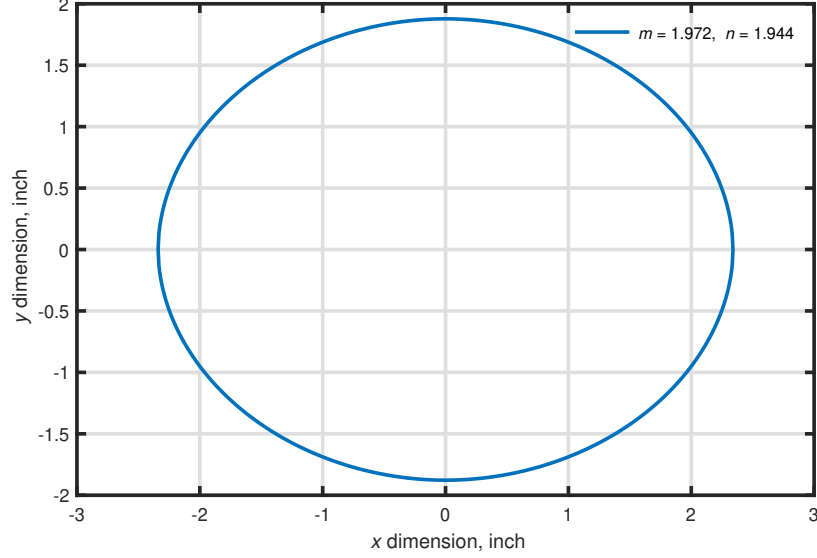


Figure 3.1: Dimension of the elliptical choke vane from the inverse design process.

is due to the constant RPM resulting in a shorter duration when the throat area is defined by the major axis. In other words, a sharp area change occurs at this instance which the analysis can not perfectly resolve. Nonetheless, the deviation is not too significant and this analysis presents a basis for generating unsteady flow in this blow-down type facility. A modification to improve this design may rely on controlling the RPM dynamically through the motor or an elliptical sprocket design, however, these modifications were found to be very challenging and costly and were not applied in the current work.

The set of four elliptical choke vanes was driven by a stepper motor, which was slaved to another 3.7kW motor. The stepper motor was only used for surging flow experiments and the 3.7kW motor was used for either surging flow experiments or pitch oscillation experiments [14, 28]. This configuration allows synchronization of the frequency of the two unsteadiness at any desired phase delay through a predetermined

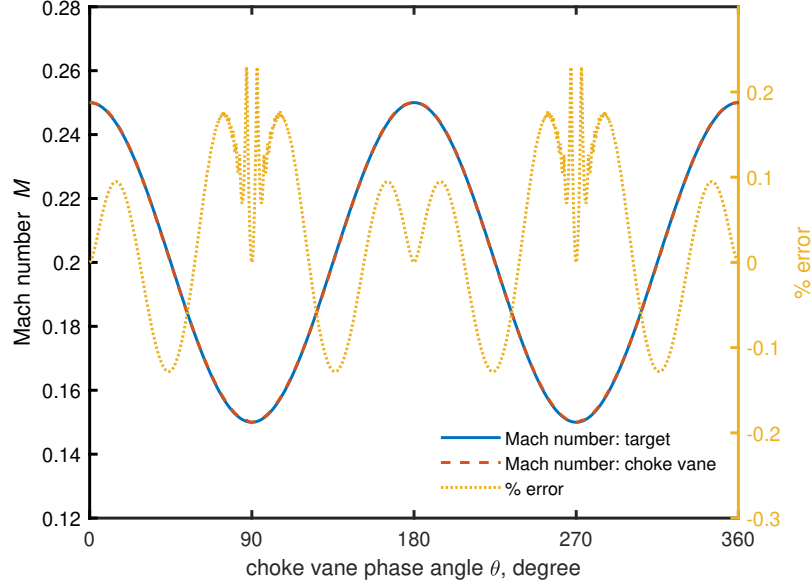


Figure 3.2: Error analysis of the sinusoidal freestream Mach number based on the choke vane design.

phase offset. In addition, the drive chain was configured such that adjacent choke vanes were counter-rotating to minimize any asymmetric unsteady aerodynamic loads on the transmission. Together, freestream velocity modulation up to 20Hz can be achieved where the forcing frequency was measured by incremental shaft encoders on each motor.

3.2 Motivations and Objectives

A few fundamental questions are essential in order to thoroughly characterize the wind tunnel for the subsequent surging airfoil investigation. First, it is important to understand the limit of possible surging flow conditions that can be achieved in this facility, such as the oscillating amplitude σ and reduced frequency k . Despite

the inviscid area-Mach relation provides a means to generate surging flow, the experimental flow condition has been found to be able to deviate from the target condition [11, 13, 14, 17]. This is due to challenges associated with viscous effects, acoustic resonance, etc. For this wind tunnel and similar facilities, because of the presence of downstream devices (choke vanes or louvers) perturbations are traveling upstream at the speed of sound and the resulting acoustic resonance is likely to alter the magnitude and shape of the time-varying velocity [11, 17]. In addition, a resulting cut-off frequency was found to limit the actual feasible reduced frequency [11, 14]. As a result, the experimental flow parameters (*i.e.*, mean velocity, advance ratio, and reduced frequency) can be limited by the tunnel dynamics. Moreover, during the actual wind tunnel operation, there is a time-varying boundary layer along the sidewalls and unsteady viscous effects around the rotating choke vanes. However, this is very challenging to be taken into consideration at the design phase. Greenblatt [11] attempted a modification to the continuous operation of the louver system in an ad-hoc manner to obtain the best sinusoidal freestream. However, this method is too costly to be repeated in the current facility. As a result, we investigated a range of freestream flow conditions in this wind tunnel and defined a practical test matrix for the subsequent surging airfoil experiment.

Additionally, in order to quantify the flow condition in the wind tunnel, it is equally important to understand the accuracy of measurements made in such experiments. The parameters of interest are stagnation and static pressure, stagnation temperature, and velocity in the test-section. Whereas it is simpler to quantify these parameters under the steady flow condition, more challenges exist when the

flow is oscillatory. In this facility, freestream pressure is measured through pitot-static probes. However, not only is there bias error associated with the instrument, but also dynamic effects of the finite-length tubing need to be compensated for the unsteady experiments. What's more, proper phase correction is needed for measurements at different streamwise locations, which is important since the test article is located further downstream. Similarly, the accuracy of temperature measurements depends on the bias error, the frequency response of the sensor, and the sampling rate available. With the time-varying pressure and temperature during the wind tunnel operation, accurate measurement of the unsteady velocity is another challenge. Isentropic relation had been applied to calculate unsteady velocity based on pressure and temperature measurements [17]. However, the assumption of quasi-steady flow becomes inaccurate as frequency increases and the accuracy of velocity measurements need to be investigated. In this chapter, we investigated these three parameters utilizing pitot-static probes, a fast-response temperature probe, and a hot-film constant-temperature-anemometer (CTA) system. Their uncertainties were also quantified. Hot-film measurements were compared with isentropic analysis of pressure measurements to understand when unsteady effects become significant.

Last but not least, it is also important to quantify the freestream flow quality, *i.e.*, the freestream turbulence intensity. The state of the freestream turbulence intensity affects the extent of laminar and turbulent flow over an airfoil, which is crucial in establishing the aerodynamic forces and moments. In this chapter, we characterized the freestream turbulence intensity for both steady and various unsteady scenarios. The proper investigation of these fundamental questions helped us design the experiments in the following chapters.

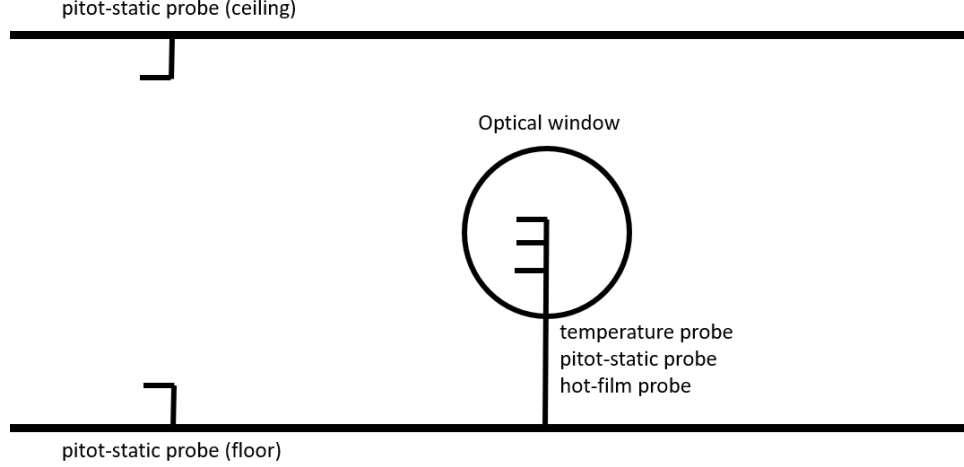


Figure 3.3: Schematic of instruments installed in the wind tunnel test-section.

3.3 Methods

3.3.1 Experimental Instrumentation

To characterize the wind tunnel flow conditions, several different instruments were applied to measure the flow pressure, temperature, and velocity. The overall setup of the instruments in the test-section is shown in Figure 3.3, where two upstream pitot-static probes near the tunnel ceiling and floor, respectively, were used to measure freestream pressures. An array consisting of pitot-static, temperature, and hot-film probe was located at the downstream optical access, where an airfoil is typically installed.

Multiple pitot-static probes were installed at different locations in the test section, as shown in Figure 3.3. Each pitot-static probe has a 1.19mm sensing orifice for stagnation pressure measurements and four 0.46mm static ports for static pressure measurements. In addition, the probes have a small outer-diameter of 3.175 mm and were located at the spanwise center of the test-section. A schematic showcasing the

pitot-static probe dimension is shown in Figure 3.4. The pitot-static probes were connected to the ESP 32HD pressure scanner via durable plastic tubing of 1.4mm inner-diameter and approximately 76cm long. The ESP scanner was mounted just outside the wind tunnel and is a miniature electronic differential pressure measurement unit consisting of an array of 32 silicon piezoresistive pressure sensors. The operating range of this pressure scanner is $\pm 206\text{kPa}$ differential and the transducers are accurate to within $\pm 0.03\%$ of the full scale pressure range with individual sensor thermal compensation to minimize zero and span shifts. The electrical outputs of the sensors were electronically multiplexed through a single onboard instrumentation amplifier at switching rates up to 70kHz using binary addressing. The multiplexed amplified analog output was connected to a DTC Initium scanner interface and streamed via Ethernet to a dedicated hard disk. In order to ensure data streaming at a stable sampling interval, the DTC Initium was triggered with an external TTL pulse train which was generated by the data acquisition computer. In the current setup, the pressure data were sampled at 1000 samples per second per channel and no additional filter was applied.

Besides the pitot-static probes, fast-response temperature probes were installed in the tunnel to measure the stagnation temperature of the flow, one installed inside the settling chamber and another located at the streamwise location of the optical access. The fast-response temperature probe was based on the design of Winkler [57], shown in Figure 3.5. This design was originally intended for supersonic and hypersonic experiments and the important design consideration is to minimize heat losses through conduction, convection, and radiation between the flow and the probe material. Thus, vent holes are used to replenish the flow around the sensing element

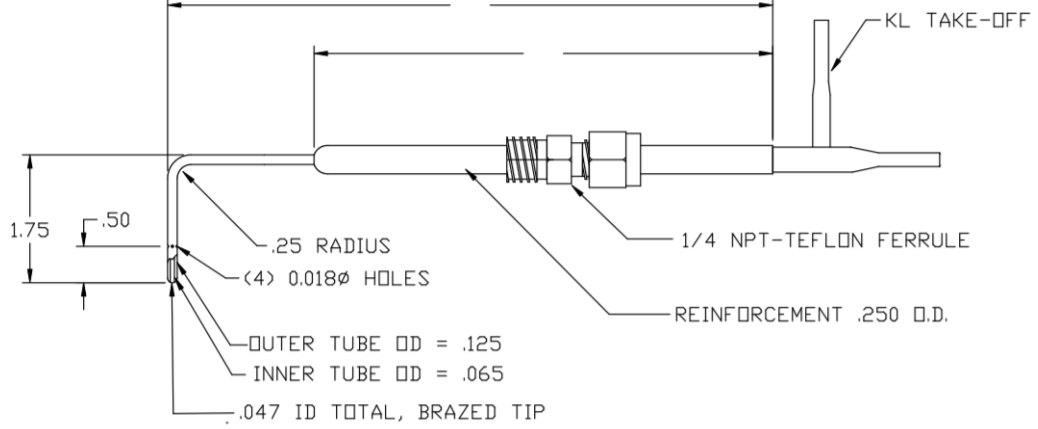


Figure 3.4: Schematic of the pitot-static probe made by United Sensor, dimensions are in inches.

(*i.e.*, the thermocouple) and the probe tip is made of silica shield. In the current experiment, the probe was made in the lab with stainless steel probe supports, heat-shrinking tubes, and Omega type-J thermocouple of 0.127mm diameter. Because the expected working environment is incompressible flow around room temperature, stainless steel supports were used without the silica shield. A calibration of the temperature probe was conducted with a TSI velocity calibrator (model 1127). Results indicated that such temperature probe has a recovery factor near unity under the incompressible flow regime.

Another important consideration of temperature measurements is the frequency response of the temperature probe. This depends on the medium and the flow speed, the material of the thermocouple, and the thermocouple junction type. Typical thermocouple probes have a frequency response on the order of 1Hz or worse. However, this is not sufficient for unsteady flows on the order of 1Hz or above. Since the thermocouple is a first order system, the frequency response can be estimated using

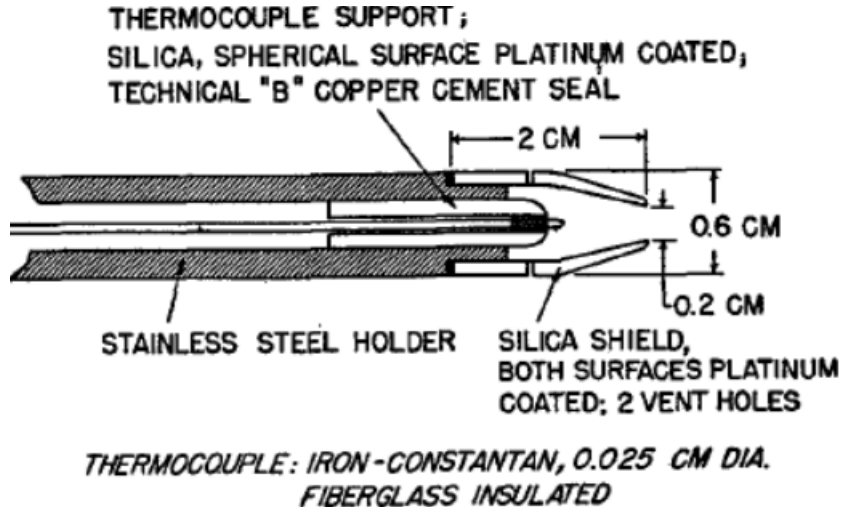


Figure 3.5: Schematic of the fast-response temperature probe designed by Winkler 1945.

Equation 3.2 assuming the junction is a sphere and each material occupies half the volume [58]

$$\tau = \frac{mC}{hA} \quad (3.2)$$

where m is the mass of the material, C is the specific heat of the material, h is the convective heat transfer coefficient of the flow, and A is the surface area. Based on this estimate, the extremely thin bare wire (*e.g.*, diameter of 0.127mm) can have a response time of 0.02s. The actual response time is likely to be slightly slower because of the probe assembly. Additionally, with a National Instrument (NI) 9212 temperature module capable of sampling 95 samples per second, the stagnation temperature can be measured for unsteady forcing frequency on the order of 10Hz. The accuracy of the stagnation temperature measurement is 0.71 Kelvin, based on the thermocouple purchased from the vendor.

From pitot-static probe and temperautre probe measurements, flow velocity in the test section can be calculated based on the isentropic relation with the quasi-steady assumption,

$$M = \sqrt{\left(\left(\frac{p_s}{p_t}\right)^{\frac{\gamma-1}{\gamma}} - 1\right) \frac{2}{\gamma-1}} \quad (3.3)$$

$$T_s = T_t \left(1 + \frac{\gamma-1}{2} M^2\right)^{-1} \quad (3.4)$$

$$u = Ma = M \sqrt{\gamma R T_s} \quad (3.5)$$

however, the quasi-steady assumption becomes invalid for highly unsteady flows. Thus, a Dantec hot-film probe was included (see Figure 3.3) and served as another way to measure the unsteady velocity in the test-section. A Dantec hot-film probe (model 55R01) with a Dantec CTA (model 54N80) was used in the current experiment, shown in Figure 3.6. The working principle of the hot-film CTA system is based on the heat transfer balance of the thin film, in which the circuitry of the system leads to electrical heating of the film and it is balanced by the cooling of the air flow over it

$$\dot{q}_h = I_w^2 R_w \quad (3.6)$$

$$\dot{q}_c = hA(T_w - T_f) \quad (3.7)$$

where $\dot{q}_h$ is the heat transfer due to electrical heating, I_w is the electrical current through the film, R_w is the film resistance, $\dot{q}_c$ is the heat transfer due to flow cooling over the film, h is the convective heat transfer coefficient, A is the area of the film immersed in the flow, T_w is the film operating temperature, and T_f is the flow temperature. Among these parameters, T_w is maintained at constant value by the CTA which includes a Wheatstone bridge that constantly adjusting the output voltage E . In the current experiment, the thin platinum film has a diameter of $70\mu\text{m}$ and a

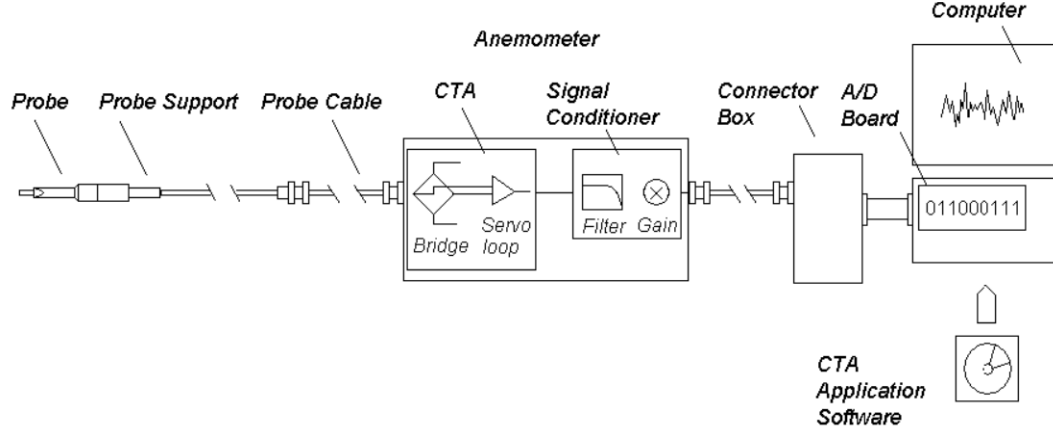


Figure 3.6: Schematic of the Dantec hot-film CTA system.

length of 1.25mm. Its resistance at 20°C is 5.79Ω. For best sensitivity, the overheat ratio was selected for a thin film operating temperature of 280°C. The hot-film signal (*i.e.*, the CTA output voltage E) was sampled at 10kHz and low-pass filtered at 5kHz to attenuate noise. A probe array of one pitot-static probe, one fast-response temperature probe, and one hot-film probe to measure flow properties at the same streamwise location as the test article (*e.g.*, an airfoil) is shown in Figure 3.3 and 3.7.

3.3.2 In-Situ Calibration of the Hot-Film Probe

The hot-film probe needs to be calibrated in order to measure flow velocity. When a hot-film probe is operated in incompressible flow, a priori calibration is typically conducted and the calibration curve of CTA voltage E and flow velocity u is acquired through polynomial relations or power laws. The calibration relation is then applied to experimental measurements to determine unknown velocity and turbulence intensity. However, when a hot-film probe is used in a compressible (or pressurized) environment the calibration curve should be (or is preferred to be) constructed using film Nusselt

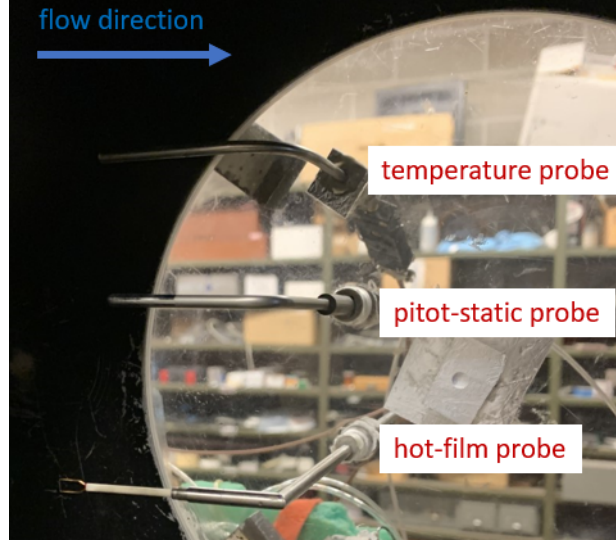


Figure 3.7: Probe array at the test-section of the unsteady wind tunnel.

number Nu and film Reynolds number Re_d [58, 59]

$$Nu = \frac{hd}{k} = \frac{E^2 \left[\frac{R_w}{(R_w + R_d)^2} \right] Pr^{1/3}}{\pi kl(T_w - T_f)} \quad (3.8)$$

$$Pr = \frac{c_p \mu}{k} \quad (3.9)$$

$$Re_d = \frac{\rho u d}{\mu} \quad (3.10)$$

where d is the film diameter, k is the thermal conductivity of the flow, R_d is the decade resistance in series with the hot-film in the Wheatstone bridge, Pr is the Prandtl number of the flow, l is the length of the film, c_p is the specific heat of the flow, ρ is the flow density, u is the flow velocity, and μ is the dynamic viscosity of the flow. The flow properties ρ , μ , and k are evaluated at the average of T_w and T_f . Sutherland's law can be applied to calculate μ and a similar relation can be applied to calculate k [60]. At last, a power law is found to connect Nu and Re_d . The first and probably the most famous relation is given by King [61], referred to as the King's

law. More similar relations have been developed over the years. In this work, the relation described by Collis & Williams was used [59]

$$Nu = aRe_d^n + b \quad (3.11)$$

where a , b , and n are calibration coefficients. The hot-film signal, however, is sensitive to temperature change. Thus, calibration curves (*i.e.*, Nu vs. Re_d) are also functions of the temperature. A temperature correction term is needed in order to obtain a universal calibration relation for temperatures of interest. The temperature correction term is following that in reference [59]:

$$Nu_{cor} = Nu \left(\frac{T_{avg}}{T_f} \right)^p \quad (3.12)$$

where T_{avg} is an average of T_w and T_f , p is an empirical coefficient that would make calibration curves of Nu at various temperatures collapse as one. As a result, the final calibration relation between a temperature-corrected Nu and Re_d is

$$Nu_{cor} = aRe_d^n + b \quad (3.13)$$

The calibration process requires sufficient data that cover the range of the target flow condition, *i.e.*, the film Reynolds number Re_d under the surging flow experiment. In addition, it requires at least two different operating flow temperatures to construct the calibration relation, Equation 3.13. However, the facility of the current experiment is pressurized and the flow condition varies with the operating duration, especially the stagnation temperature due to the pressurized blow-down nature of the wind tunnel, illustrated in Figure 3.8. Thus, a priori calibration of the hot-film probe (typically with a velocity calibrator) to match the possible flow condition, such as the test-section flow density, velocity, and temperature, is very difficult. As a result, an

in-situ calibration experiment was designed. In this calibration experiment, the wind tunnel was operated in the unsteady mode for a long duration (approximately 40s). The unsteady mode allowed the calibration to cover the Reynolds number variation of the surging flow condition, whereas the long operating duration helped to create different stagnation temperatures in the test-section. In addition, the frequency of the rotating choke vanes was low (approximately 1.5Hz) and kept constant at the beginning of the tunnel operation. After the tunnel was fully pressurized, the frequency was increased gradually to 11Hz initially and then gradually reduced to 1.5Hz near the end of the tunnel operation. The measurements at the beginning and the closing stage of the experiment were taken to construct the calibration relation, whereas the high frequency period was intended to highlight the unsteadiness of the measured properties and to compare the response of the sensors (*e.g.*, pitot-static probe vs. hot-film probe).

The time-resolved information of the flow condition for a full unsteady tunnel operation is illustrated in Figure 3.9. The frequency was varied from 1.5Hz at the beginning (approximately 0-10s) to a maximum of 11Hz and back to 1.5Hz near the end as described above (*i.e.*, a frequency-sweep experiment). An interesting finding is that the magnitude of the pressure oscillation captured by the pitot-static probe was reduced as the forcing frequency increases. This indicates a low-pass filter characteristic of the wind tunnel. The details of the effects of the forcing frequency and the causes will be discussed later. For hot-film in-situ calibration, parameters measured under low-frequency at the starting and the ending stage were utilized (*i.e.*, from 5 to 10 seconds and from 35 to 40 seconds in Figure 3.9).

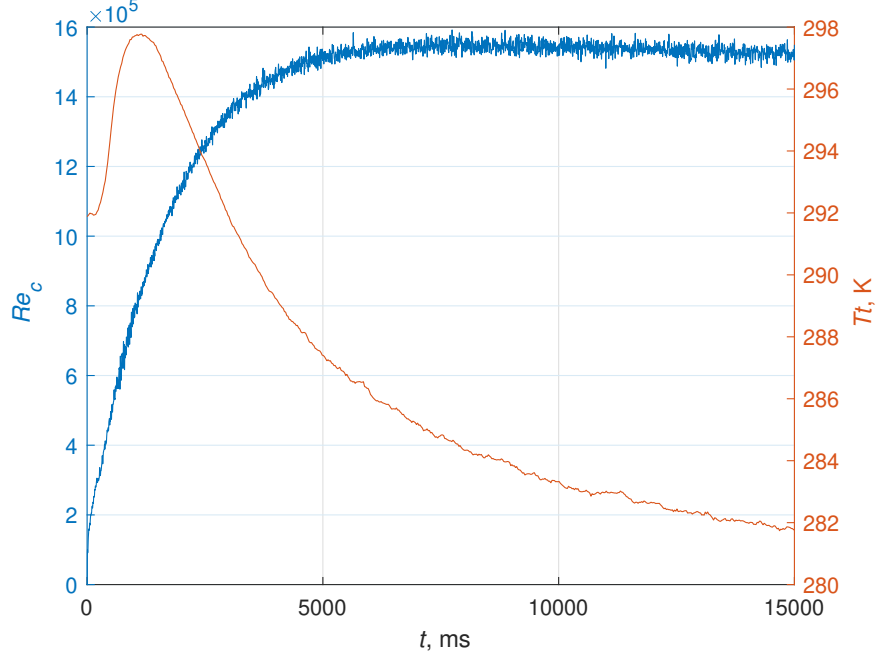


Figure 3.8: Time history of flow parameters during a typical 15-second steady blow-down operation.

Figure 3.10 shows the Nu and Re_d relation from the measurements based on the low-frequency data. The Nu was derived from the hot-film CTA voltage and temperature measurements (Equation 3.8), whereas the Re_d was derived from the pitot-static probe and temperature measurements (Equation 3.10). The calibration coefficients p , a , b , and n were found through an iterative process until the best curve fit was found to collapse the calibration data. The resulting calibration coefficients were found to be $p = -0.4402$, $a = 0.0048$, $b = 0.0093$, and $n = 0.4882$.

Figure 3.9 also highlights the transient behavior of the flow parameters during an unsteady wind tunnel operation. That is, flow parameters start from ambient condition to the pressurized condition and drop gradually during the tunnel operation. The decreasing magnitude of flow properties is because of the decreasing tank pressure

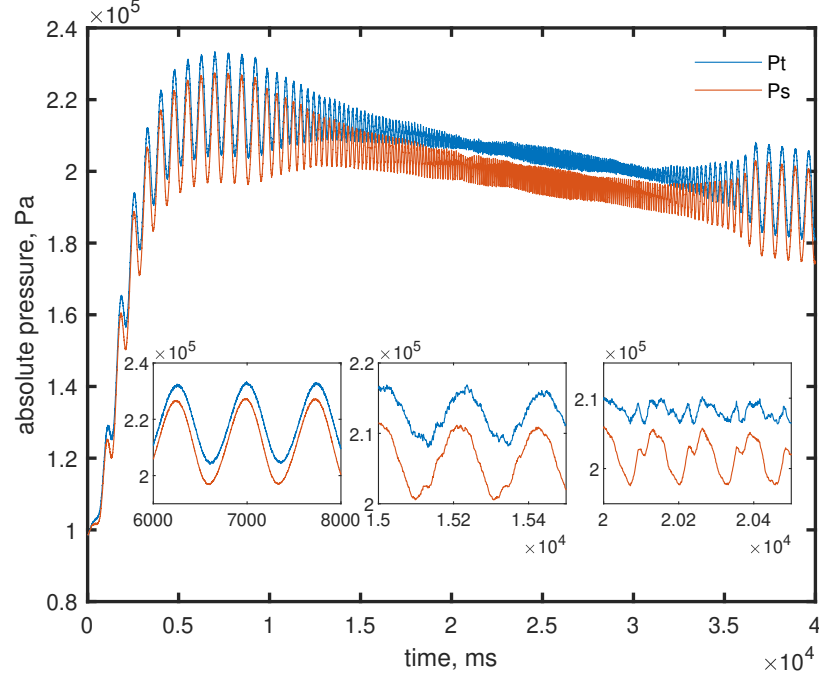


Figure 3.9: Time history of pressure measured during a frequency-sweep wind tunnel experiment. Measured pressure at different intervals (7-8s, 15-15.5s, 20-20.5s) and frequencies (1.5Hz, 5Hz, 8Hz) are highlighted.

(and energy) and the absence of a feedback mechanism in the upstream solenoid valve to maintain a constant mass flow rate throughout the operation. As a result, for both steady and unsteady test conditions, the mass flow rate gradually drops as the tank pressure is depleted. The actual rate of change of parameters was found to depend on the target flow conditions (in terms of test-section Mach number and Reynolds number). For example, for a 30s steady operation of $M_\infty = 0.2$ and $Re_c = 1,500,000$, pressure measurements have shown that the stagnation pressure in the test-section can drop by 15,000Pa, stagnation temperature can decrease by 10 Kelvin, and the Reynolds number can change by approximately 45,000 (a 3% variation).

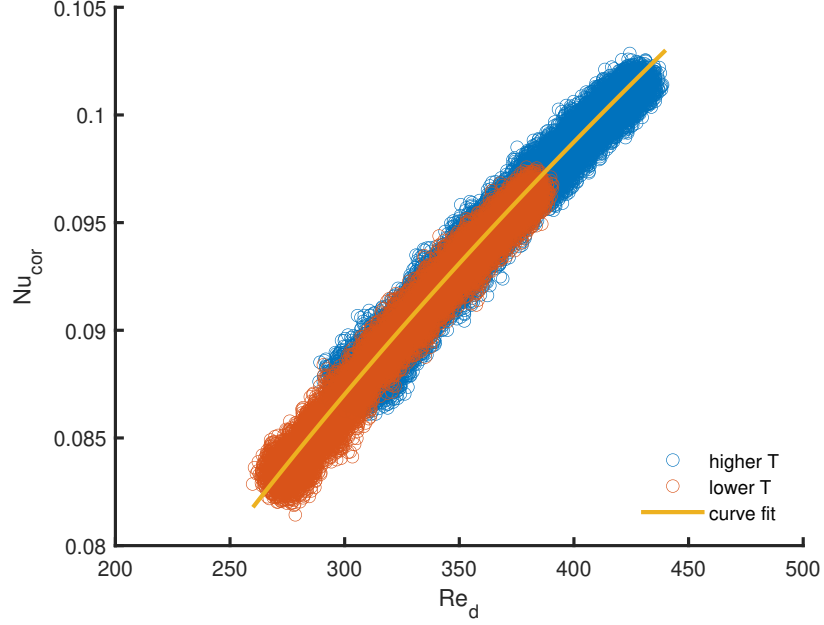


Figure 3.10: Calibration curve of the hot-film probe.

3.3.3 Numerical Model

The choke vane is designed to generate the target Mach number in the test section. However, the inviscid one-dimensional analysis only considers two points of interest, that is the throat and the test-section, and neglects any flow physics in between and along the entire wind tunnel. To help understand the dynamics of the unsteady wind tunnel, particularly from the settling chamber to the throat, a numerical code was developed as a reduced-order model. This method approximates the flow condition in the wind tunnel as quasi-one-dimensional, inviscid, and unsteady. The governing equations for such flow are described by Euler's equations [62]. As a result, the unsteady transonic wind tunnel dynamics were modeled by solving the compressible Euler's equations in terms of primitive variables (*i.e.*, velocity u , density ρ , and temperature T). The governing equations are given by Equation 3.14 to 3.16, which also

include the effect of area change along the wind tunnel.

$$\frac{\partial \rho}{\partial t} + u \frac{\partial \rho}{\partial x} + \rho \frac{\partial u}{\partial x} + \rho u \frac{\partial A}{\partial x} = 0 \quad (3.14)$$

$$\rho \frac{\partial u}{\partial t} + \rho u \frac{\partial u}{\partial x} = - \frac{\partial P}{\partial x} \quad (3.15)$$

$$\rho \frac{\partial e}{\partial t} + \rho u \frac{\partial e}{\partial x} = -P \frac{\partial u}{\partial x} - \frac{P u}{A} \frac{\partial A}{\partial x} \quad (3.16)$$

Equation 3.14 to 3.16 includes the continuity, the x -direction (streamwise) momentum, and the energy equation. In addition, the calorically perfect gas and the ideal gas law are included to solve the four unknowns in these equations. The governing equations can be solved by a finite difference method. MacCormack's predictor-corrector technique [62] was applied to compute the dynamics of the unsteady wind tunnel. This technique is an explicit approach and second-order accurate in both space and time. This numerical method was used to simulate tunnel dynamics starting from the settling chamber (downstream of the last screen) to the constant-area duct just upstream of the throat for a variety of freestream flow conditions and forcing frequencies. The total length L of the tunnel being modeled was 4m. The initial condition was chosen to be the mean (velocity) of the unsteady experiment. The inlet boundary condition represents the stagnation condition in the settling chamber upstream of the nozzle. In this case, density and temperature were specified to be stagnation condition and velocity was allowed to float. In contrast, the outlet boundary condition represents the unsteady flow caused by the variation of the throat area due to the rotation of the choke vanes. Thus, velocity was defined by the sinusoidal throat area from the choke vane design, whereas density and temperature were allowed to float. This finite difference method was written in MATLAB. Courant number was modified for different flow conditions to allow sufficient temporal resolution and convergence. The

stagnation temperature was chosen to be the standard day temperature of 15°C since the stagnation temperature during tunnel experiments usually falls between 10°C and 20°C. A grid independence study was performed among 1000, 2000, and 4000 spatial grid points to compare their convergence quality and numerical cost. Although all of them converged, a fine case of 4000 spatial grid points was finally chosen to improve the temporal resolution of the numerical results.

3.4 Results and Discussion

3.4.1 Effects of the Forcing Frequency

Figure 3.9 has already shown that the flow variables during the unsteady experiment are a strong function of the forcing frequency. Both the oscillating amplitude and the shape of the static and stagnation pressure were found to vary with the frequency. This is further illustrated in Figure 3.11 where the static and stagnation pressure measurements were phase-averaged after correcting the declining trend after the tunnel pressurization ($t \geq 5$ s). In Figure 3.11, it is observed that the pressure time-history is sinusoidal, especially at low forcing frequencies. However, as the frequency increases the pressure time-history is reduced in magnitude and higher frequency components start to show up. For example, when the forcing frequency equals to 10.5Hz the oscillating amplitude has reduced by approximately 75% compared to when the forcing frequency equals to 1.5Hz. In addition, there are two peaks within the pressure cycle at this frequency, one at $\phi = 200^\circ$ and the other at $\phi = 315^\circ$. Similarly, the freestream Mach number, if evaluated based on the isentropic relation (Equation 3.3), also exhibits a sinusoidal-like waveform at low-frequencies but is greatly distorted as the frequency grows.

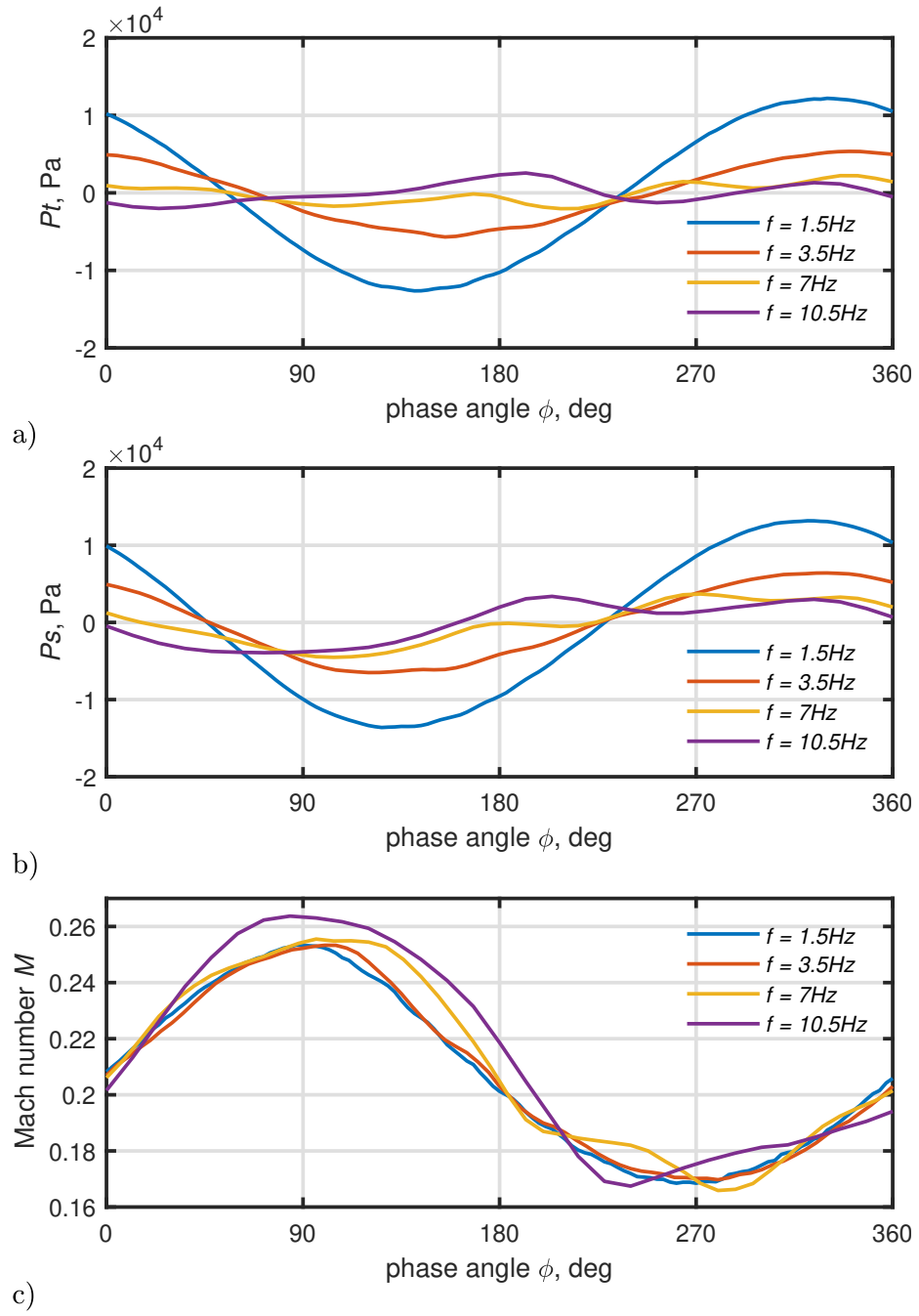


Figure 3.11: Phase-averaged parameters vs. the forcing frequency, subplots are aligned based on the accelerating mean of the Mach number, a) stagnation pressure, b) static pressure, c) Mach number from isentropic relation.

A few reasons are hypothesized to cause such phenomenon. First, the pressure measuring system may lack the dynamic response to resolve the unsteady pressure. Thus pressure measurements may have suffered from magnitude attenuation and phase shift. Second, because the downstream rotating choke vanes are causing perturbations that travel upstream at the speed of sound to update the freestream condition, it is possible that these propagating waves can interact with each other. The wave propagation, if coalesced with acoustic resonance, can lead to large pressure distortions inside the wind tunnel. Each of these hypotheses is evaluated below.

For the dynamic response of the pressure measuring system, Bergh and Tijdeman [63] developed an analytical model that compensates the attenuation and phase lag associated with pneumatic tubing. Their model characterizes the dynamic response of the pneumatic system as a function of the inner diameter and length of the individual pneumatic line and the sensor volume, along with the ambient conditions. The model (referred to as the B-T model) is a transfer function that ratios the pressure measured by the transducer with the actual unsteady pressure in the frequency domain.

The accuracy of the B-T model was demonstrated with an acoustic excitation test in which sinusoidal pressure oscillations were generated across a range of frequencies. The details of this experiment are given in Appendix C, including the dimension of the pneumatic line. The amplitude ratio and phase shift are plotted with excitation frequency in Figure 3.12. It can be observed that the experimental data and the model predictions are in reasonable agreement up to 100Hz. Overall, the B-T model is proved to be a good predictive tool to compensate the dynamic response of the pneumatic tubing. For the flow conditions (mean $Re_c = 1,500,000$, mean $M_\infty = 0.2$) and range of frequencies (1.5-11Hz) in the current wind tunnel experiment, results

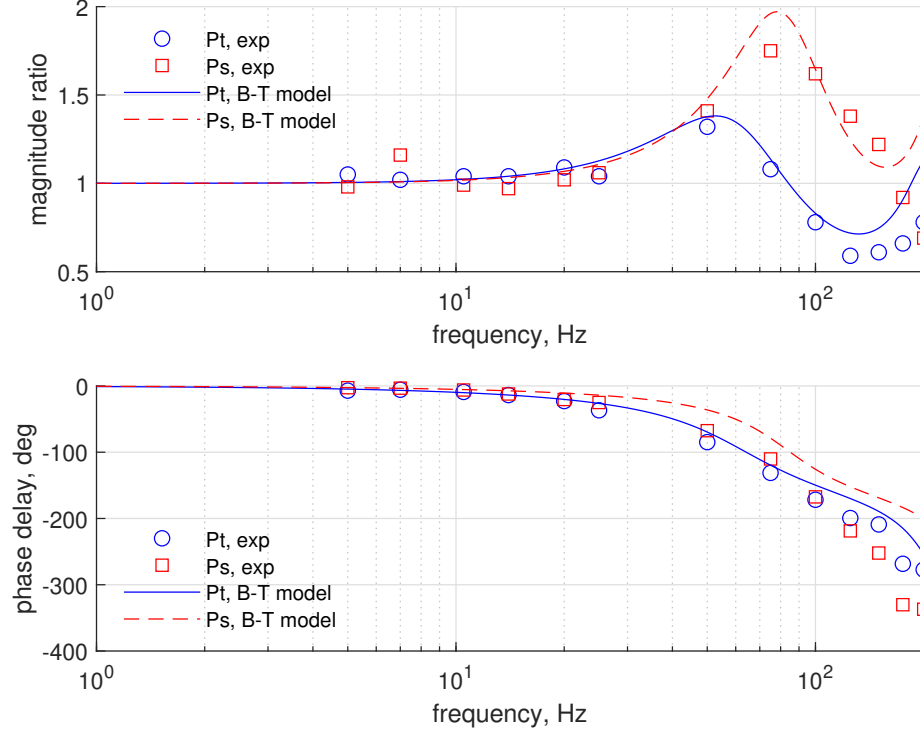


Figure 3.12: Transfer function of the pressure measuring system from the acoustic excitation experiment and compared with predictions from the B-T model.

show negligible effects to the measurements in terms of magnitude (below 5%) and phase delay (below 10 degrees). Thus, it is concluded that although the pneumatic line contributes to errors in pressure measurements, it can be corrected for and is not the dominant effect leading to the large pressure variations shown in Figure 3.11.

For the tunnel dynamics, the wave speed can be approximated as

$$c = a - u \quad (3.17)$$

where c is the wave speed of information traveling upstream relative to the stationary ground, a is the instantaneous speed of sound, u is the instantaneous freestream velocity. The speed of sound depends on the local temperature and is relatively

constant since the temperature variation is not significant (less than 2% change in sound speed for a 10-Kelvin variation), whereas the freestream velocity is oscillating with time and approximately by 25% in the current experiment. Thus, during the deceleration part of the velocity cycle, the wave speed is increasing and can catch up to the slower wave generated earlier. As a result, multiple waves can coalesce into one and distort the local pressure, and thus possible distortion to Mach number and velocity. This seems to agree with the isentropic Mach number evaluated in Figure 3.11 where more distortions are found during the decelerating phase (between $\phi = 90^\circ$ and $\phi = 270^\circ$).

Furthermore, the power spectrum of the pressure measurements is plotted in Figure 3.13. Possible sources of pressure oscillations include the fundamental frequency of rotating choke vanes, the acoustic resonance, and the choke-vane shedding. For the power spectrum shown, the fundamental frequency of the oscillatory choke vanes always possesses the greatest energy content. In addition, as the frequency increases to 7Hz, the first few higher harmonics of the forcing frequency start to possess similar order of magnitude as the fundamental frequency in the power spectrum. This coincides with secondary peaks of the pressure signal in Figure 3.13 (*i.e.*, higher-frequency oscillation of two to three times the forcing frequency). For acoustic resonance, the tunnel can be modeled similarly to a Helmholtz resonator where air in the test-section acts as an oscillating mass. Greenblatt [11] provided a detailed derivation of such a system where the resonance is superimposed on the mean flow. The Helmholtz resonance frequency can be calculated by

$$f_H = \frac{c}{2\pi} \sqrt{\frac{A_{ts}}{L_{ts} V_{sc}}} \quad (3.18)$$

where c is the speed of sound, A_{ts} is the cross-sectional area of the test section, L_{ts} is the length of the test-section, and V_{sc} is the volume of the stagnation chamber. Equation 3.18 predicts a resonance frequency at approximately 7Hz for this facility. However, this is not evident in the power spectrum of pressure measurements. Another mode of acoustic resonance is the organ-pipe quarter-wave resonant mode [64]. This can occur to a blow-down tunnel with long test-section and large contraction volume. The quarter-wave resonant mode frequency is estimated as $f_{1/4} = c/(4L_{ts})$ which predicts approximately 30Hz for the current tunnel configuration. An excitation at this frequency can be observed in Figure 3.13. Finally, the choke-vane shedding frequency is estimated based on the Strouhal number

$$f_{shedding} = \frac{uSt}{L} \quad (3.19)$$

where the shedding frequency depends on the instantaneous freestream velocity u , the Strouhal number which is assumed to be 0.2, and the characteristic length L . This gives a first approximation assuming the choke vane is quasi-steady. The combination of velocity magnitudes and the major and minor axes of the choke vane predicts a shedding frequency between 120 and 160 Hz. However, this is not discernible from the broadband noise in the spectra. Instead, a broad range of excitation is distinguishable around 80Hz which might be the actual shedding frequency of the system of four counter-rotating choke vanes.

Based on these results, it is presumed that unsteady pressure in the wind tunnel suffers from distortions caused by acoustic waves traveling upstream which is dominated by the forcing frequency. Acoustic resonance and vortex-shedding of the choke-vane also contributes to the flow fluctuations in the test-section. Overall, the reduced pressure magnitudes at higher forcing frequencies highlight the low-pass filter

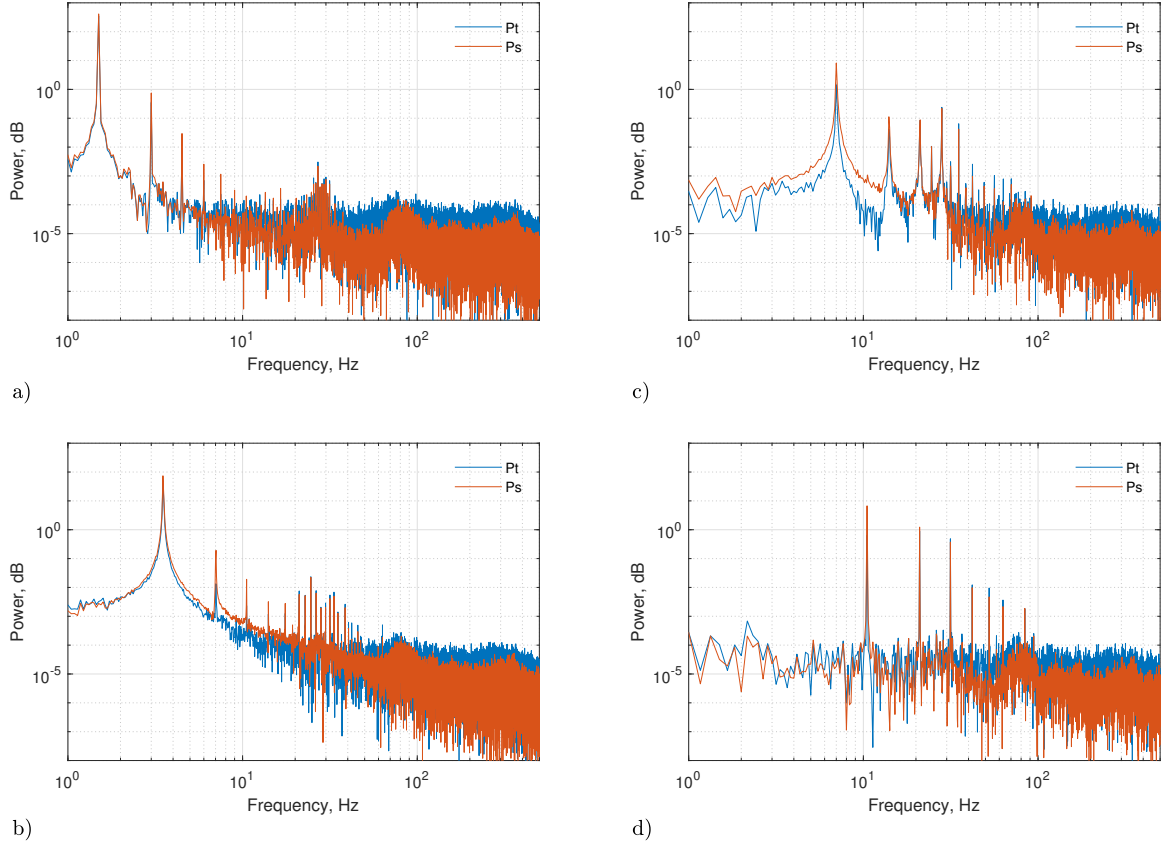


Figure 3.13: Power spectrum of the pressure measurements from the unsteady experiment, a) forcing frequency = 1.5Hz , b) forcing frequency = 3.5Hz, c) forcing frequency = 7Hz, d) forcing frequency = 10.5Hz.

characteristics of the wind tunnel. The limited dynamic response of the pressure measuring system is evaluated to have a negligible effect for the current range of forcing frequencies. As a result, the effect of forcing frequency to the test-section pressure is believed to be dominated by wave propagation.

3.4.2 Evaluation of the Time-Varying Freestream Velocity

For the pressurized blow-down type wind tunnel, the freestream velocity can be determined from the pressure and temperature measurements and the isentropic relation assuming the condition is quasi-steady. However, in the current investigation, the frequency has reached to the order of 10Hz and the validity of the isentropic relation becomes questionable. This is the reason that hot-film CTA system was introduced as an alternate method for determining the freestream velocity.

The calibration relation had been established for the hot-film probe from the in-situ calibration process and was applied to derive the freestream velocity (Equation 3.8 to 3.13). Because the calibration relation was based on Nu and Re_d , velocity was derived depending on all three measurements, *i.e.*, measurements from the hot-film probe, the pitot-static probe, and the temperature probe. To be more specific, the hot-film and temperature measurements were acquired to derive Nu , then calibration relation was applied to find Re_d , and finally velocity u was derived through definition of Re_d which requires pressure and temperature measurements. As a result, there are two ways to derive Re_d and u from current instrumentation, one from the hot-film, pressure, and temperature measurements, and the other from pressure and temperature measurements with the isentropic relation. The two methods are compared below.

Figure 3.14 shows the comparison of the Reynolds number based on the two methods as a function of the sweeping frequency and a bode plot of the two signals is also illustrated. The result shows that at low frequencies, the two calculated Re_d signals have excellent agreement. However, as the frequency increases the Re_d based on the pitot-static probe measurements and the isentropic relation shows a greater maximum

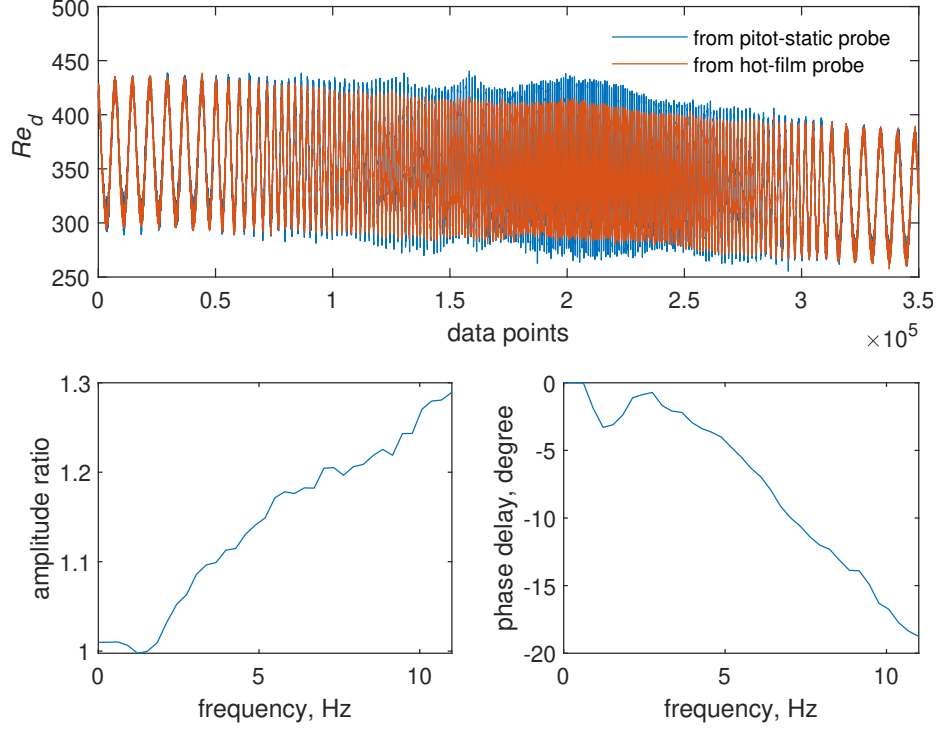


Figure 3.14: Time history of Re_d and comparison of hot-film and pitot-static probe measurements.

and a lower minimum values than that based on the hot-film probe measurements. This disagreement grows as the frequency increases. The amplitude ratio (defined as the ratio of the difference between the maximum and the minimum value of the two signals, respectively) shows that a deviation up to 30% can be reached at 11Hz. In addition, there is a growing phase delay between the two signals with the forcing frequency. The cause of this deviation is believed to be the violation of the quasi-steady assumption in applying the isentropic relation to calculate Re_d . The bode plot in Figure 3.14 indicates that the effect of the forcing frequency on the isentropic relation is only negligible when the frequency is below 3Hz.

The power spectrum of the Reynolds number evaluated from the isentropic relation and the hot-film measurements, respectively, is also plotted in Figure 3.15 to examine the unsteady effects. Similar to the pressure signals, there are multiple peaks associated with the forcing frequency and its higher harmonics. The quarter-wave resonant mode at 30Hz is still captured in Re_d by both methods but there are no evident peaks corresponding to the vortex-shedding frequency of the choke vanes (probably buried in the broadband noise). In addition, the comparison shows that isentropic Re_d has a higher broadband noise (from 50 to 500Hz) than the hot-film measurements. This indicates that the pitot-static probes with pneumatic tubing experience a higher broadband noise than the hot-film probe. Overall, the two methods appear to capture similar frequency contents of the Reynolds number but differ in the energy and noise level.

After comparing the film Reynolds number, the freestream velocity is also compared between the two methods, one based on the hot-film measurements and the other based on the isentropic relation. Re_d based on the hot-film measurements has been derived in the above discussion. To obtain velocity, pressure and temperature measurements are required to calculate density, then velocity can be derived from Equation 3.10. Two important factors are emphasized here. First, to be consistent with derivations involved in the hot-film calibration, flow properties such as ρ and μ are evaluated at T_{avg} as long as it is associated with the hot-film measurement. Second, the static and stagnation pressure measured in the tunnel are affected by the tunnel dynamics, however, they represent physical properties of the flow inside the tunnel and are applied directly. This describes the details of deriving the freestream

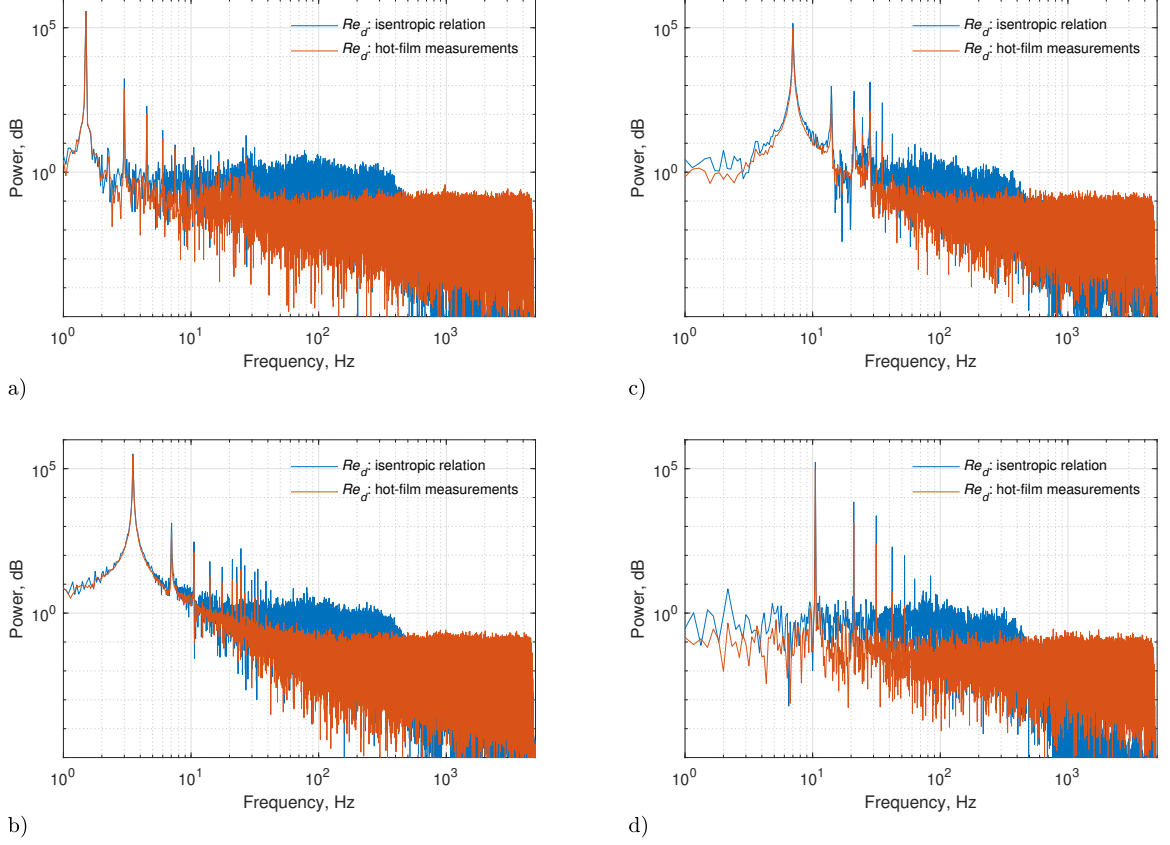


Figure 3.15: Comparison of the power spectrum of the film Reynolds number Re_d , a) forcing frequency = 1.5Hz , b) forcing frequency = 3.5Hz, c) forcing frequency = 7Hz, d) forcing frequency = 10.5Hz.

velocity from the hot-film measurements. The freestream velocity based on the isentropic relation, on the other hand, utilizes Equation 3.3 to 3.5.

The power spectrum of the freestream velocity evaluated by the hot-film measurements is shown in Figure 3.16. Similar to the pressure signals and the film Reynolds number, the quarter-wave resonant frequency of 30Hz is still visible in the velocity signal in addition to peaks at the forcing frequency and its higher harmonics. Although

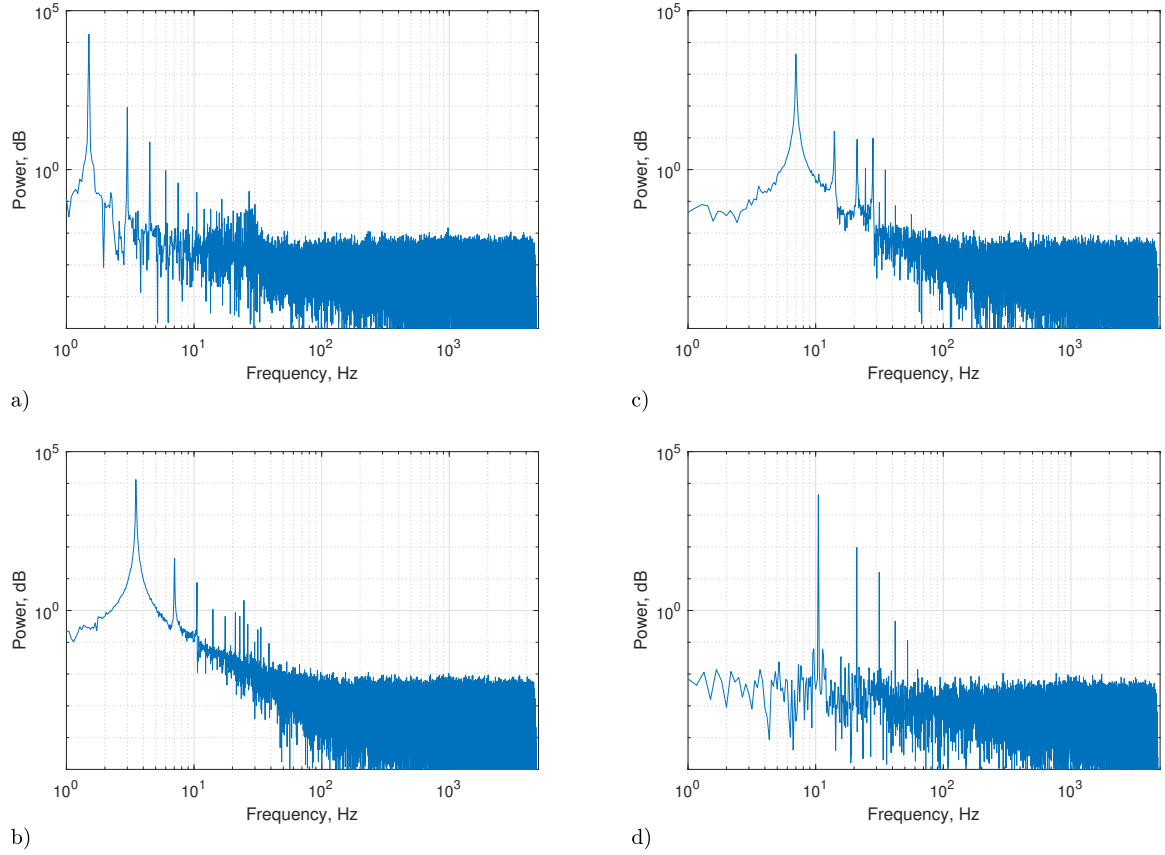


Figure 3.16: Power spectrum of the freestream velocity measured by the hot-film probe, a) forcing frequency = 1.5Hz , b) forcing frequency = 3.5Hz, c) forcing frequency = 7Hz, d) forcing frequency = 10.5Hz.

the choke-vane shedding frequency is not evident in the spectrum, it is believed it still plays a role in the dynamics of the wind tunnel.

Freestream velocity derived from both methods is also compared in Figure 3.17. Similar to Re_d , both methods output excellent agreement at low-frequency conditions since the quasi-steady assumption is still satisfied for the isentropic relation. However, as frequency increases deviations between the two methods are clearly observed. The transfer function from the two signals is also included in Figure 3.17

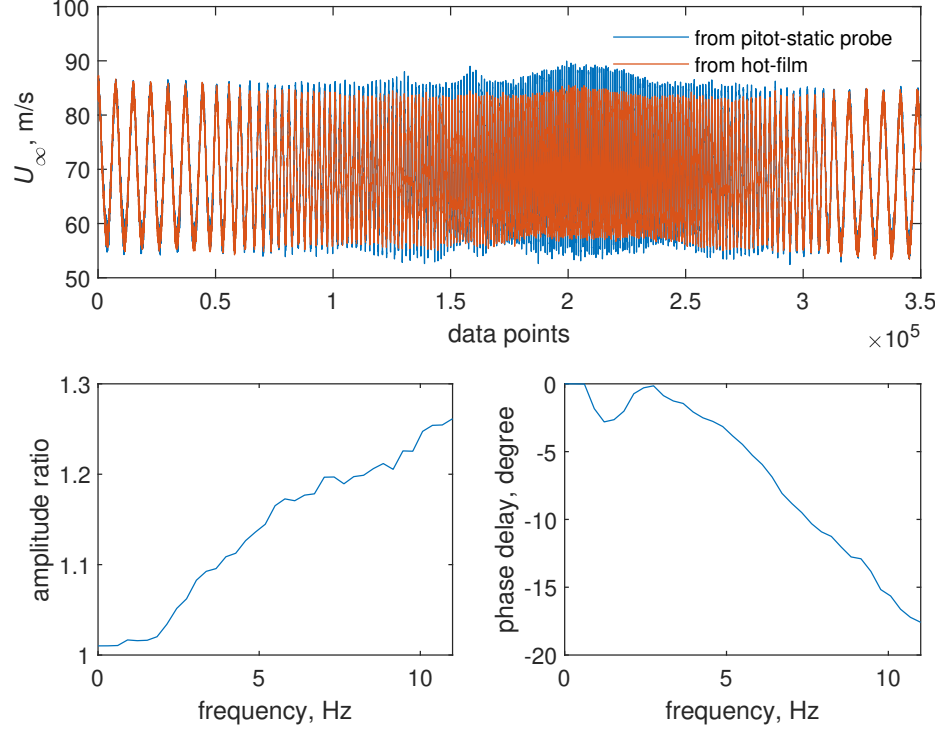


Figure 3.17: Time history of the freestream velocity and comparison of hot-film and pitot-static probe measurements.

and looks similar to that shown in Figure 3.14. Because the Reynolds number and the freestream velocity based on the hot-film measurements do not rely on additional quasi-steady assumption that has been violated, it is reasonable to treat them as the more accurate representation of the true flow condition. As such, it indicates that the pressure measurements alone with the isentropic relation can overestimate the freestream velocity at high-frequency unsteady experiments. A further comparison is given in Figure 3.18, where the film Reynolds number, the freestream velocity, and the freestream Mach number from the two techniques are phase-averaged for four fixed forcing-frequency experiments. Again, the results show the effect of the forcing frequency in distorting the flow parameters from the ideal sinusoidal wave.

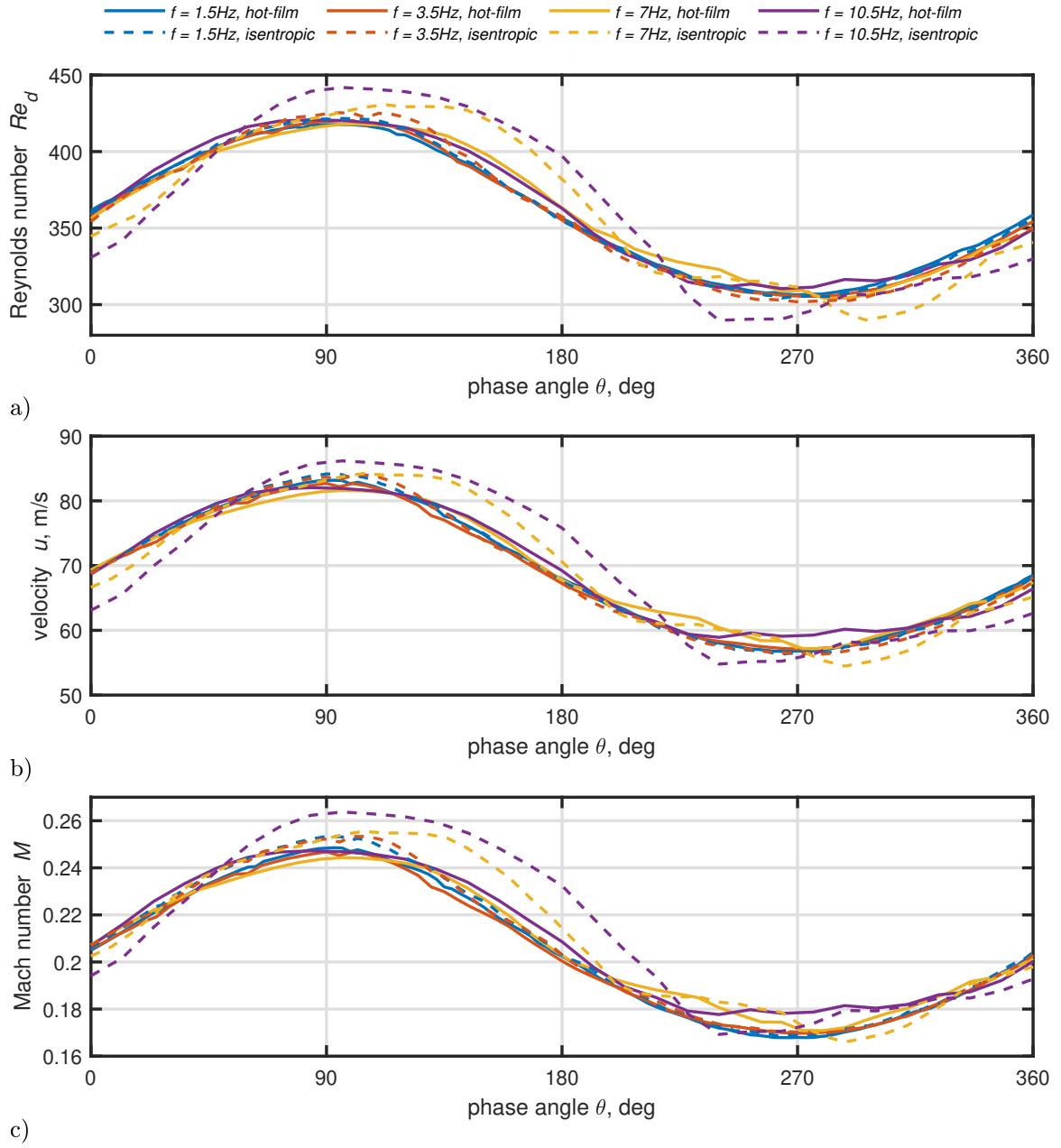


Figure 3.18: Comparison of freestream parameters based on hot-film measurements and the isentropic relation, a) film Reynolds number, b) velocity, c) Mach number.

The importance of the accuracy of the freestream velocity measurements in the wind tunnel is emphasized with Equation 3.20.

$$C_p = \frac{P - P_\infty}{\frac{1}{2}\rho_\infty u_\infty^2} \quad (3.20)$$

The pressure coefficients C_p are utilized to calculate the aerodynamic coefficients C_l , C_d , and C_m of an airfoil. In Equation 3.20, C_p depends on the local pressure and the freestream pressure, density, and velocity. Assuming all parameters other than the freestream velocity are measured accurately, a 30% error in the freestream velocity can lead to a 69% error in the pressure coefficients and additional errors when integrated for lift and moment coefficients. Thus, accurate measurements of the freestream velocity are crucial for the subsequent airfoil studies.

With the characterization of the freestream velocity measurements, the unsteady response of the wind tunnel is further examined to understand how the freestream velocity varies with the forcing frequency. An analytic model utilizing the conservation of momentum inside the wind tunnel was developed, following the same principle in reference [11]. Details of the derivation are listed in Appendix D. The key assumptions of this analytic model are that the mass flow rate along the tunnel is updated instantaneously and the parameters can be decomposed into the steady and unsteady components. As a result, it predicts the tunnel dynamics as a first-order linear system and the amplitude ratio of the velocity is shown in Figure 3.19.

Figure 3.19 shows that the amplitude ratio of velocity from the analytic model decreases with the forcing frequency, as the low-pass filter characteristic of the wind tunnel discussed above. Results from individual fixed-frequency experiments are also plotted and share the general trend with the analytic model. However, it appears that the analytic model underestimate the amplitude ratio and there is a bump around

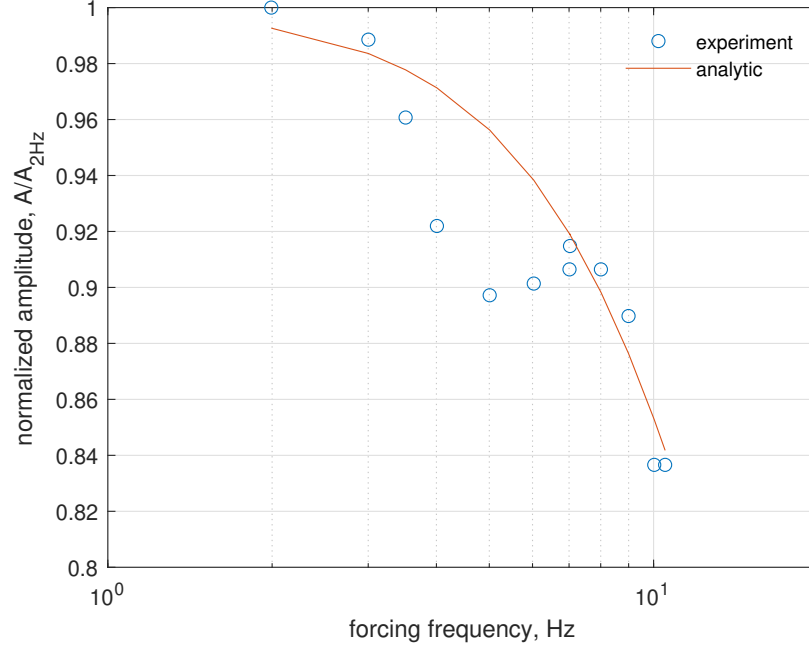


Figure 3.19: Ratio of velocity amplitude as a function of the forcing frequency.

$f = 7\text{Hz}$ from the experiment showing the amplitude is enhanced. The frequency agrees with the acoustic resonance frequency predicted by the Helmholtz resonator. Thus, it appears that this resonance mode affected the velocity amplitude.

Furthermore, there are three pitot-static probes in the current experiment, two at different streamwise locations in the test-section, see Figure 3.3. They all have the same dimension in terms of the probe body and pneumatic tubing. As a result, comparison of the measurement from the two pitot-static probes at different streamwise locations can demonstrate the effect of wave propagation. Figure 3.20 shows the phase difference between the two measurements as a function of the forcing frequency. The positive phase difference represents the signal from the upstream probe always lagging behind the signal from the downstream probe. The analytic result is based on

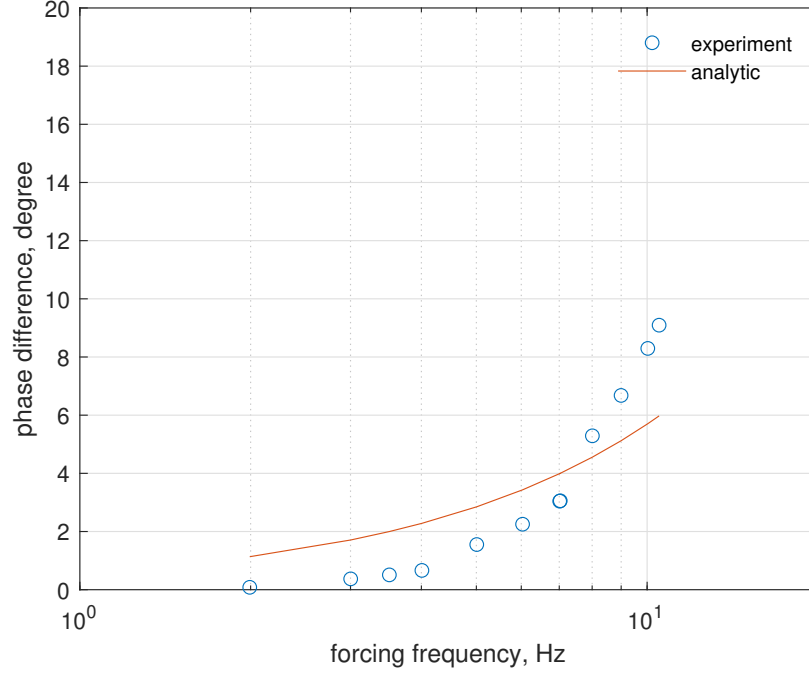


Figure 3.20: Phase difference of pressure at different streamwise locations as a function of the forcing frequency.

$\varphi = \omega L/c$, where $\omega = 2\pi f$ is the oscillating frequency, L is the streamwise distance between the two probes (0.43m), and c is the wave propagation speed. Both the experimental and the analytic results show that information is propagated towards the upstream and the phase difference increases with the forcing frequency.

Lastly, velocity based on the choke-vane design, the unsteady Euler code, and the experimental measurements are compared in Figure 3.21. The choke-vane design corresponds to the area-Mach relation utilized to inversely design the rotating choke vane. However, this analysis does not take into consideration the tunnel dynamics and is essentially a two-point inviscid analysis and independent of the forcing frequency. The Euler code is based on the numerical model assuming the flow inside the wind tunnel is quasi-one-dimensional, inviscid, and unsteady. Thus, this reduced-order

model captures some of the tunnel dynamics between the settling chamber and the throat. As a result, it predicts the test-section velocity as a function of the forcing frequency and it shows that the velocity magnitude increases with the forcing frequency. Thus, the comparison of the choke-vane design and the Euler code highlights the effects of information propagating along the wind tunnel, whereas the comparison of the Euler code and experimental measurements highlights the unsteady viscous effects (an oscillatory effective area ratio) and additional tunnel dynamics (wave propagation, shedding of the choke vane, etc.). In Figure 3.21, the measured freestream velocity always has a reduced magnitude and distorted profile, particularly around the decelerating phases. This is likely due to the complex unsteady viscous effects around the rotating choke vane. In addition, it is speculated that the flow around the choke-vane can not adjust to the shape of the vane at higher frequencies as much as with lower frequencies. This results in a more bluff profile seen by the flow and subsequently, a much thicker wake. This effect is likely more prevalent when the vanes are perpendicular to the freestream (corresponds to the velocity minimum) as opposed to when they are parallel to the freestream (corresponds to the velocity maximum), see Figure 3.22. Overall, the presence of wave distortion and unsteady viscous effects highlight the difficulties in producing a wide range of desired surging flow conditions in the pressurized blow-down type wind tunnel.

3.4.3 Turbulence Intensity

After characterizing the accuracy of the freestream velocity measurements and understanding the effect of the forcing frequency, the freestream turbulence intensity is characterized. The importance of the freestream turbulence intensity lies in that it

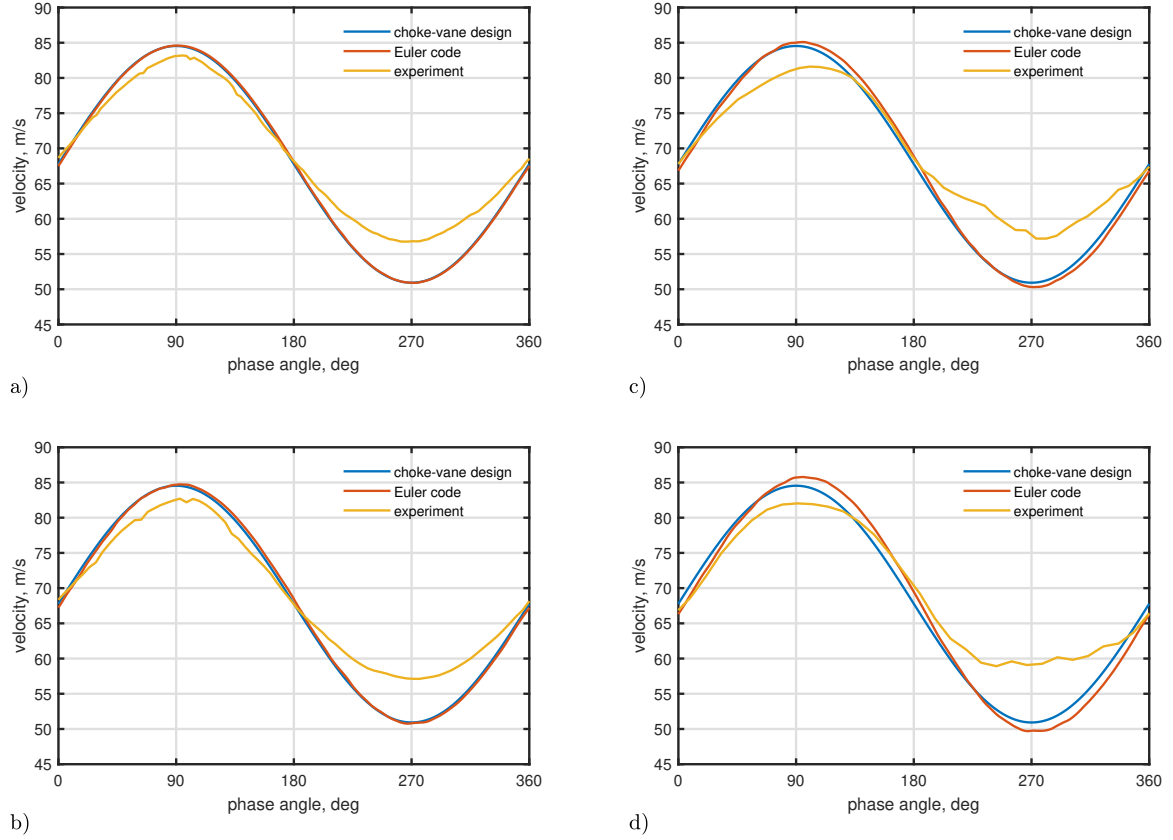


Figure 3.21: Comparison of freestream velocity based on the choke-vane inverse design, the unsteady Euler code, and the experimental measurements, a) forcing frequency = 1.5Hz, b) forcing frequency = 3.5Hz, c) forcing frequency = 7Hz, d) forcing frequency = 10.5Hz.

affects the state of flow around an airfoil, to be more specific, the extent of laminar or turbulent flow across the airfoil surface. These viscous effects are essential in establishing the aerodynamic forces of the airfoil and phenomena such as transition and laminar separation bubble can greatly influence the aerodynamic variables. In addition, turbulence intensity is an important input for computational fluid dynamics (CFD) studies.

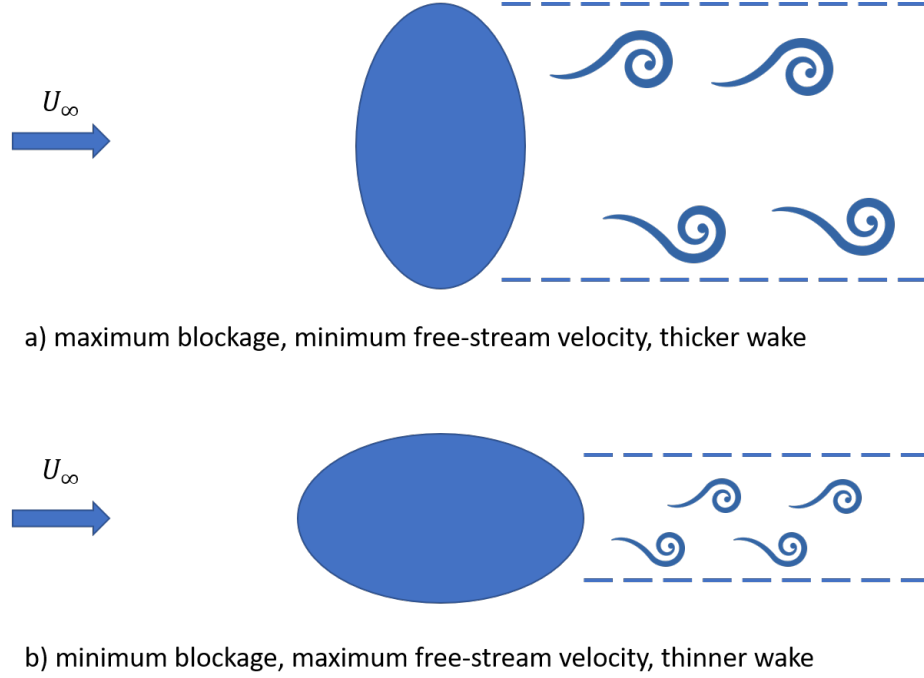


Figure 3.22: Schematic of the wake of the rotating elliptical choke vane.

The freestream velocity, especially during the unsteady experiments, can be decomposed into three parts (a triple-decomposition [65]). Equation 3.21 shows the three components of velocity

$$u(t) = \bar{u} + \tilde{u}(t) + u'(t) \quad (3.21)$$

where $\bar{u}$ is the time mean value, $\tilde{u}(t)$ is the phase dependent part due to the forcing frequency, and $u'(t)$ is the stochastic part of velocity. If each component can be isolated, then the turbulence intensity can be calculated based on Equation 3.22

$$I = \frac{u_{rms}}{\bar{u}} \quad (3.22)$$

where u_{rms} is the root-mean-square of the turbulent velocity fluctuation u' .

Because the hot-film CTA measurements are low-pass filtered at 5kHz and the fundamental frequency of the flow and its several higher harmonics are on the order of 10Hz, a Chebyshev Type II digital bandpass filter ($100\text{Hz} \leq \text{passband} \leq 5\text{kHz}$) is applied to isolate the turbulent velocity fluctuation. For the current investigation, the Mach number of the steady experiment is 0.2 and a resulting turbulence intensity was found to be approximately 1%. For unsteady cases, two approaches were attempted. First, the turbulence intensity is approximately 0.9% if using Equation 3.22 without considering the phase-dependent velocity $\tilde{u}(t)$. The slightly lower turbulence level is due to the slightly higher mean Mach number (and velocity) than the steady case. However, in unsteady cases phase-dependent velocity is a significant component, varying by approximately $\pm 25\%$ over the mean. So the second approach to evaluate the unsteady freestream turbulence intensity is by segmenting the velocity cycle into smaller portions. A 10ms interval is utilized to ensure sufficient segments for all frequencies investigated and a turbulence intensity from approximately 0.7% at the maximum velocity to 1.2% at the minimum velocity was found throughout the range of the forcing frequency, see Figure 3.23. This shows that not only the dynamic pressure but also the freestream turbulence is varying for a surging airfoil experiment. The measured turbulence intensity is higher than what was expected, perhaps due to the design of the tunnel was originally intended for high subsonic Mach number experiments where the honeycomb section and screens in the settling chamber are more effective.

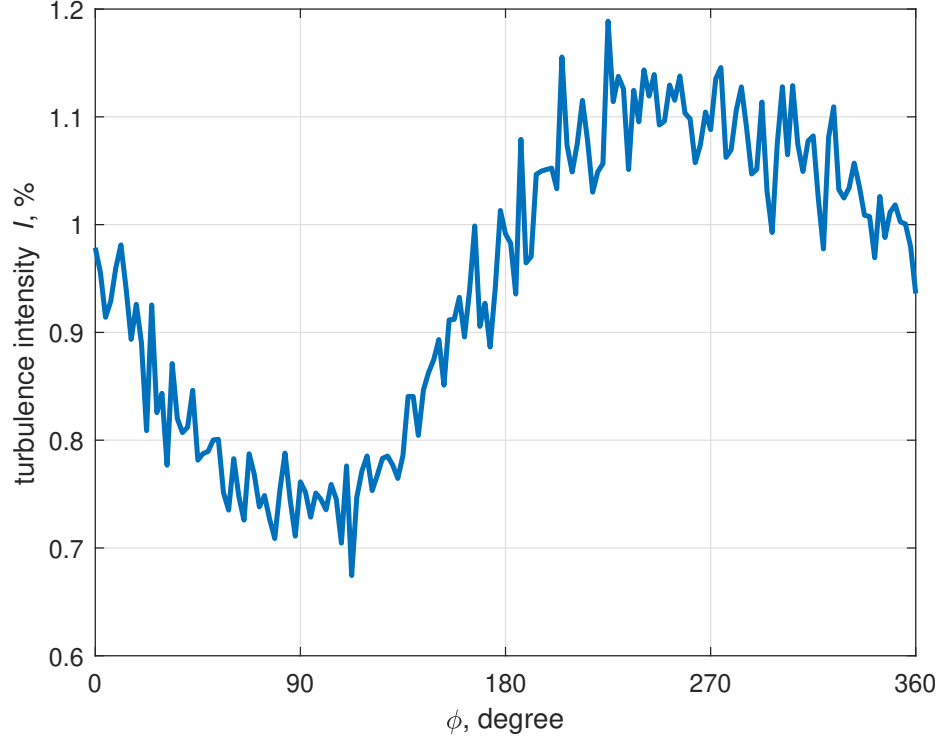


Figure 3.23: Turbulence intensity of the sinusoidally-oscillating freestream $M \approx 0.2 \pm 0.05$, $k = 0.025$.

3.5 Summary

The OSU unsteady wind tunnel and its characteristics are presented in this chapter. The pressurized blow-down type facility is capable of simulating multiple unsteady motions relevant to rotorcraft aerodynamics for high Reynolds number and compressible flow experiments. Elliptical choke vanes downstream of the test-section were designed based on an inviscid area-Mach relation to generate time-varying freestream velocity in the test-section and multiple sensors were utilized to study the freestream flow. The experimental investigation shows that wave propagation

along the tunnel, acoustic resonance inside the tunnel, and vortex shedding of the rotating choke-vane can lead to pressure distortions dependent on the forcing frequency, whereas proper dynamic compensation to the pneumatic line of the pressure measuring system is necessary. Because of the tunnel dynamics, the resulting freestream velocity can deviate from a desired sinusoidal shape. Additional limitation to produce a desired sinusoidal freestream comes from the lack of viscous effects during the choke vane design process. However, this factor cannot be easily resolved. Nonetheless, sinusoidal-like freestream can still be well produced at low frequencies. As such, it is recommended to avoid high frequency conditions to maintain a close simulation of the surging airfoil condition. In addition, accurate measurements of the freestream velocity requires the use of a hot-film CTA system accompanied with pressure and temperature measurements for unsteady experiments. An in-situ calibration of the hot-film probe was utilized in the current investigation to derive the freestream velocity. A transfer function was obtained between the isentropic relation and the hot-film measurements, which highlights when unsteady effects become significant. This transfer function can be used to correct isentropic relations, but it is recommended to always include a hot-film measurement during such unsteady experiments. Finally, the turbulence intensity of the tunnel operating under steady freestream of Mach 0.2 was found to be 1% for the turbulence bandwidth of 0.1 - 5kHz, whereas the turbulence intensity of an unsteady freestream with a mean Mach of 0.2 and advance ratio of 0.25 was found to vary between 0.7 to 1.2%. This turbulence level may promote an early airfoil boundary layer transition. Thus, flow physics associated with viscous effects on an airfoil should take this fact into consideration.

Chapter 4: Utilization of Background-Oriented Schlieren Method for Boundary Layer and Wake Measurements in A Blow-Down Type Wind Tunnel

BOS technique is a relatively new optical method to measure flow field since it was invented in 2000 [53]. Its fundamental working principle is related to first derivative of the flow density. Due to the blow-down nature of the facility in the current investigation, the technique was chosen to study the flow field of a surging airfoil and to help understand the flow physics, specifically, the development of the shear layer and the wake. This might be the first time it is utilized to measure viscous shear layer of an airfoil in a blow-down type facility. Thus, this chapter is aimed to build a bridge between the measured flow field and the physical interpretation of the flow. To better understand the causes of the density gradient and their effects in the current investigation, the characteristics of the BOS measurements are compared with an analytic and a computational model.

4.1 Density Gradients in the Flow Field

Density gradient is expected to be present around the airfoil during the experiment, both due to the blow-down nature of the current facility where there can be

heat transfer between the airfoil and the freestream flow

$$\dot{q} = -k\nabla T \quad (4.1)$$

and due to flow physics such as viscous dissipation within the shear layer

$$\Phi = \tau'_{ij} \frac{\partial u_i}{\partial x_j} \quad (4.2)$$

and shear layer vorticities

$$\omega = \nabla \times u \quad (4.3)$$

The heat transfer during a wind tunnel experiment can be evaluated by the infrared thermography. Figure 3.8 has illustrated the transient flow variables during a typical blow-down operation of the wind tunnel. Despite stagnation pressure, Reynolds number, and velocity in the test section can reach relatively steady magnitudes as the target flow condition after the tunnel is fully pressurized, the stagnation temperature continues to fall during the tunnel operation. Thus, there is a transient heat transfer between the flow and the test article. The infrared thermography measurements can shed light on flow transition and flow three-dimensionality, utilizing the different heat transfer coefficient between laminar and turbulent boundary layer. In addition, it offers information about the airfoil surface temperature during the tunnel operation.

A snapshot of the surface temperature of the NACA 0018 airfoil during the wind tunnel experiment is shown in Figure 4.1a. Only about half of the airfoil chord and span can be viewed by the IR camera in the current optical setup so two tunnel runs were carried out to measure the chord-wise temperature distribution. The circular contour in Figure 4.1a indicates the circular optical window where the higher temperature region (red in color) is outside of the wind tunnel. In Figure 4.1a, there

is a distinct temperature gradient at $x/c \approx 0.25$ across the airfoil span. The large temperature gradient indicates boundary layer transition from laminar to turbulent. Upstream of the transition, the convective heat transfer coefficient of the laminar boundary layer is smaller and as a result the airfoil surface is warmer (weaker cooling effect from the flow); while downstream of the transition, the convective heat transfer coefficient of the turbulent boundary layer is higher and the airfoil surface becomes cooler. Similar features can be found in airfoils tested by Gompertz *et al* [56]. The transition location is found to be slightly upstream than predicted by the viscous panel method [33] with a natural transition setting of $n = 9$ (the e^9 method [60]). This is likely due to the high freestream turbulence intensity in this facility (see Chapter 3) and the surface roughness of the prototyped airfoil. In addition, the staggered surface pressure taps are visible in the thermography and the surface pressure taps create a perturbation to transition the flow immediately downstream. This resulted in an additional local region of turbulent flows where the surroundings are still laminar.

Because of the pressurized blow-down nature of the wind tunnel, the flow in the test section is not ideally steady. The primary dynamic parameter is the flow temperature which continues to drop after the initial tunnel pressurization. Because of the flow temperature variation, surface temperature of the test article is also unsteady. This is illustrated in Figure 4.1b. It is observed that the temperature difference between the flow and the airfoil varies in time and is typically around 10 to 15 Kelvin for the flow condition and the test duration in the current investigation. It is necessary to emphasize that the surface temperature of the airfoil is not uniform. Nonetheless,

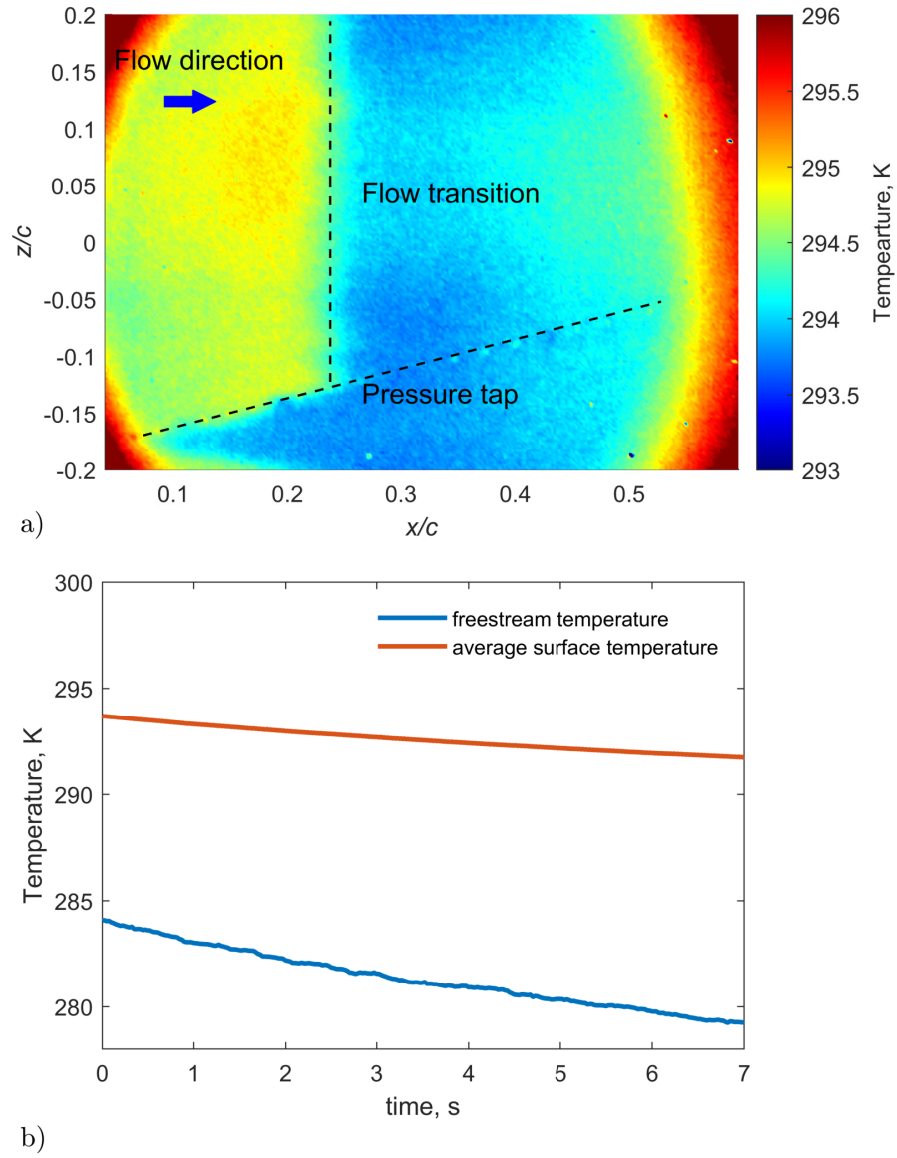


Figure 4.1: Surface temperature of steady NACA 0018 airfoil at $\alpha = 4^\circ$, a) snapshot of the instantaneous temperature field of the airfoil suction surface measured by the IR camera, b) comparison of the instantaneous freestream flow temperature and the airfoil surface temperature (averaged over the chord at mid-span).

the temperature variation across the airfoil suction surface was found to be typically no more than 2 Kelvin.

Measurements of the surface temperature provided essential information about flow transition. In addition, the heat transfer process affects the density and density gradients in the flow. Thus, it affects the BOS measurements. Moreover, the measured temperature difference helps to define the boundary condition in the CFD study. Thus, 10 and 15 Kelvin temperature difference were applied in the computational model and will be discussed in the following sections.

Assuming a symmetric airfoil at zero angle of attack, it is expected that a thermal boundary layer will form on both the suction and the pressure side of the airfoil and results in density gradient vectors symmetric about the chord line. Aft of the airfoil, the wake extends further downstream where flow past the upper and the lower surface would mix and eventually develop into the freestream. As a result, density gradients are also expected to be present in the near wake and reach zero where the upper wake and the lower wake reach an equilibrium.

Figure 4.2 shows the pixel displacement field around the NACA 0018 trailing-edge. The pixel displacement is proportional to density gradient by Equation 2.2. The Cartesian coordinates are established to match with the ground reference frame outside of the wind tunnel where the airfoil trailing-edge is always defined as $x/c = 1$ and $y/c = 0$. Only the y -component of the pixel displacement vector field is shown to highlight the zero-displacement line downstream of the trailing-edge, where displacement vector is positive above and negative below. At the current low angles of attack, the contribution of the x -component of the pixel displacement, particularly to the zero displacement line is negligible. The different direction of the displacement

vector is caused by the heat transfer between the flow and the airfoil. During tunnel operation, the cold flow is cooling the warm airfoil surface. Thus, temperature of the flow grows towards the suction surface while density of the flow decreases, leading to a positive density gradient above the suction surface. Similarly, temperature of the flow grows towards the pressure surface while density of the flow decreases, leading to a negative density gradient below the pressure surface. A comparison between the experimental BOS measurements and the CFD simulations at $\alpha = 4^\circ$ in terms of the pixel displacement is also illustrated in Figure 4.2. Both show a pixel displacement field similar to each other as expected.

In Figure 4.2a, the pixel displacement field from the experimental BOS measurement was averaged over the steady tunnel operation. Large pixel displacements are distinct near the airfoil surface and approximately $0.1c$ thick on the suction surface and $0.05c$ thick on the pressure surface. The layer of large pixel displacement can also be observed to grow along the airfoil surface, analogous to a boundary layer. These displacement layers continue aft of the airfoil representing the airfoil wake. In Figure 4.2b, the pixel displacement field from the CFD simulation was created with numerical density gradients and a scaling factor based on the experimental setup following Equation 2.2. There are remarkable similarities between the experimental BOS and the CFD simulation, except that the background noise (density gradients further away from the airfoil) is significantly smaller and displacement patterns near the airfoil surface are more refined in the CFD results. These differences are due to vibrations of the wind tunnel and the spatial resolution available in the optics for experimental measurements, respectively. As a result, a polynomial surface fit was established based on the displacement further away from the airfoil in every frame

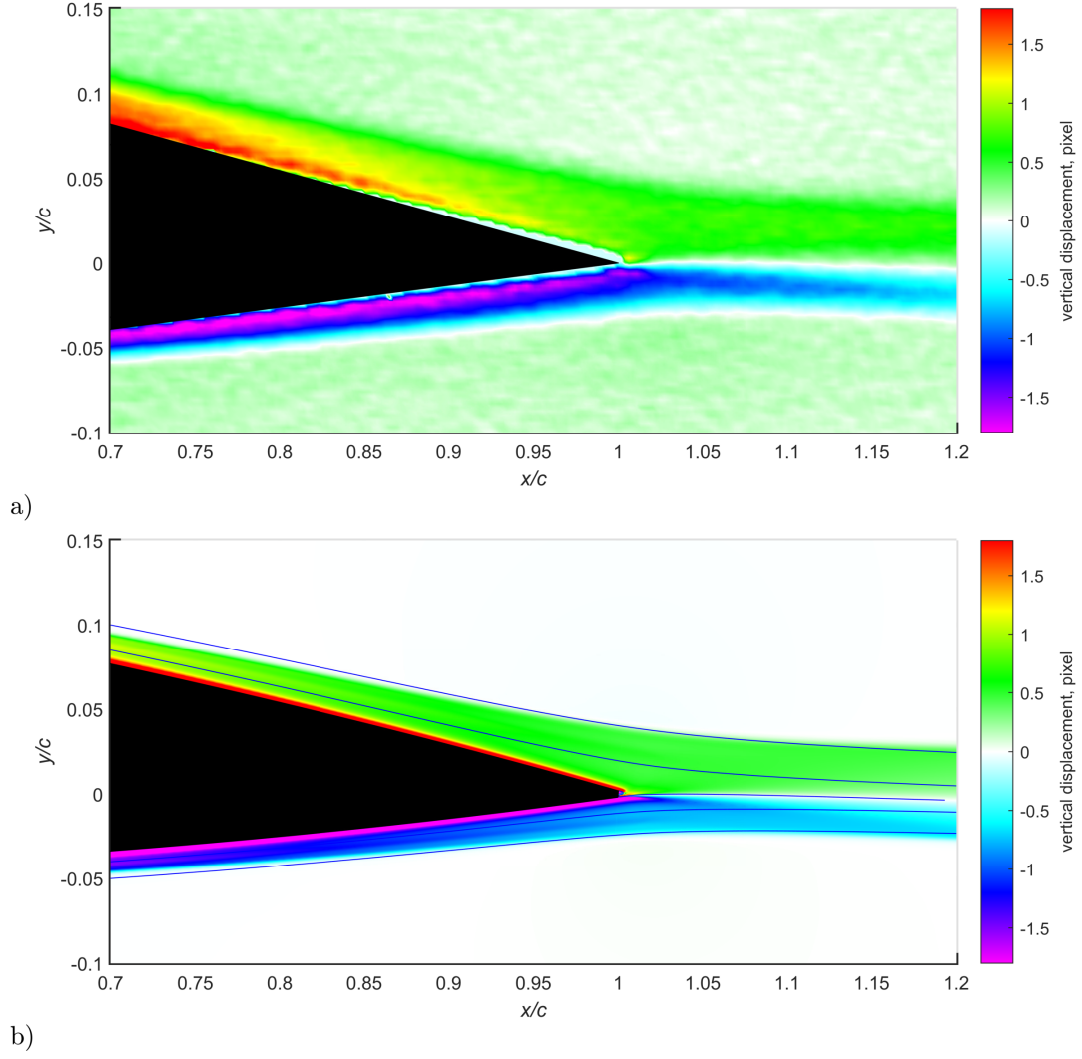


Figure 4.2: Steady pixel displacement field around the airfoil trailing-edge, $\alpha = 4^\circ$, a) from experimental BOS measurements, b) from CFD simulation with highlighted streamlines.

and was offset from the raw displacement field. This was found to greatly improve the quality of the data without biasing the features. In Figure 4.2b, a few streamlines are plotted on top of the displacement field to validate the zero-displacement line as the airfoil trailing-edge stagnation streamline.

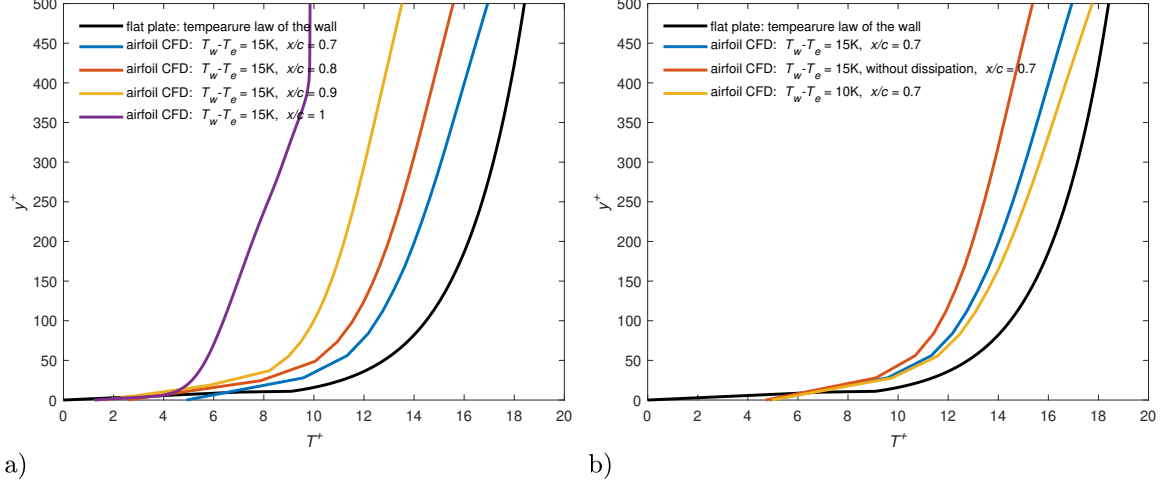


Figure 4.3: Normalized temperature profiles from the suction surface of the airfoil, $\alpha = 4^\circ$, a) effects of streamwise location, b) thermal effects.

4.2 Characteristics of the Steady Boundary Layer

To further analyze the flow around the airfoil surface which results in the displacement field, an analytical model and the two-dimensional RANS CFD were utilized where the fundamental parameters can be assessed. The analytical model is based on the temperature law of the wall for turbulent flow over an isothermal flat plate with zero pressure gradient [60, 66]. The results are shown in Figure 4.3. The dimensionless parameters are defined as

$$y^+ = \frac{yv^*}{\nu} \quad (4.4)$$

$$T^+ = (T_w - T) \frac{\rho c_p v^*}{q_w} \quad (4.5)$$

where $v^* = \sqrt{\tau_w/\rho}$ is the wall-friction velocity, τ_w is the wall shear stress, ν is the kinematic viscosity, T_w is the wall temperature, T is the freestream flow temperature, c_p is the specific heat of the flow, and q_w is the wall heat flux.

The temperature law of the wall T^+ vs. y^+ for the flat plate case in Figure 4.3 follows the linear relation in the thermal sublayer

$$T^+ = y^+ Pr \quad (4.6)$$

where Pr is the Prandtl number of air, and the logarithmic relation in the logarithmic layer is

$$T^+ \approx \frac{Pr_t}{\kappa} \ln y^+ + A(Pr) \quad (4.7)$$

where Pr_t is the turbulent Prandtl number and is assumed to be 1, κ equals 0.41, and A is a curve-fit function

$$A(Pr) \approx 13Pr^{2/3} - 7 \quad (4.8)$$

valid for $Pr \geq 0.7$ or

$$A(Pr) \approx (3.85Pr^{1/3} - 1.3)^2 + 2.12 \ln Pr \quad (4.9)$$

given by Kader [67].

The temperature law of the wall from the CFD simulation at selected airfoil streamwise locations are plotted in Figure 4.3 following the definition of y^+ and T^+ in Equation 4.1. For convenience, properties plotted in the following figures from the CFD or the BOS data were sampled along the global y -axis instead of normal to the local surface. Nonetheless, results show similar profiles as the flat plate case. The differences in the magnitude of y^+ and T^+ are primarily due to the adverse pressure gradients along the airfoil surface. It was found in the CFD simulation that wall shear stress at the trailing-edge can be one to two orders of magnitude smaller than further upstream. It was also found that for the relatively thick airfoil separation at the trailing-edge is imminent at angles of attack as low as 4 degrees. In Figure

4.3b, thermal effects are highlighted with different $T_w - T_e$ (where T_e is the thermal boundary layer edge temperature) and if the viscous dissipation term is included. The greater T^+ when $T_w - T_e = 10K$ compared to $T_w - T_e = 15K$ is primarily due to the smaller wall heat flux, whereas the smaller T^+ without calculating dissipation is due to altered temperature distribution within the thermal boundary layer. This comparison shows that viscous dissipation within the shear layer is an important part of the temperature profile and the effect of heat transfer (convective heat transfer in the current analysis) can greatly affect the thermal characteristics. In the wind tunnel experiment, there are additional heat transfer terms, *i.e.*, conduction in the airfoil model and radiation from the tunnel walls. However, these effects are treated negligible in the current discussion.

The other fundamental fluid parameter, density, is examined in terms of density boundary layers in Figure 4.4. Density is normalized by the freestream condition and the distance is normalized by the airfoil chord for better association with Figure 4.2. The $y/c = 0$ in Figure 4.4 is defined from the local airfoil surface for each streamwise location instead of the global $y/c = 0$. The temperature law of the wall for the turbulent flow over a flat plate is transferred into density by assuming a one-seventh power-law turbulent velocity profile [60], applying the experimental condition (ρ_∞, T_∞) as the freestream condition, and using the airfoil chord as the characteristic length. Thus, all intermediate parameters can be calculated.

Figure 4.4a shows the growth of the density boundary layer along the chord-wise direction. The derived density profile from the analytic flat plate case still exhibits a good approximation of the computational results. The thickness for $\rho/\rho_\infty = 0.99$ matches very well between the analytic model and the CFD results. Thermal effects

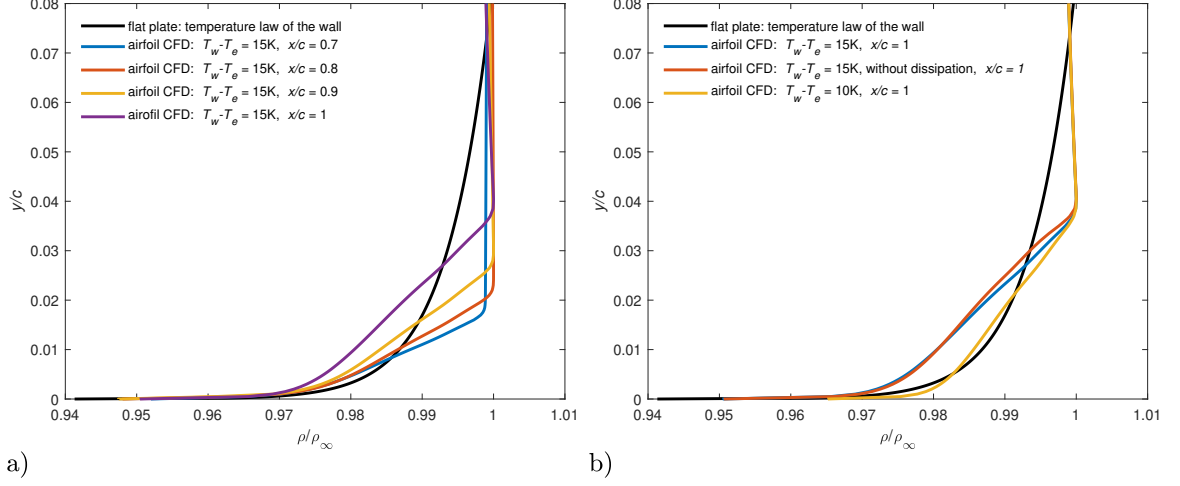


Figure 4.4: Normalized density profiles from the suction surface of the airfoil, $\alpha = 4^\circ$, a) effects of streamwise location, b) thermal effects.

are assessed again in terms of density profiles in Figure 4.4b. The dominant effect of the convective heat transfer to the density profile is illustrated between the 15K and 10K temperature difference. The effect of viscous dissipation to the local density distribution is demonstrated, similar to the temperature profiles discussed earlier.

Next, the density gradient profile is evaluated and shown in Figure 4.5. The analytic model predicts a growing density gradient toward the wall. This large density gradient near the wall is expected following the turbulent thermal boundary layer and density boundary layer seen in Figure 4.3 and Figure 4.4, respectively, which are analogous to the large velocity gradient near the wall in a turbulent velocity boundary layer. Profiles from the CFD simulation presents a similar pattern where it is observed that the thickness of the density gradient profile grows along the airfoil while the magnitude of the gradient reduces. However, near the edge of the profile, the magnitude of the density gradient did not follow the logarithmic curve of the

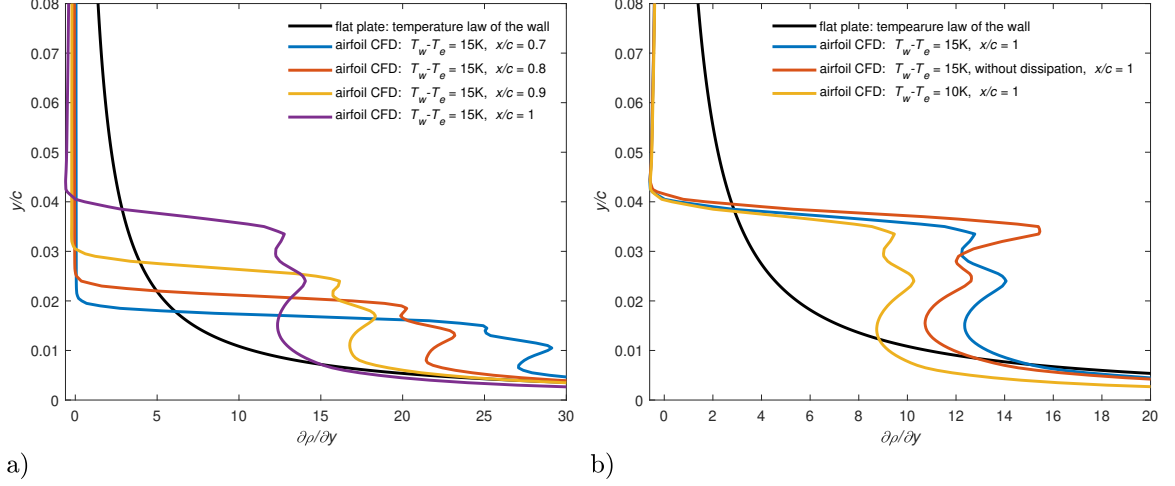


Figure 4.5: Density gradient profiles from the suction surface of the airfoil, $\alpha = 4^\circ$, a) effects of streamwise location, b) thermal effects.

analytic model. This is due to the difference in density (and temperature) profiles shown in Figure 4.4 and also the pressure profile extended from the airfoil surface where it is assumed constant in the flat plate case.

The thermal effects to the density gradient are shown in Figure 4.5b and it is observed that the temperature difference only affected the density gradient magnitude and the profile is ‘self-similar’. On the contrary, because dissipation effects vary within the shear layer, the density gradient distribution is changed and a lower magnitude is observed near the wall while a greater magnitude is observed near the edge of the thermal boundary layer.

From this point, the pixel displacement due to density gradient can be obtained with a scaling factor given in Equation 2.2 assuming the density gradient is uniform along the light path (*i.e.*, flow is two-dimensional). Notice that the unit of Δy in Equation 2.2 is meter and the unit in Figure 4.2 is pixel. An additional conversion

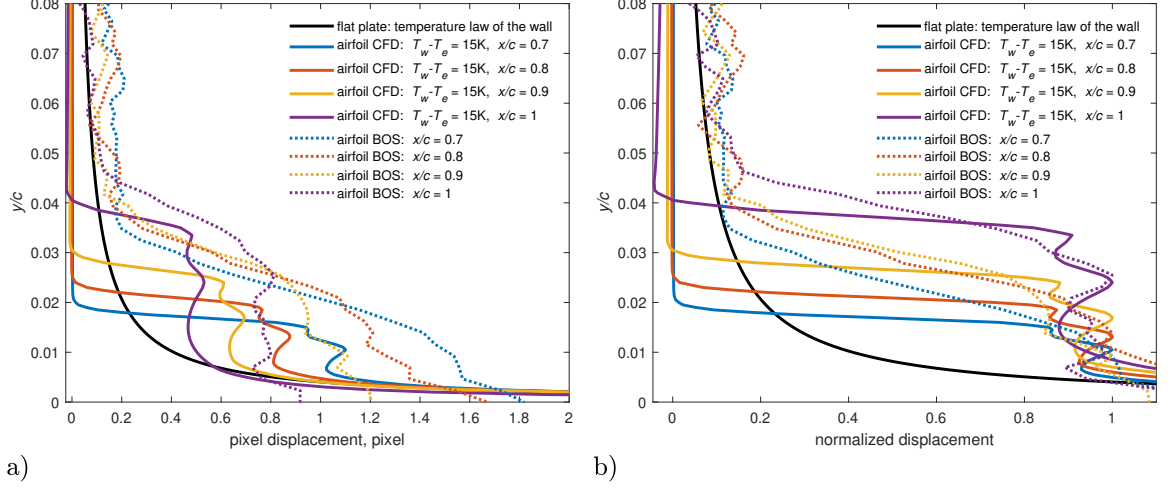


Figure 4.6: Pixel displacement profiles from the suction surface of the airfoil, $\alpha = 4^\circ$, a) comparison of the analytic model, the CFD simulation, and the BOS measurements, b) displacement profiles normalized by local maximum magnitude.

factor is acquired by measuring the physical length of the pixels captured by the camera. The overall conversion is applied to the CFD results and the pixel displacement profiles are shown in Figure 4.6, along with a direct comparison to the experimental BOS results.

Figure 4.6 shows that the displacement profile thickness grows along the chord while the magnitude decreases. These have been observed in Figure 4.2 and the global trend matches with the computational results. It is encouraging that there is a general agreement between the experimental measurements and the computational model about the displacement profile and its thickness. This is further demonstrated in Figure 4.6b where the profile is normalized by the local maximum in the shear layer to compare the profile shapes. For example, displacement layer at $x/c = 1$ from experimental measurements and CFD reach the farfield condition at $y/c \approx 0.04$ and share a similar profile for $y/c \leq 0.04$. However, there are a few noticeable

differences between the experimental BOS and the CFD simulation. One is that the experimental BOS does not resolve the displacement near the wall. This is due to the spatial resolution available of the camera and overall limitations of the optical setup (such as image blur surrounding the airfoil surface). The other difference is the magnitude of the pixel displacement. Many factors can contribute to this and are listed below.

Firstly, flow over the airfoil in the experiment is initially laminar and then transitions to turbulent (see Figure 4.1a) whereas the turbulence model used in CFD assumes fully turbulent flow from the leading-edge. This difference affects the convective heat transfer since turbulent flow has a greater convective heat transfer coefficient than the laminar flow. Secondly, flow over the airfoil is three-dimensional in the experiment whereas the computational model is two-dimensional. The three-dimensional flow is due to horseshoe vortex created at the intersection of the airfoil and the tunnel sidewalls and due to the transition affected by the surface pressure taps (which only contributes to a small region before the global transition over the airfoil surface, see Figure 4.1a). Both of these effects depend on the freestream flow condition and the angle of attack. Because BOS method utilizes the density gradient integrated along the light path, errors are introduced when assuming a uniform density gradient along the light path to convert the density gradient into the pixel displacement. Thirdly, the computational model assumes constant temperature of the airfoil (isothermal) and a constant temperature difference between the surface and the flow. However, both are dynamic during experiments and vary along the airfoil surface (see Figure 4.1). However, matching all of the experimental conditions in the CFD and improving the fidelity (such as using Large-Eddy Simulation (LES))

for turbulent flow calculations) is expensive and the goal here is to provide immediate insights of the flow physics and to validate the experimental BOS measurements. The relatively simple analytic model and the two-dimensional RANS CFD have demonstrated reasonable comparison to the experimental measurements and can serve as a first approximation tool for such flow scenarios.

4.3 Characteristics of the Steady Wake

The airfoil boundary layer flow has been evaluated by comparing the analytic model, the CFD simulation, and the BOS measurements. Another very important flow feature of interest is the airfoil wake. In Figure 4.2, it has been shown that the wake flow near the airfoil trailing-edge is also captured in the pixel displacement field. In Figure 4.2, the zero-displacement line is identified in the wake, both from the computational result and from the BOS measurements. The overlap of the trailing-edge stagnation streamline and the zero-displacement line has been demonstrated. A comparison of the zero-displacement line between the CFD simulation and the BOS measurement is plotted in Figure 4.7 for various angles of attack. The comparison shows a good agreement between the two. In general, it is observed that the stagnation streamline curves down as the angle of attack increases. Near the vicinity of the airfoil trailing-edge (*i.e.*, $x/c < 1.05$), however, the stagnation streamline is found to curve up in both the CFD simulation and the BOS measurement. This is believed to be due to a pressure gradient present in this region of strong viscous interaction right aft the airfoil trailing-edge [36]. In other words, flow field in this experiment can be viewed as an effective airfoil defined in a potential flow field, in which the effective airfoil is the physical airfoil plus the displacement thickness of the boundary layer

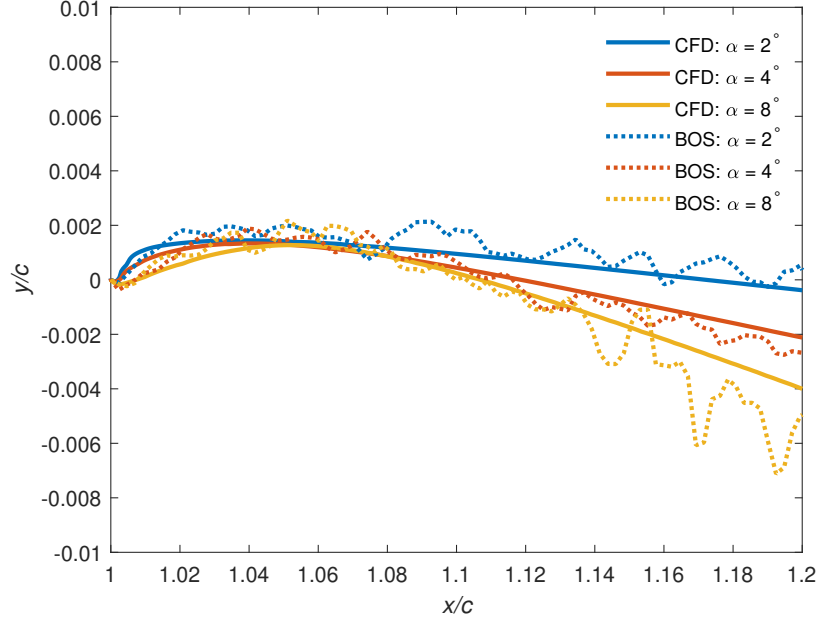


Figure 4.7: Comparison of the zero-displacement line from CFD simulations and BOS measurements as a function of the angle of attack.

and the trailing-edge of the effective airfoil (in a corresponding inviscid flow field) is at $x/c \approx 1.05$ where Kutta condition applies.

Aft of the trailing-edge, the wake profile in terms of the pixel displacement magnitude is plotted in Figure 4.8. The development of the wake flow is shown at two downstream locations. The magnitude of the displacement in the wake is shown to decrease as it moves further downstream because of the mixing process in the wake. In addition, effects of the angle of attack show that flow from the lower surface possesses a stronger density gradient than the flow from the upper surface, also see Figure 4.9. As the angle of attack grows, the upper wake (in terms of density gradient) becomes weaker while the lower wake becomes stronger. This is likely due to a strong adverse pressure gradient on the suction side to reduce the local velocity. As a result, the

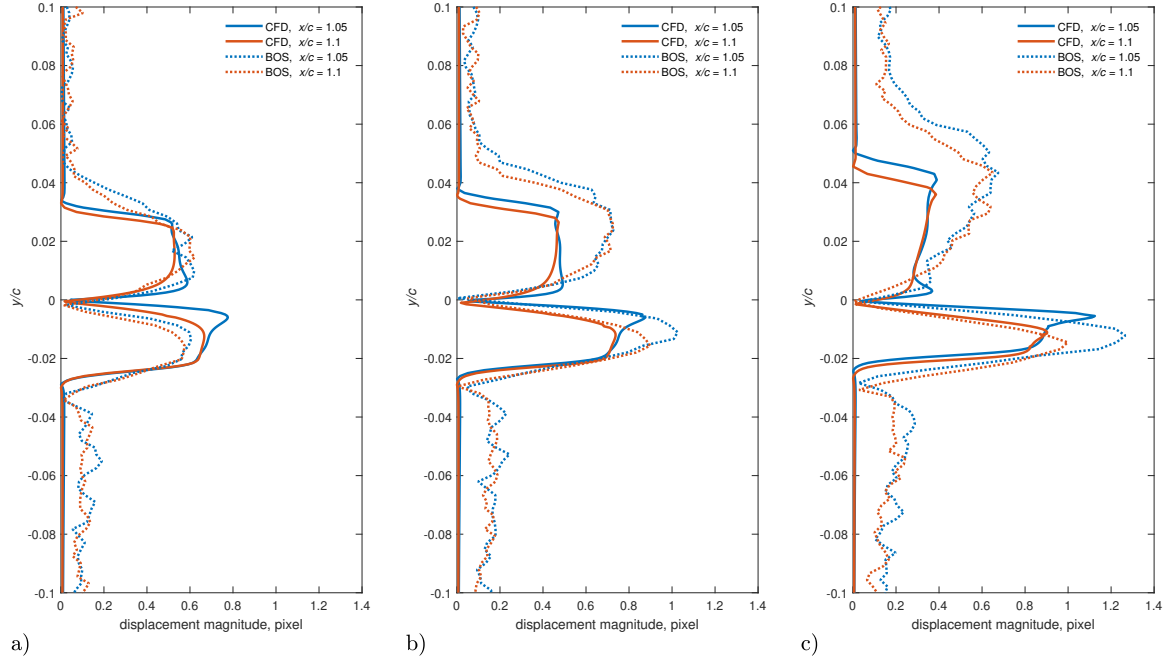


Figure 4.8: Comparison of the wake profile in terms of the absolute displacement magnitude from CFD simulations and BOS measurements, a) $\alpha = 2^\circ$, b) $\alpha = 4^\circ$, c) $\alpha = 8^\circ$.

heat transfer effect is also weakened in the upper wake relative to that in the lower wake. Similar to the flow profiles around the airfoil surface, the wake profiles from the CFD simulation show a good agreement with the BOS measurements.

4.4 Summary

The BOS technique is developed to characterize the flow field by utilizing the change in index of refraction caused by density gradient in the flow field. The technique is applied to measure the flow field of a NACA 0018 airfoil in a blow-down type wind tunnel in the current experimental investigation. As a result, this chapter is

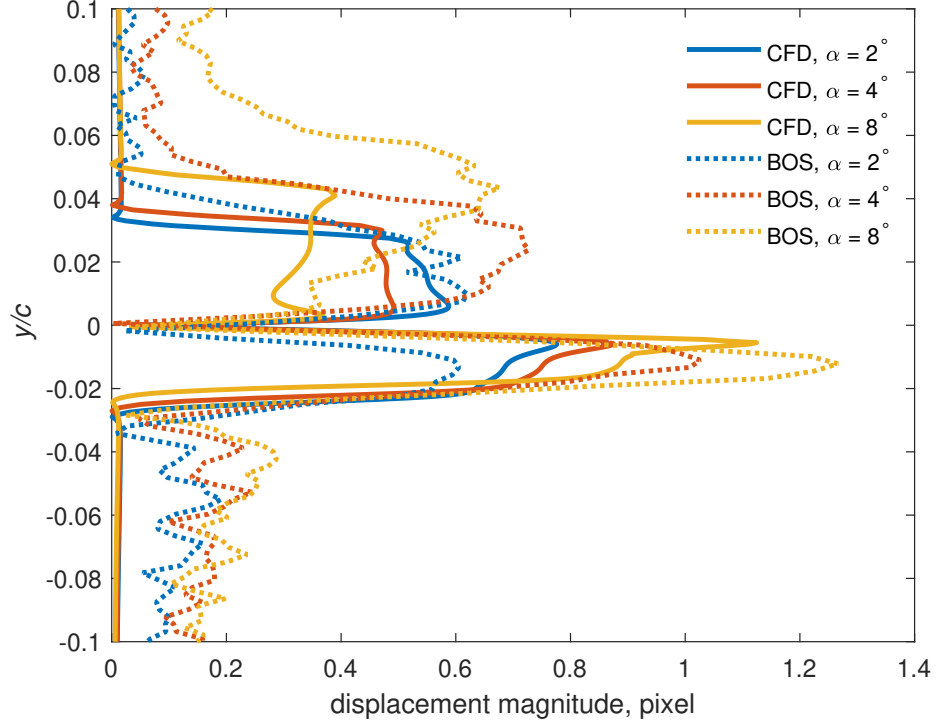


Figure 4.9: Comparison of the wake profile at $x/c = 1.05$ in terms of the absolute displacement magnitude from CFD simulations and BOS measurements as a function of the angle of attack.

intended to interpret the measured flow field in such application. From BOS measurements, the displacement field is proportional to the density gradient, which is caused by convective heat transfer between the warmer airfoil and the colder freestream flow, dissipation in the local shear layer, and shear layer vorticities. As a result, displacement vectors (or density gradient) are found to form along the airfoil surface and in the wake, with opposite directions with respect to the wake centerline. An analytic model based on temperature law of the wall for an isothermal flat plate with zero pressure gradient and turbulent boundary layer, and a CFD simulation of the experimental airfoil condition are compared with the experimental BOS measurements.

The comparison not only verified the displacement field in the local thermal boundary layer and the wake, but also highlight the thermal effects to the density gradient. Overall, it is found that the convective heat transfer is the dominant effect in establishing the local density gradient and the wake centerline can be identified by the BOS technique.

Chapter 5: Validation of the Classic Surging Airfoil Theory

This chapter is focused on validating Isaacs' surging airfoil theory experimentally with a NACA 0018 and a NACA 63-015A airfoil at high Reynolds number. The NACA 0018 airfoil was selected to compare with the investigation by Strangfeld *et al.* [7], which was at Reynolds numbers an order of magnitude smaller. The higher Reynolds number in this investigation was achieved with the unique capability of the pressurized blow-down type unsteady wind tunnel, but also serves to have a closer satisfaction of Isaacs' inviscid theory to prevent any flow separation, as suggested by Strangfeld *et al.* [7]. In this chapter, both the steady and the unsteady lift coefficients of the airfoil are sampled and discussed, with optical measurements to characterize the flow field.

5.1 Test Matrix and Uncertainty Analysis

The test matrix of the surging airfoil experiments is listed in Table 5.1. The test matrix is designed to achieve the goal of verifying Isaacs' surging airfoil theory experimentally, but also take into account the limitation of surging flow condition from the wind tunnel characterization study in Chapter 3. Specifically, the distortion to the sinusoidal freestream velocity grows with the reduced frequency and the oscillation amplitude. Thus, it can be seen that only one pair of oscillation amplitude ($\sigma = 0.25$)

Table 5.1: Test matrix of the surging airfoil experiment.

Tunnel operation	$\overline{Re}_c$	M_∞	α , deg	k
Steady	1.5×10^6	0.2	0 – 16	-
Quasi-steady	$(1.125 - 1.875) \times 10^6$	0.15 – 0.25	0 – 6	-
Unsteady	1.5×10^6	0.2 ± 0.05	0 – 6	0.025

and reduced frequency ($k = 0.025$) was thoroughly investigated. For the test matrix, three tunnel operating modes were conducted, namely steady, quasi-steady, and unsteady freestream. The steady case serves as the reference where the static lift coefficient of the airfoil was investigated. The quasi-steady case is achieved by operating the tunnel in the steady mode at several selected Reynolds- and Mach-number combinations based on the target unsteady flow condition. The unsteady case is the surging airfoil condition where the freestream condition (Reynolds number, Mach number, and velocity) is harmonically varying in time. The reduced frequency was chosen to be sufficiently high without significantly deviating from a sinusoidal waveform. The operating duration of an unsteady experiment was 30s to allow sufficient cycles (approximately 90) for phase-averaging. Overall, this experimental setting was a compromise between sufficient unsteady cycles and the transient flow condition in the pressurized blow-down facility (*i.e.*, dropping in tunnel stagnation temperature and pressure over time).

In Chapter 3, freestream static and stagnation pressures were measured by a pitot-static probe mounted in the tunnel sidewall, located at the spanwise center of the test-section $3c$ upstream and $1.3c$ below the airfoil. Stagnation temperature was measured by a type K thermocouple positioned in the upstream settling chamber. The

Table 5.2: Uncertainty of the experimental measurements.

Parameters	P , Pa	T , K	Re_c	M_∞	C_p
Uncertainty	± 300	± 0.7	± 28000	± 0.004	± 0.0004

time-resolved freestream Mach number can be calculated by applying the isentropic relation for steady and quasi-steady operations. For unsteady cases, the investigated frequency was sufficiently low to satisfy the quasi-steady assumption so the isentropic relation was also applied. For best accuracy, correction was made based on the hot-film investigation in Chapter 3.

For airfoil load measurements, time-resolved $C_l(t)$ was calculated through trapezoidal integration of $C_p(x, t)$. The pressure drag and pitching moment can also be calculated but were not parameters of interest in the current work. For steady and quasi-steady cases the time-resolved parameters were time-averaged, whereas in unsteady experiments the time-resolved parameters were phase-averaged with respect to the freestream velocity cycle. An uncertainty analysis using the Monte-Carlo method [58] with a Gaussian distribution was conducted to both the freestream and local measurements. The uncertainty of the input variables (*i.e.*, pressure and temperature) are provided by the manufacturer of the instruments, whereas the uncertainty of the output variables (*e.g.*, Re_c, M_∞, C_p) are calculated based on the uncertainty of the input variables and the mean flow condition in the current investigation. The uncertainty of these parameters are listed in Table 5.2.

5.2 Characteristics of Steady Lift Coefficients

The steady lift characteristic of the NACA 0018 and NACA 63-015A airfoil were evaluated with surface pressure measurements at $Re_c = 1,500,000$ and $M_\infty = 0.2$. To satisfy the small disturbance assumption in Isaacs' theory, small angles of attack were the primary focus of the experimental investigation.

The lift curve of the NACA 0018 airfoil is shown in Figure 5.1, along with thin airfoil theory [8] and prediction from a two-dimensional panel code that includes viscous effects [33]. It is found that the experimentally-measured lift coefficients match very well with the predictions in terms of the $C_l - \alpha$ trend, where the C_l is linear up to $\alpha = 8^\circ$. However, it is clearly observed that the experimental C_l data has a smaller lift curve slope ($\approx 5.65 \text{ radians}^{-1}$) than thin airfoil theory ($2\pi \text{ radians}^{-1}$). This could be due to the thickness of the airfoil being considered 'thick', where boundary layer growth can affect flow separation and lift [68, 69]. However, similar $C_l - \alpha$ is found on the thinner NACA 63-015A airfoil, shown in Figure 5.2. In Figure 5.2, steady lift data are measured for the NACA 63-015A airfoil until static-stall is encountered. The experimental C_l is linear up to $\alpha = 8^\circ$ and the $C_{l,max}$ occurs around $\alpha = 14^\circ$. The measured lift curve slope in the linear range is approximately 5.8 radians^{-1} , also smaller than thin airfoil theory and the panel method.

Steady C_p distribution of the airfoil is examined to better understand the steady flow physics of the airfoil and the measured lower lift curve slope. Figure 5.3 shows the steady C_p distribution of the NACA 0018 airfoil at selected angles of attack. A few of the tap data were corrupted due to air leaks along the pressure line and were neglected. Overall, it is observed that the suction-side pressure decreases with the growing angles of attack while the pressure-side pressure increases. Transition

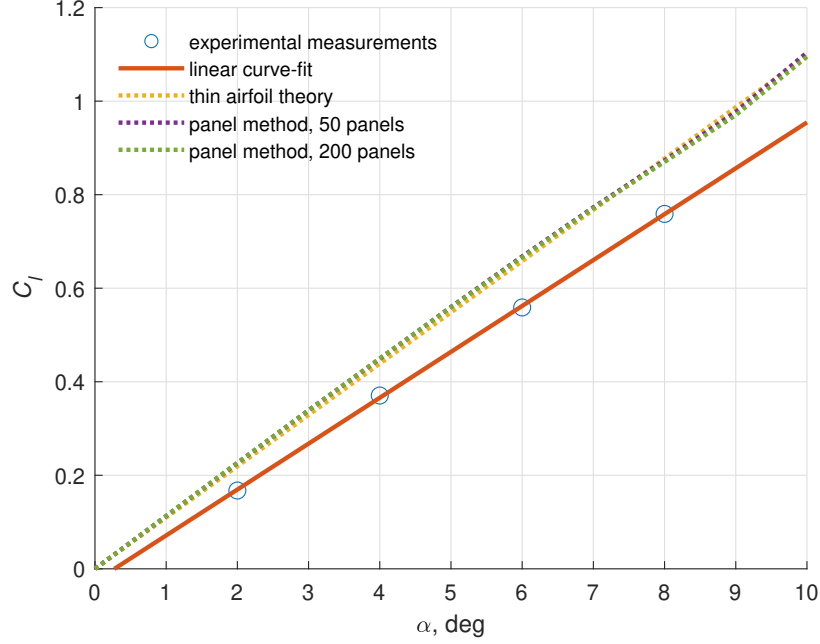


Figure 5.1: $C_l - \alpha$ of the NACA 0018 airfoil under steady freestream conditions.

behavior can not be clearly identified from the surface pressure data due to the limited tap resolution. One method to identify flow transition is to compare with the data from the panel code [33], shown in Figure 5.4. After examination, the $n = 4$ transition setting in the panel code was found to exhibit the best agreement with the experimental measurements. The change from default $n = 9$ (the e^9 method [60]) to $n = 4$ indicates a stronger freestream disturbance or a stronger local instability during the experiment that can trigger early transition, which is consistent with the freestream turbulence measurements in Chapter 3 and the surface roughness of the rapid-prototyping airfoil, respectively. In addition, Figure 5.4 highlights that the surface pressure tap data are not able to always capture the suction peak and the leading-edge stagnation pressure. This leads to a reduction in measured lift coefficient and could be the reason for reduced lift curve slope.

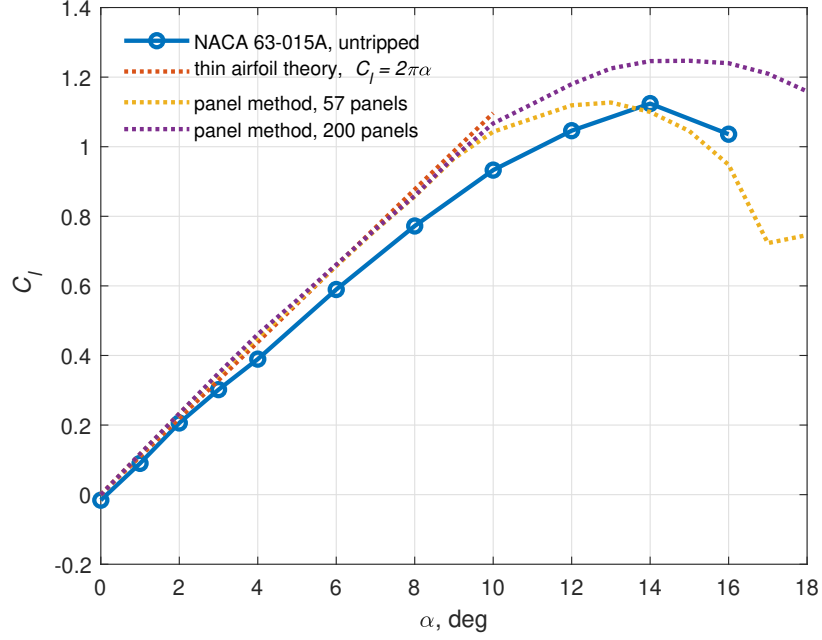


Figure 5.2: $C_l - \alpha$ of the NACA 63-015A airfoil under steady freestream conditions.

Similarly, the steady C_p distribution of the NACA 63-015A airfoil from $\alpha = 1^\circ$ to $\alpha = 6^\circ$ is plotted in Figure 5.5. The C_p distribution shows how the suction-side and the pressure-side pressure evolve with the angle of attack. When the airfoil is at very low angles of attack ($\alpha \leq 2^\circ$), the suction peak is mild and a plateau of low pressure can be found to expand from approximately $x/c = 0.05 - 0.3$. When the angle of attack reaches $\alpha = 3^\circ$, a distinct suction peak begins to appear near the leading-edge and grows rapidly with the angle of attack. A comparison to the panel code with $n = 4$ is also conducted and is shown in Figure 5.6. Figure 5.6 also highlights the limitation of utilizing finite surface pressure taps to capture the suction peak ($x/c < 0.017$ on the suction side) and the leading-edge stagnation pressure ($x/c < 0.017$ on the pressure side). Thus, the measured lift coefficient and the lift curve slope is smaller than the actual value.

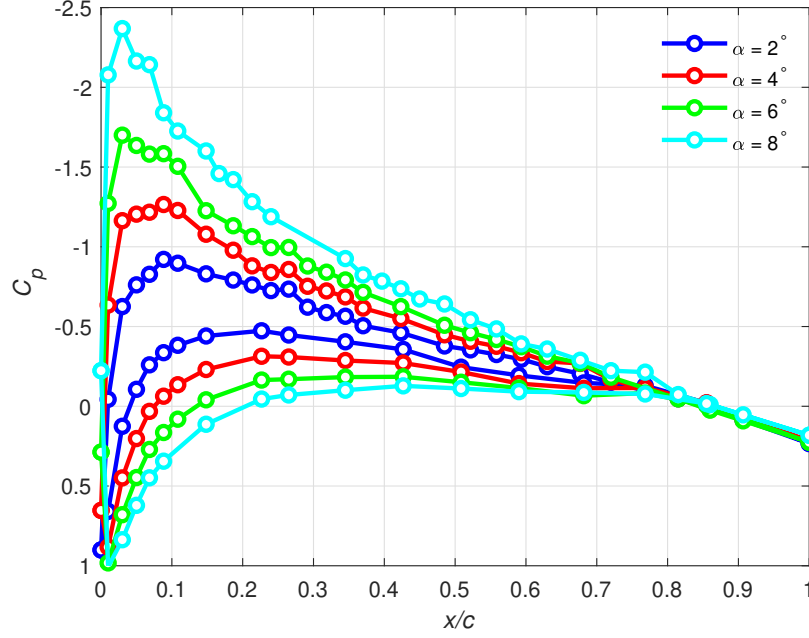


Figure 5.3: Experimentally-measured C_p distribution of the NACA 0018 airfoil under steady freestream conditions.

Besides surface pressure distribution, flow transition can be identified by the IR measurements, discussed in Chapter 4 and shown in Figure 4.1a for NACA 0018 airfoil at $\alpha = 4^\circ$. Similarly, the transition measurements were conducted on the NACA 63-015A airfoil and an example is shown in Figure 5.7. In Figure 5.7, the spanwise transition is found to occur around $x/c = 0.15$, illustrated in both the contour plot of the surface temperature and the temperature along the chord at mid-span. A comparison of predicted and measured flow transition on the upper surface of the NACA 63-015A airfoil is listed in Table 5.3. The IR measurements of transition are found to agree very well with the prediction from the panel code with $n = 4$, which indicates an earlier flow transition on the airfoil due to the freestream turbulence and the airfoil surface roughness.

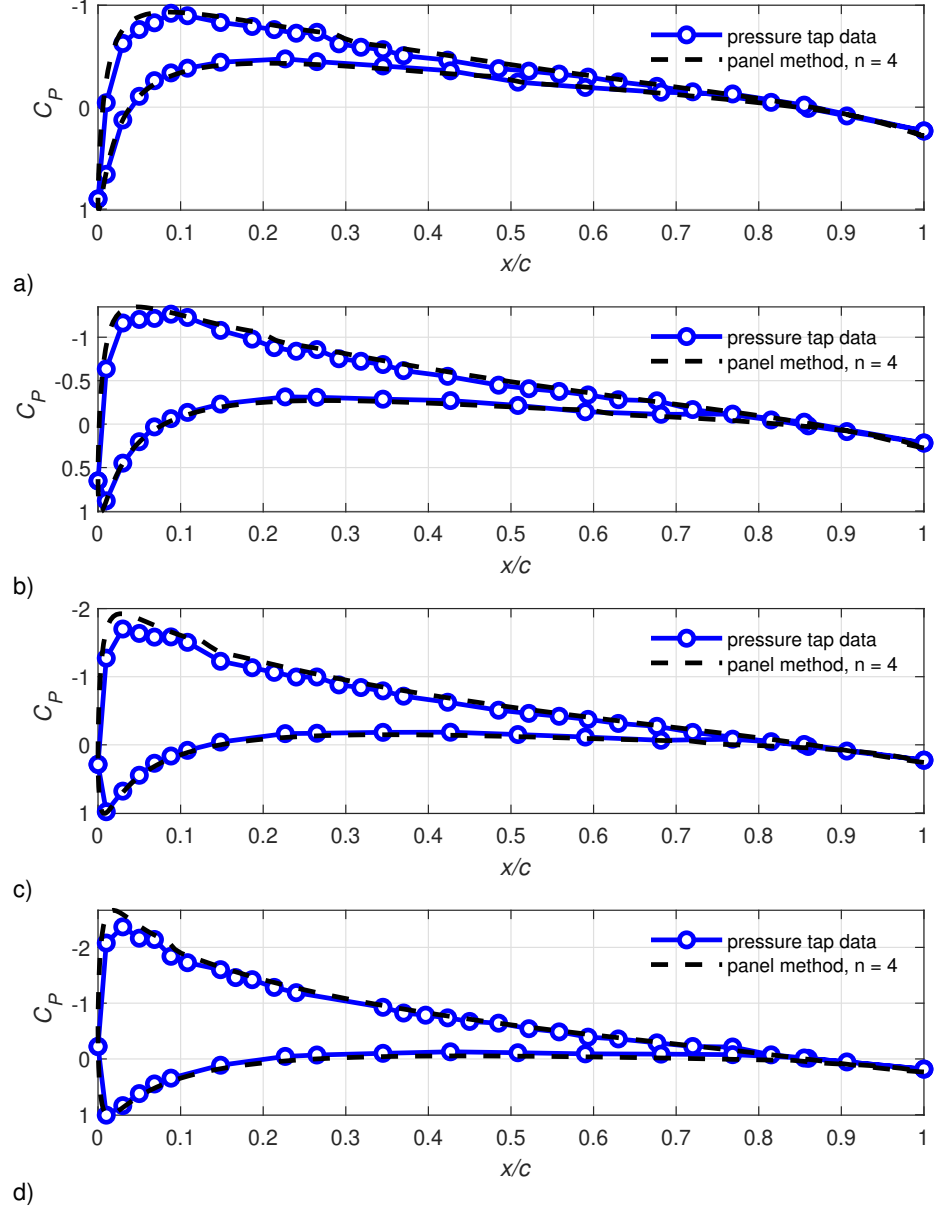


Figure 5.4: Comparison of C_p distribution between analytic prediction and experimental measurements of the NACA 0018 airfoil under steady freestream conditions, a) $\alpha = 2^\circ$, b) $\alpha = 4^\circ$, c) $\alpha = 6^\circ$, d) $\alpha = 8^\circ$.

In addition, the steady flow field is examined with the BOS method. A time-averaged displacement field of the NACA 0018 airfoil under steady freestream is

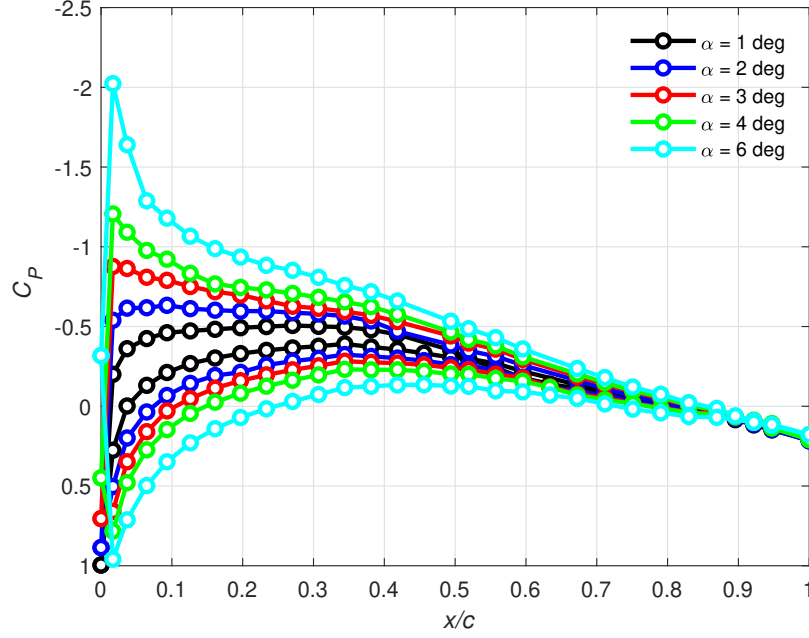
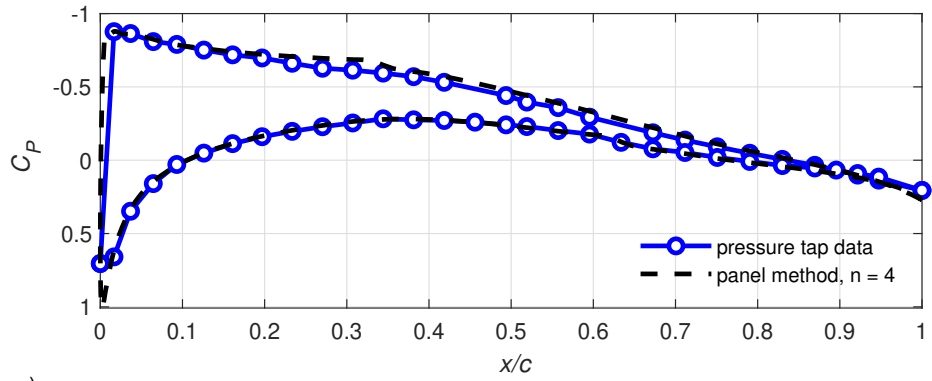


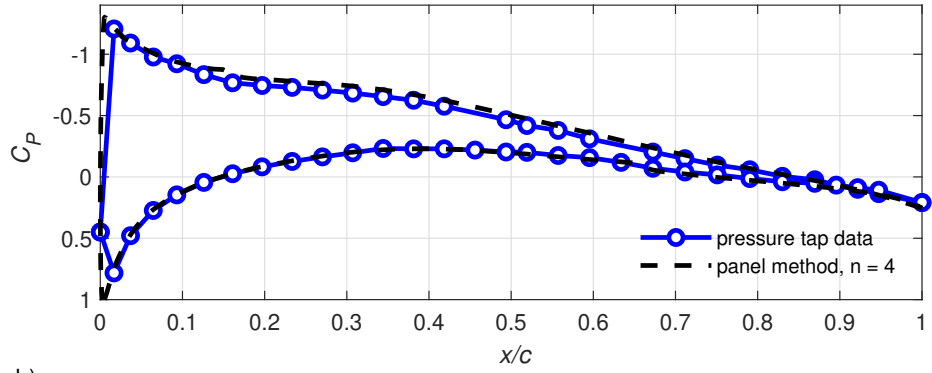
Figure 5.5: Experimentally-measured C_p distribution of the NACA 63-015A airfoil under steady freestream conditions.

already shown in Figure 4.2 and is re-plotted in Figure 5.8, along with the displacement field of the steady NACA 63-015A airfoil. It is observed that the shear layer grows along the airfoil surface with opposite displacement magnitudes and in the wake downstream. The zero-displacement line represents the centerline of the wake where the two shear layers mix and reach homogeneity and thus it represents the stagnation streamline. These are the key features from the current BOS measurements, discussed earlier in Chapter 4.

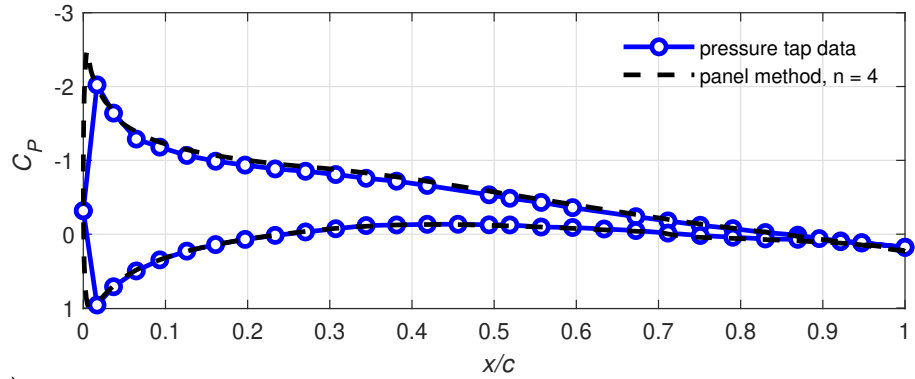
From BOS measurements, a very important flow structure is the trailing-edge stagnation streamline. The trailing-edge stagnation streamline as a function of the airfoil angle of attack is illustrated in Figure 5.9. It is observed that there is a local curvature change very near wake ($x/c \leq 1.05$) with respect to the entire stagnation



a)

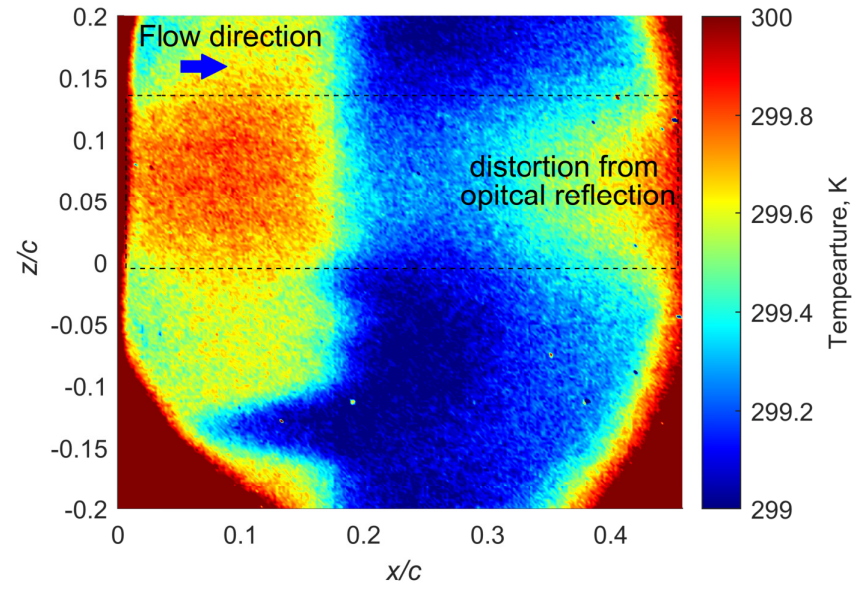


b)

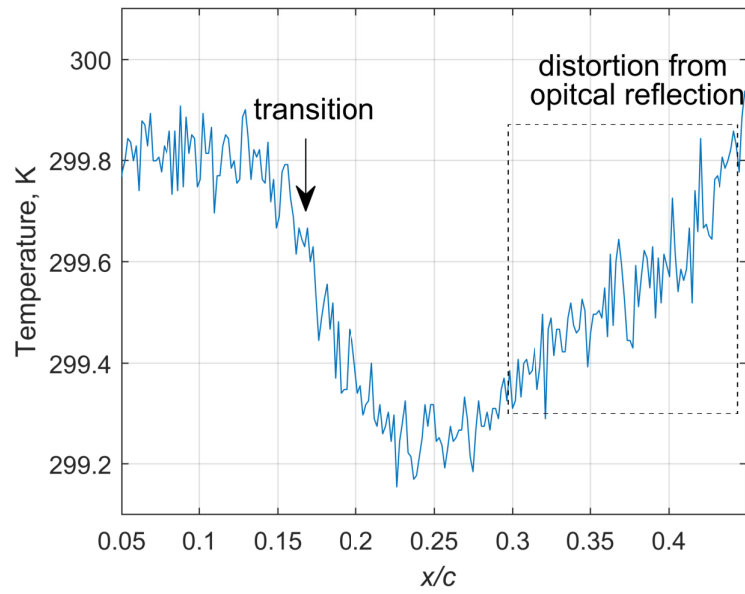


c)

Figure 5.6: Comparison of C_p distribution between analytic prediction and experimental measurements of the NACA 63-015A airfoil under steady freestream conditions, a) $\alpha = 3^\circ$, b) $\alpha = 4^\circ$, c) $\alpha = 6^\circ$.



a)



b)

Figure 5.7: Flow transition of the NACA63-015A airfoil under steady freestream measured by IR camera, $\alpha = 4^\circ$, a) instantaneous thermography, b) instantaneous surface temperature along the chord at $z/c = 0$.

Table 5.3: Predicted and measured flow transition on the upper surface of the NACA 63-015A airfoil.

α , deg	panel code $n = 9$	panel code $n = 4$	experimental C_p	IR measurements
-4	0.73 <i>c</i>	0.64 <i>c</i>	0.63 <i>c</i>	0.62 <i>c</i>
-2	0.67 <i>c</i>	0.59 <i>c</i>	0.56 <i>c</i>	0.56 <i>c</i>
0	0.61 <i>c</i>	0.54 <i>c</i>	NA	NA
1	0.58 <i>c</i>	0.51 <i>c</i>	0.49 <i>c</i>	NA
2	0.54 <i>c</i>	0.45 <i>c</i>	0.42 <i>c</i>	NA
3	0.49 <i>c</i>	0.33 <i>c</i>	0.31 <i>c</i>	NA
4	0.41 <i>c</i>	0.15 <i>c</i>	0.13 <i>c</i>	0.15 <i>c</i>
6	0.08 <i>c</i>	0.03 <i>c</i>	0.04 <i>c</i>	0.04 <i>c</i>

streamline in the near wake for both airfoils. Thus, the stagnation streamline does not bisect the trailing-edge immediately. This may be due to the strong viscous interaction when the boundary layers from the airfoil upper and lower surface mix together at the trailing-edge. This strong viscous region is not described by the classic Kutta condition of the potential flow but has been discussed among previous studies in terms of shear layer momentum, pressure gradients within the shear layer, etc [36, 37, 38]. Thus, it can be hypothesized that the stagnation condition in the potential flow perspective is altered. This effect could also have a contribution to the measured smaller lift curve slope. In addition, as the angle of attack increases, it is observed that the slope of the overall stagnation streamline increases. This agrees with the expectation as the local flow orients with the angle of attack. Further downstream in the far wake ($x/c > 2$), it is expected that the stagnation streamline will eventually follow the freestream flow direction, however, this far wake is not in the current field of view.

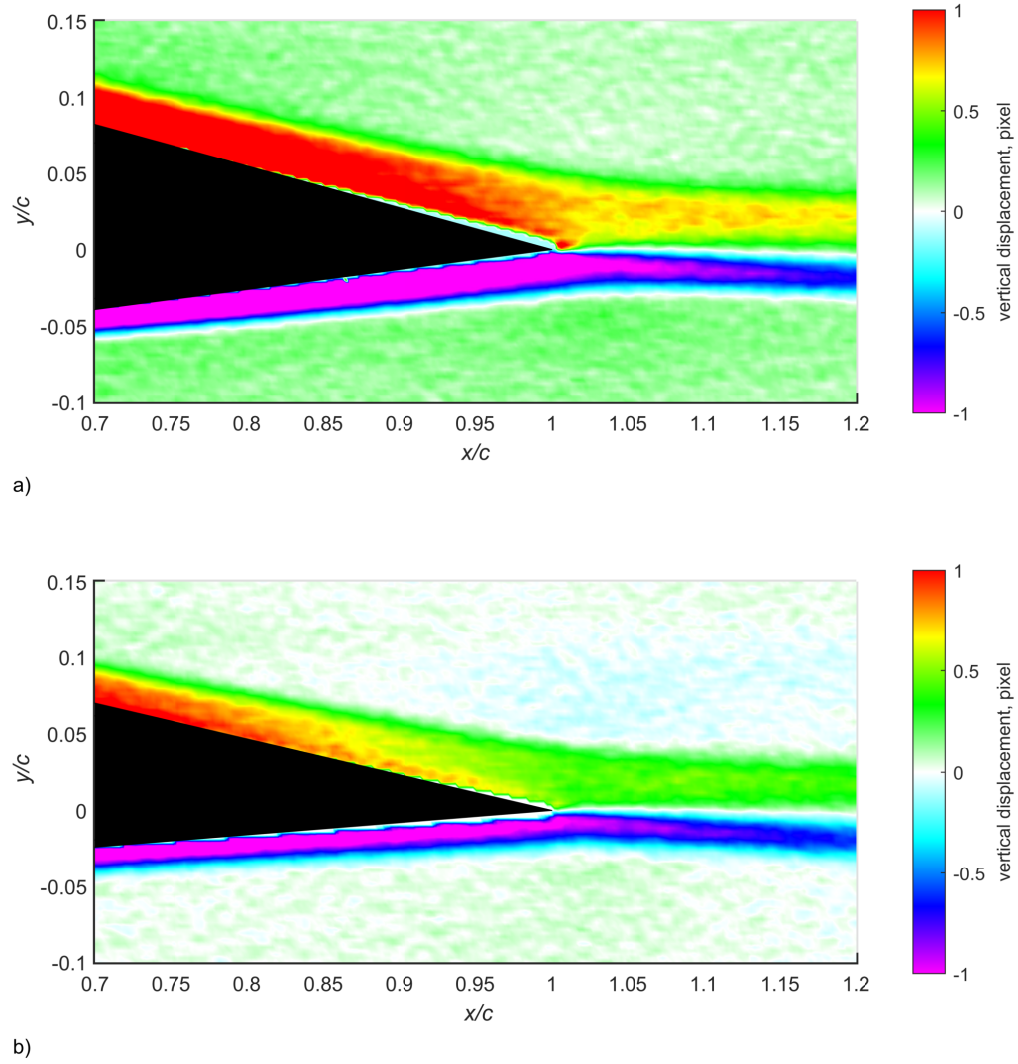
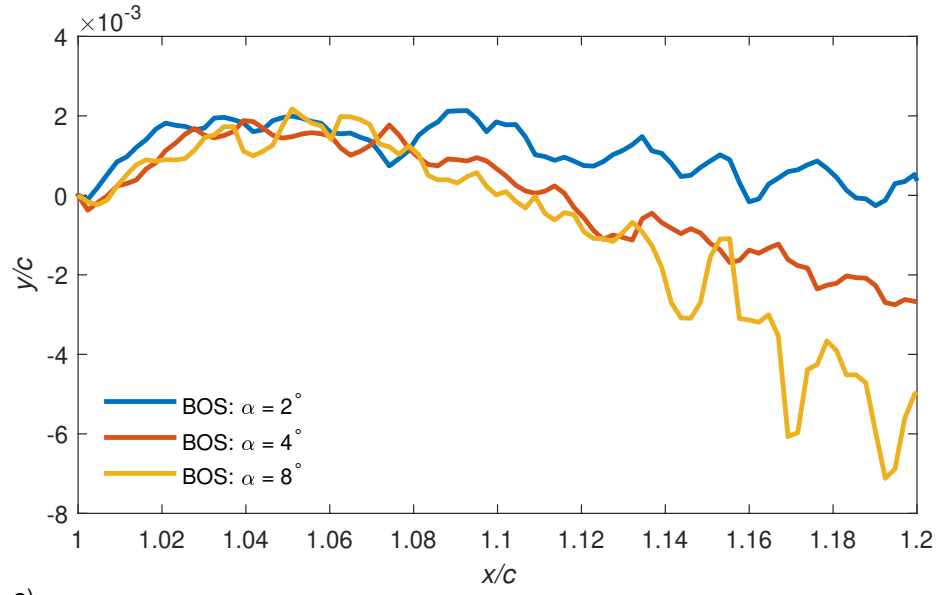


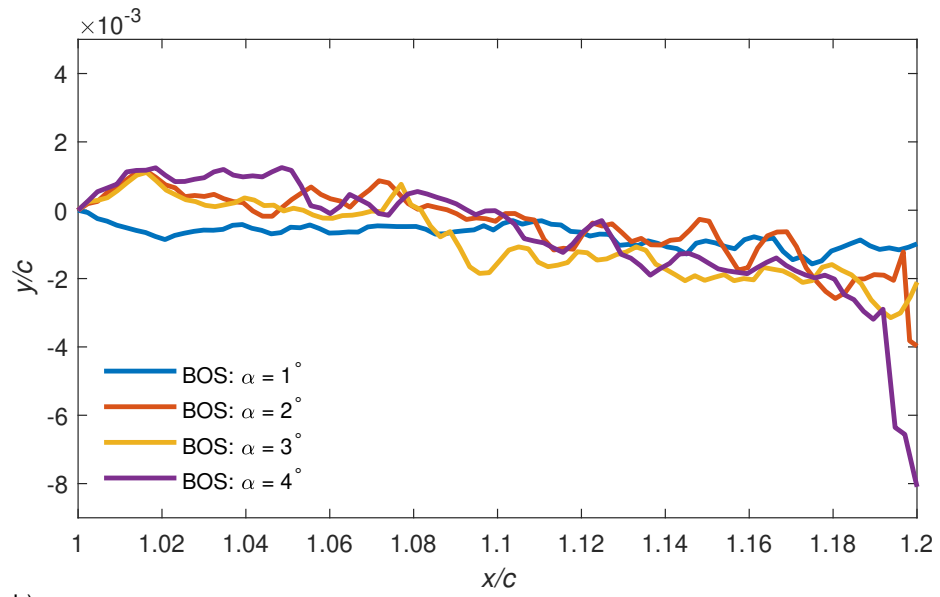
Figure 5.8: Displacement field of the airfoil under the steady freestream condition, $\alpha = 4^\circ$, a) NACA 0018, b) NACA 63-015A.

5.3 Quasi-Steady Lift Coefficient

The quasi-steady lift coefficient is used to normalize the unsteady lift coefficient by van der Wall & Leishman [3] and Strangfeld *et al* [7] to mask out the effects of



a)



b)

Figure 5.9: Steady trailing-edge stagnation streamline from BOS measurements, a) NACA 0018, b) NACA 63-015A.

dynamic pressure in unsteady lift measurements. In Isaacs' surging airfoil theory, the quasi-steady lift coefficient is expected to be identical as the steady lift coefficient.

On one hand, this is because the theory is inviscid so that Reynolds number does not come into play when freestream velocity oscillates. On another hand, the reduced frequency in a quasi-steady flow should be sufficiently low so that the shed vorticity has already convected far downstream to have a meaningful effect on the bound vorticity according to conservation of circulation. However, viscous effects in reality would come into play for the quasi-steady flow conditions. In addition, the quasi-steady reduced frequency for a surging airfoil is not well defined due to the lack of experimental studies on the flow phenomena and performance characteristics. The unsteady capability of the current wind tunnel facility has a lower limit of 1.5Hz (due to the current setup of the motor servo-drive system). This frequency is considered to be too large as a quasi-steady frequency for the current freestream condition. As a result, the strategy by Strangfeld *et al.* [7] was adopted where several steady Re_c and M_∞ combinations within the investigated surging flow condition were sampled and curve-fitted to predict the quasi-steady lift characteristics of the airfoil. Thus, the quasi-steady C_l would show the effects of both the Reynolds number and the compressibility, which the latter is less important due to incompressible Mach number in the current investigation.

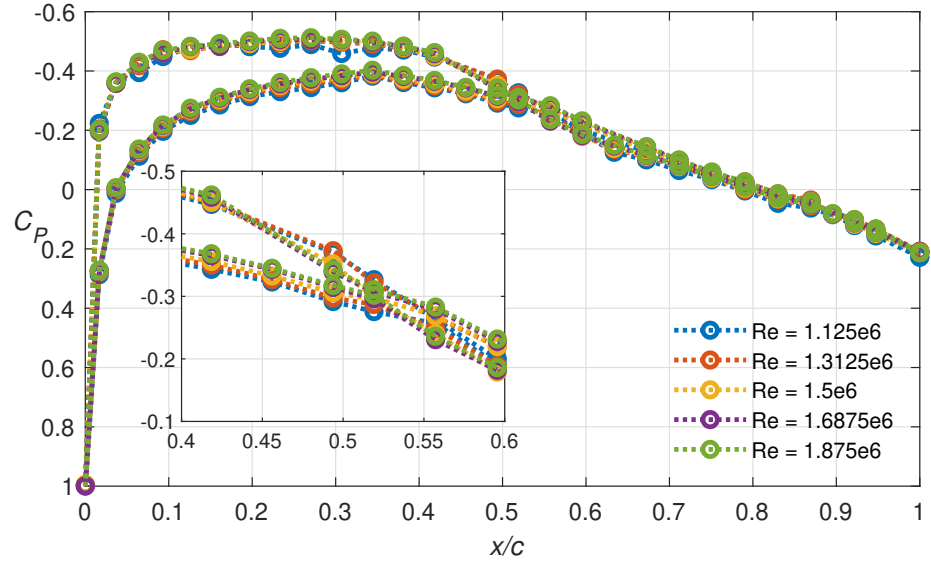
Five cases of steady Re_c are compared in Figure 5.10 where steady C_p distribution of the NACA 63-015A airfoil at $\alpha = 1^\circ$ is demonstrated. Up to 25% variation was made from mean $Re_c = 1,500,000$. Figure 5.10a shows the steady C_p distribution at five different Reynolds numbers and the results are almost identical. The major difference among these cases is the transition location on the surface of the airfoil, which occurs at approximately $x/c = 0.5$. Zoomed comparison shows slight forward movement of transition as Re_c increases. A better comparison of the effect of the

Reynolds number is given in Figure 5.10b utilizing the panel method with $n = 4$. The numerical results with 200 panels highlight the transition change due to Reynolds number, that is earlier transition at higher Re_c . Thus, the quasi-steady lift coefficient is a function of Reynolds number (or velocity amplitude), which is a result of viscous effects.

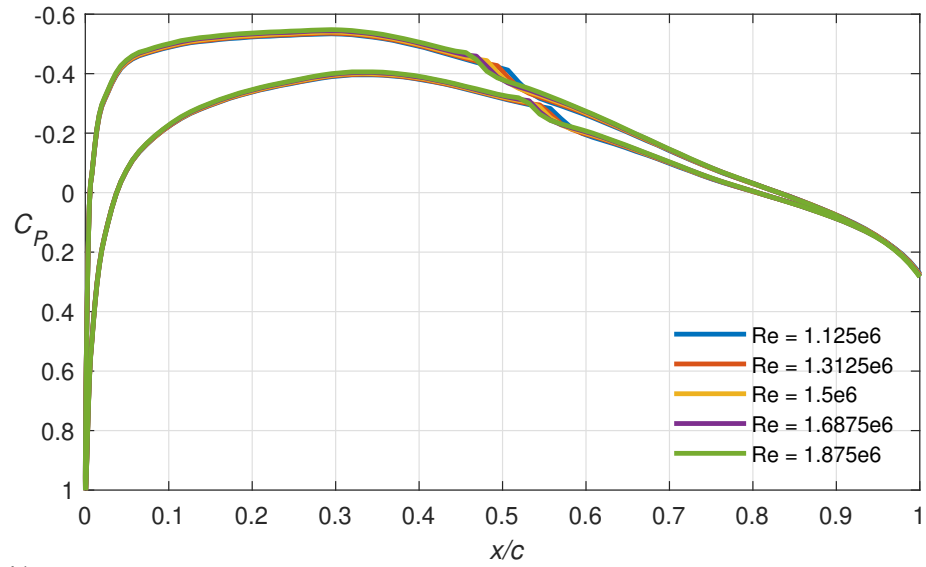
In addition, the experimental quasi-steady lift characteristics of the NACA 63-015A airfoil at different angles of attack are plotted in Figure 5.11, along with predictions from the panel code. The panel code, in general, predicts a monotonic increase in lift coefficient with Reynolds number. This is because the negative effect of the viscous contribution to the lift coefficient diminishes as Reynolds number increases. However, the experimental quasi-steady C_l does not always exhibit such behavior. There are even cases with completely the opposite trend. Similar non-monotonic behavior is also found in reference [7]. This is believed to be caused by the finite pressure tap data available, which cannot always resolve the exact transition location or the transition process as well as the suction peak and the stagnation point. Nonetheless, the small Reynolds number variation does not introduce a significant lift variation (less than 5%).

5.4 Unsteady Frequency Response

After characterizing the steady and quasi-steady lift coefficient, the normalized unsteady lift coefficient of the NACA 0018 airfoil is shown in Figure 5.12 for a mean $Re_c = 1,500,000$, mean $M_\infty \approx 0.21$, $\sigma \approx 0.21$, $k = 0.025$, and a variety of angles of attack. The prediction from Isaacs' theory at this condition is also plotted for comparison, which is not a function of the angle of attack α .



a)



b)

Figure 5.10: Quasi-steady C_p distribution of the NACA 63-015A airfoil a) untripped airfoil experimental data, $\alpha = 1^\circ$, b) prediction from the panel method, $n = 4$, $\alpha = 1^\circ$.

In Figure 5.12, Isaacs' theory predicts a maximum C_l overshoot of 1.02 at $\phi \approx 228^\circ$ when the freestream velocity is decreasing towards the minimum, and predicts a maximum C_l undershoot of 0.98 at $\phi \approx 15^\circ$ when the freestream velocity is increasing

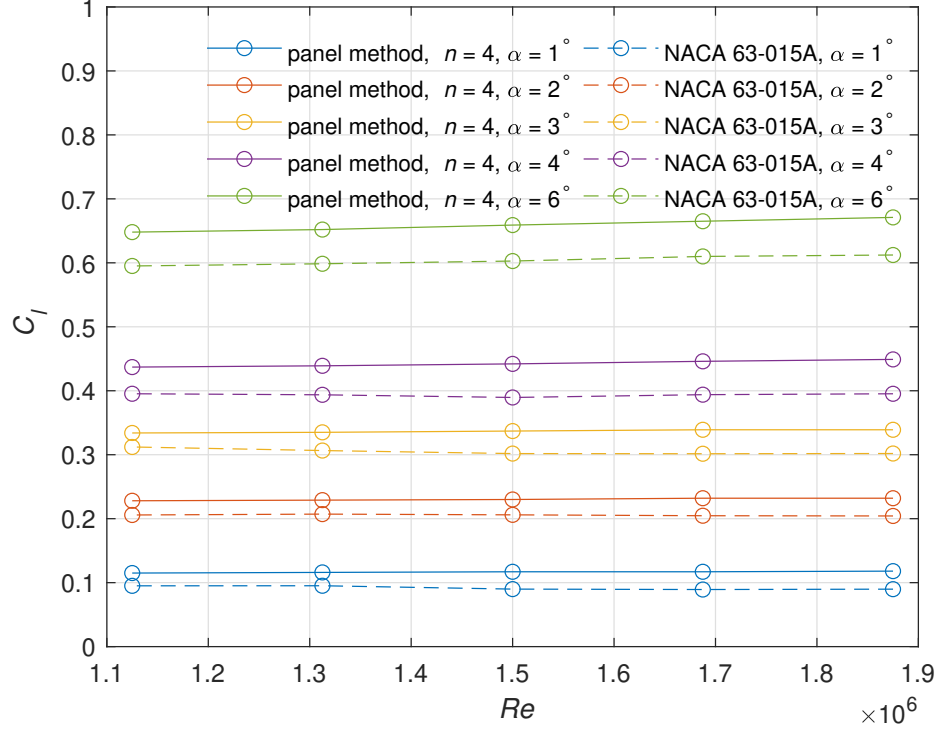


Figure 5.11: Comparison of the quasi-steady C_l of the NACA 63-015A airfoil.

towards the maximum. However, the experimental measurements indicate a maximum C_l overshoot at $\phi \approx 90^\circ$ and a maximum C_l undershoot at $\phi \approx 270^\circ$ with higher frequency oscillations. Thus, a significant phase difference between the experimental measurements and the theory for lift overshoot and undershoot is observed. In addition, both the maximum lift overshoot and undershoot are smaller from Isaacs' theory, whereas the experimental measurements are significantly larger with maximum overshoots between 1.05 and 1.1 and maximum undershoots between 0.85 and 0.95. This disagreement between experimental lift coefficient and Isaacs' theory also applies to NACA 63-015A airfoil, shown in Figure 5.13.

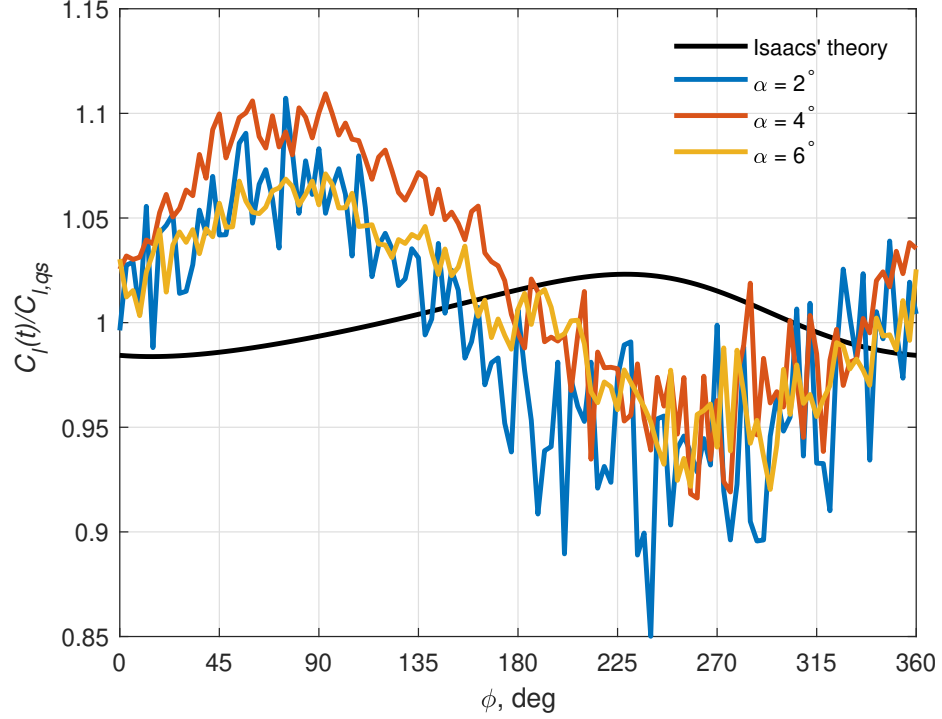


Figure 5.12: Comparison of normalized unsteady lift coefficients for NACA 0018 at mean $Re_c = 1.5 \times 10^6$, mean $M_\infty = 0.21$, $k = 0.025$, and $\sigma = 0.21$.

Although there are significant differences between Isaacs' theory and current experimental measurements, the results show some similarity to prior measurements by Strangfeld *et al.* (the smooth surface case in reference [7]), especially in terms of the phase angle. From the experimental measurements, the angle of attack was found to have a small but noticeable influence on the unsteady lift response. Figure 5.12 shows that the lift ratio is greater at $\alpha = 4^\circ$ than at $\alpha = 2^\circ$. As the angle of attack continues to increase, the maximum lift overshoot drops at $\alpha = 6^\circ$ while the maximum lift undershoot is relatively constant. This also applies to NACA 63-015A airfoil except there is also a 45° phase shift of the maximum lift overshoot as the angle of attack increases.

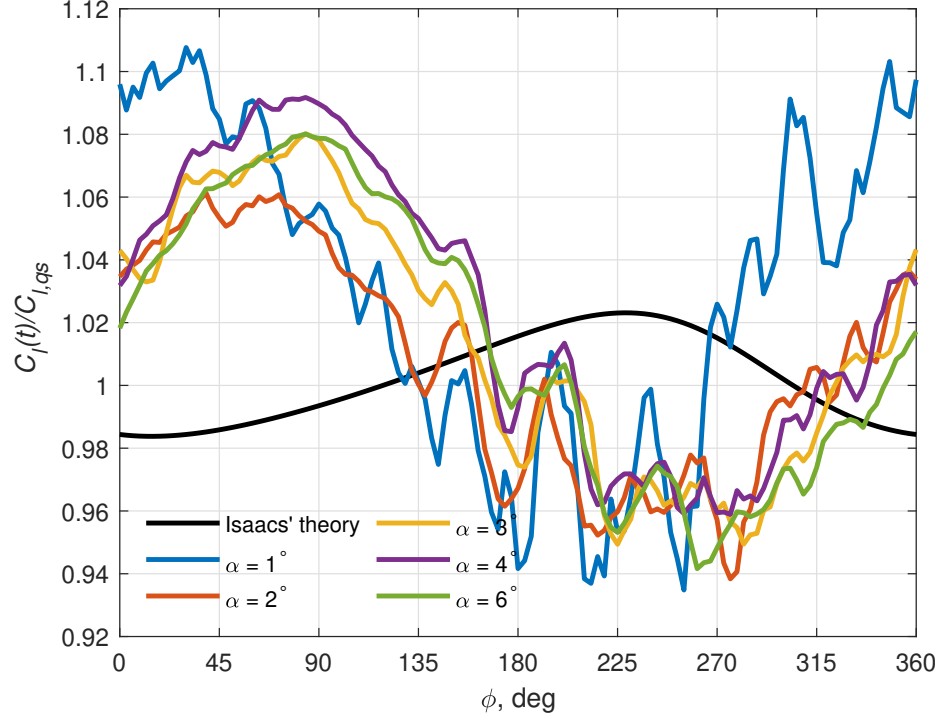


Figure 5.13: Comparison of normalized unsteady lift coefficients for NACA 63-015A at mean $Re_c = 1.5 \times 10^6$, mean $M_\infty = 0.21$, $k = 0.025$, and $\sigma = 0.21$.

In addition, the effect of reduced frequency on the unsteady lift coefficient was examined and is shown in Figure 5.14. From Figure 3.21, the sinusoidal freestream velocity in the test-section is found to experience a greater distortion at higher oscillating frequencies. When k was increased from 0.025 to 0.050, there is no appreciable change in the lift overshoot and undershoot. However, when reduced frequency k is doubled, a secondary peak at phase angle between 180° and 225° is observed. The secondary peak when $k = 0.05$ and other higher-frequency oscillations in the lift data are likely related to the acceleration change of the freestream, shown in Figures 5.15 to 5.17. In Figure 5.15a, freestream velocity experiences a significant deviation from the ideal sinusoidal wave at $\phi = 170 - 225^\circ$ coinciding with the secondary peak

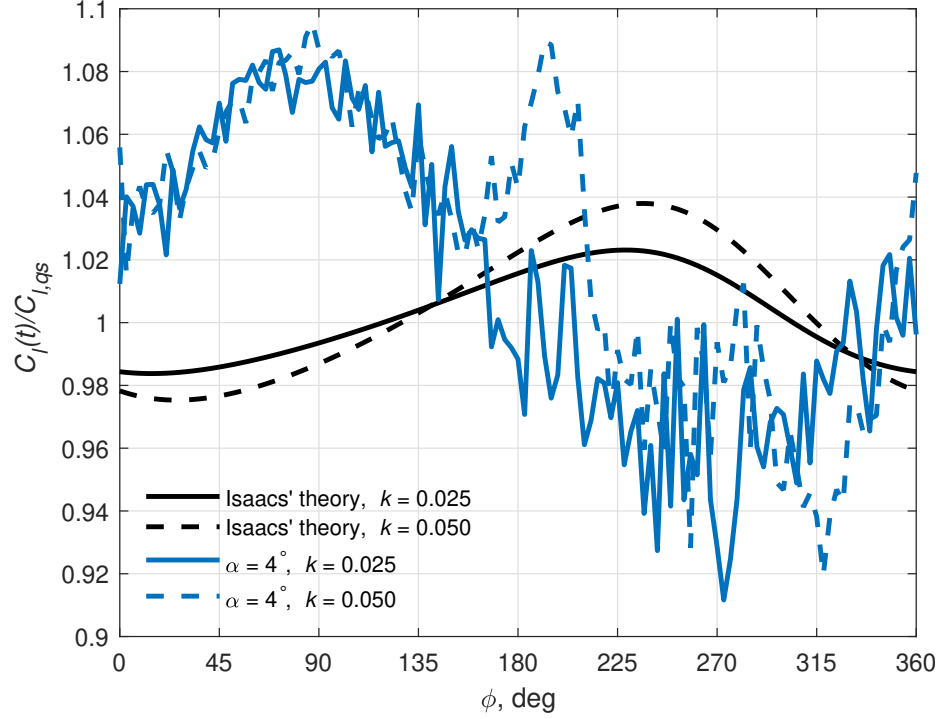


Figure 5.14: Effects of reduced frequency to normalized unsteady lift coefficients for NACA 0018 at mean $Re_c = 1.5 \times 10^6$, mean $M = 0.2$, $\sigma = 0.21$, and $\alpha = 4^\circ$.

of lift coefficient. In addition, when compared with freestream acceleration it was found that the secondary peak in C_l follows the acceleration deviation from an ideal freestream at $\phi = 135 - 190^\circ$.

Similar relations are observed for the NACA 63-015A airfoil. In Figure 5.16, the lift coefficient is decreasing when the acceleration is negative ($\phi = 90 - 270^\circ$), however, whenever the experimental acceleration oscillates with respect to the theoretical acceleration (*e.g.*, $\phi \approx 140 - 160^\circ$), there is a higher frequency oscillation in the C_l data with a slight phase lag (*e.g.*, $\phi \approx 150 - 170^\circ$). Thus, it is reasonable to conclude that the higher frequency C_l oscillation is caused by freestream kinematics. The exact reason may be related to the instantaneous shed vorticity strength and direction

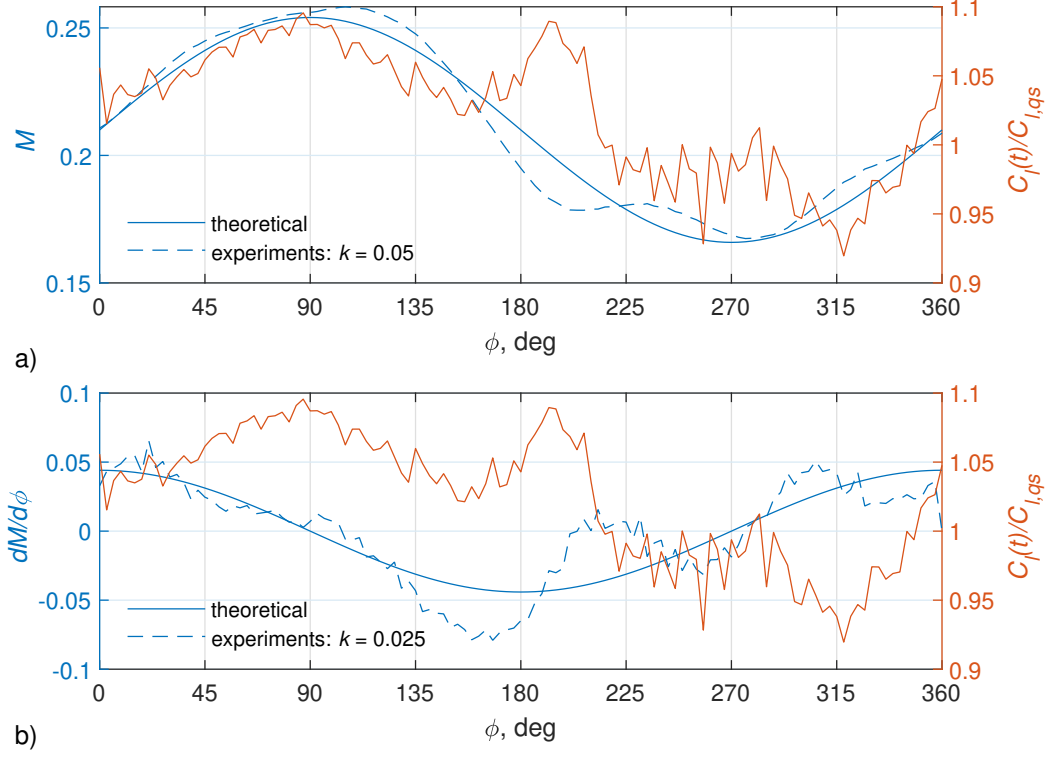


Figure 5.15: Comparison of the dynamics of the phase-averaged freestream flow and the unsteady lift frequency response, NACA 0018, $\alpha = 4^\circ$, $k = 0.05$, a) Mach number b) acceleration $dM/d\phi$.

in Isaacs' potential flow theory. That is, when the flow deceleration grows over the ideal value, a stronger clockwise vorticity is shed into the wake and increases the instantaneous lift coefficient.

Such effects of freestream acceleration are further verified by time-resolved lift data in Figure 5.17. In Figure 5.17, it is observed that the lift oscillation is closely coupled with the oscillation of freestream acceleration with some phase delays. For example, when there is an undershoot in flow deceleration at $\phi = 110 - 125^\circ$, the lift increases locally from $\phi = 115 - 130^\circ$; and when there is an overshoot in flow deceleration at $\phi = 125 - 130^\circ$ the lift decreases locally from $\phi = 130 - 140^\circ$.

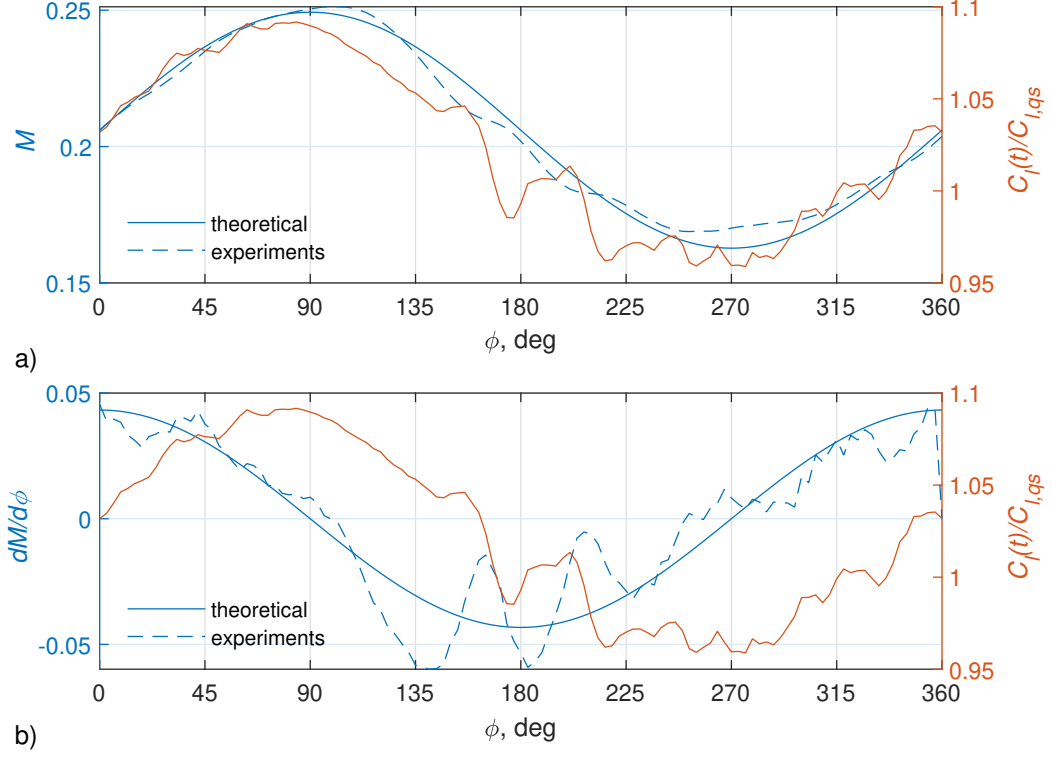


Figure 5.16: Comparison of the dynamics of the phase-averaged freestream flow and the unsteady lift frequency response, NACA 63-015A, $\alpha = 4^\circ$, $k = 0.025$, a) Mach number b) acceleration $dM/d\phi$.

Additional insight of the lift frequency response can be gained from the phase-averaged C_p distribution in Figure 5.18 and 5.19 at phase angles corresponding to the maximum lift overshoot and the maximum lift undershoot, respectively. In Figure 5.18, the suction-side pressure was reduced while the pressure-side pressure stayed relatively constant as the steady case, which led to the lift overshoot. The small change of C_p is due to the small advance ratio and reduced frequency in the current investigation. In Figure 5.19, the suction-side pressure was slightly increased, leading to the lift undershoot. From the C_p distribution, there were no apparent signs of separation bubble and shedding as was the case in reference [7]. This is reasonable

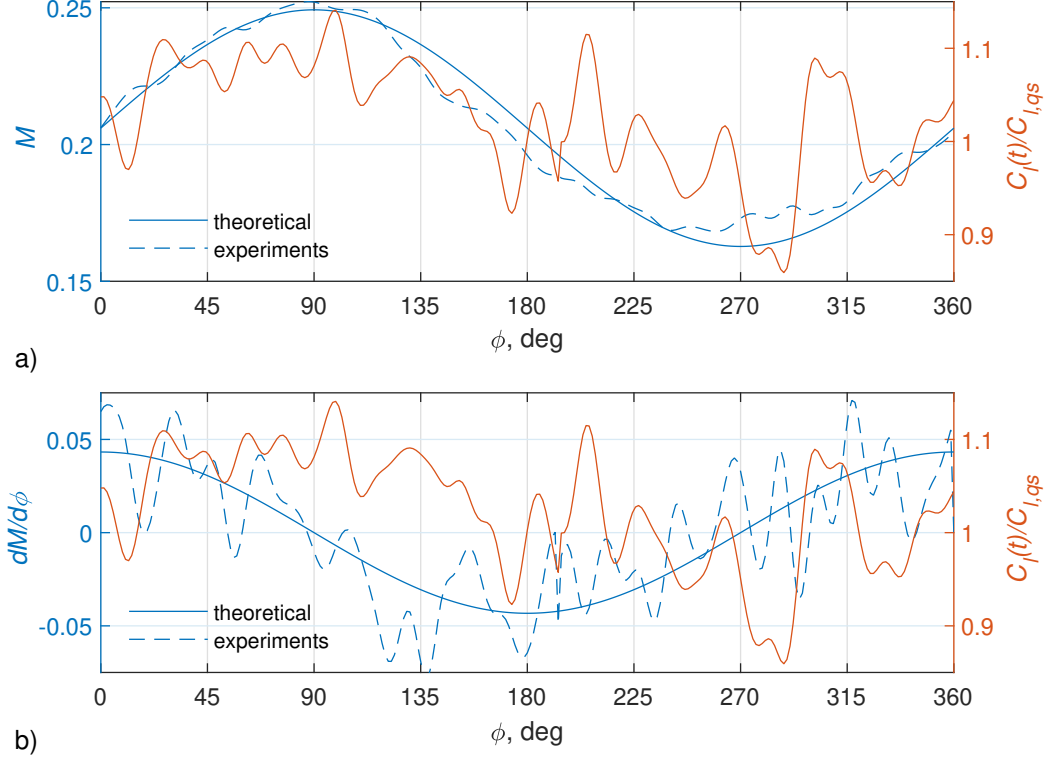
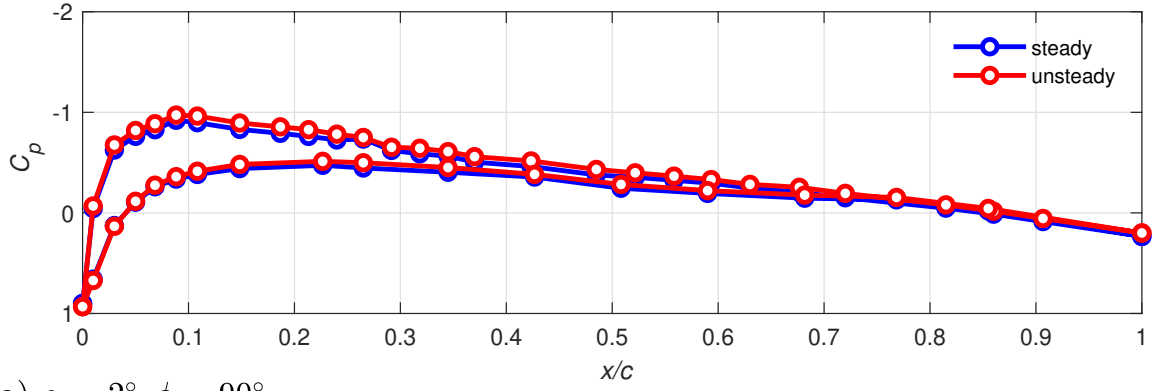


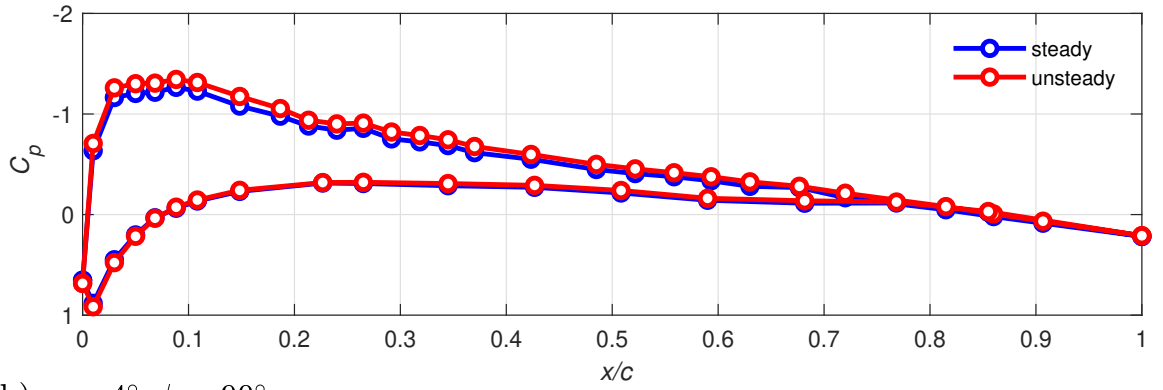
Figure 5.17: Comparison of the dynamics of the time-resolved freestream flow and the unsteady lift frequency response, NACA 63-015A, $\alpha = 4^\circ$, $k = 0.025$, a) Mach number b) acceleration $dM/d\phi$.

considering the high Reynolds number and small angles of attack in the current investigation. Thus, it is hypothesized that the deviation from Isaacs' theory was caused by a mechanism that was not considered or captured previously.

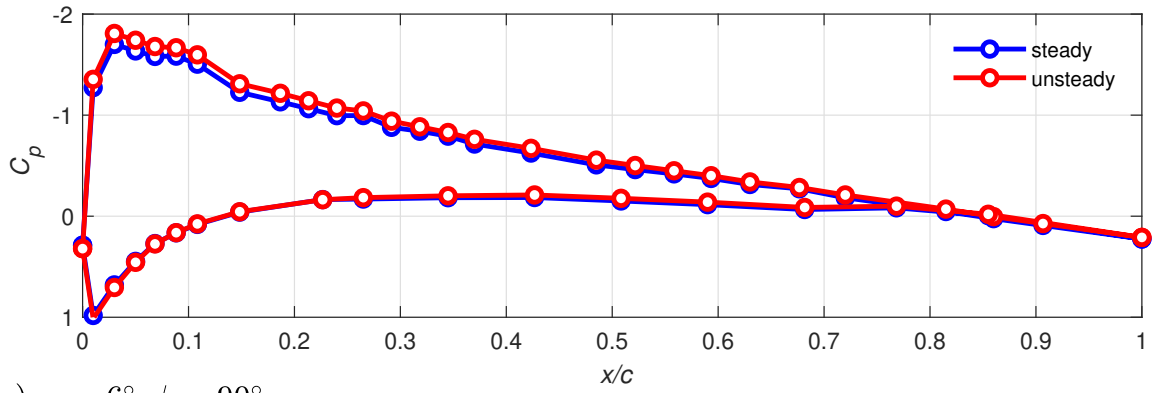
This mechanism is further investigated with the BOS measurements of the flow field. Phase-averaged displacement fields from the oscillatory freestream experiments are shown in Figures 5.20 and 5.21. A 15Hz low-pass filter was applied to the time-resolved optical measurements before phase-averaging to filter out effects of the wind tunnel vibration. Data from three angles of attack share similarities in the density gradient field so only one case is shown in Figure 5.20. Four phase angles are shown



a) $\alpha = 2^\circ, \phi = 90^\circ$



b) $\alpha = 4^\circ, \phi = 90^\circ$



c) $\alpha = 6^\circ, \phi = 90^\circ$

Figure 5.18: Phase-averaged C_p distribution of NACA 0018 at $\phi = 90^\circ$ (maximum freestream velocity).

in Figure 5.20 corresponding to the accelerating mean, the maximum, the decelerating mean, and the minimum freestream velocity, respectively. Besides the general

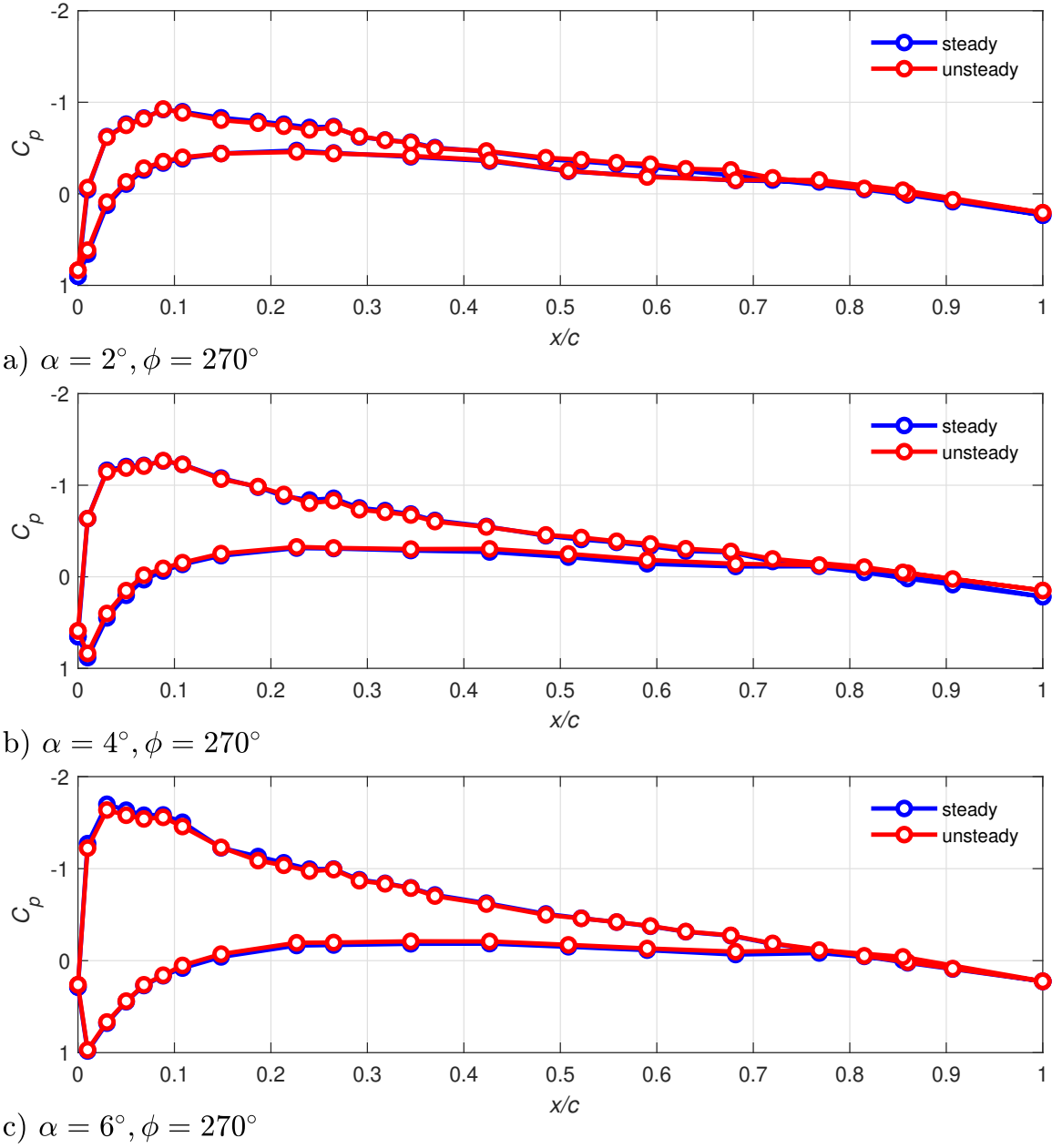


Figure 5.19: Phase-averaged C_p distribution of NACA 0018 at $\phi = 270^\circ$ (minimum freestream velocity).

patterns similar to the steady case (*i.e.*, displacement layers along the airfoil surface and the displacement wake), it is clearly observed that the wake centerline oscillates

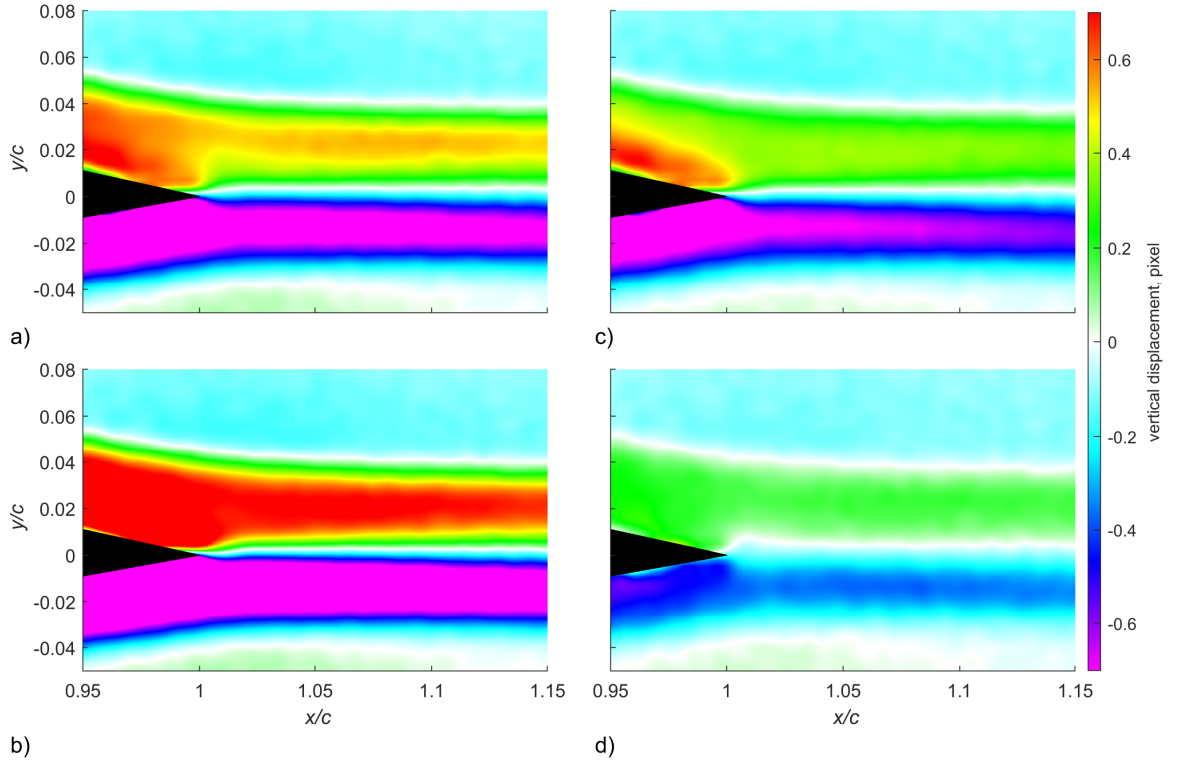


Figure 5.20: Displacement field around the surging NACA 0018 airfoil trailing-edge at selected phase angles from BOS measurements, $\alpha = 2^\circ$, a) $\phi = 0^\circ$, b) $\phi = 90^\circ$, c) $\phi = 180^\circ$, d) $\phi = 270^\circ$.

with the time-varying freestream, specifically, the curvature near the vicinity of the trailing-edge and the position of the entire zero-displacement line with respect to the airfoil. The deflection of the wake centerline is within $x/c = 1.05$, similar to the stagnation streamline observed in the steady case.

In Figures 5.20 and 5.21, when the freestream velocity increases (from $\phi = 0^\circ$ to $\phi = 90^\circ$, see Figure 3.21), the wake centerline (also refer to as the stagnation streakline) is almost a flat line emanating from the airfoil trailing edge. When velocity decreases, the stagnation streakline near the vicinity of the trailing-edge starts to

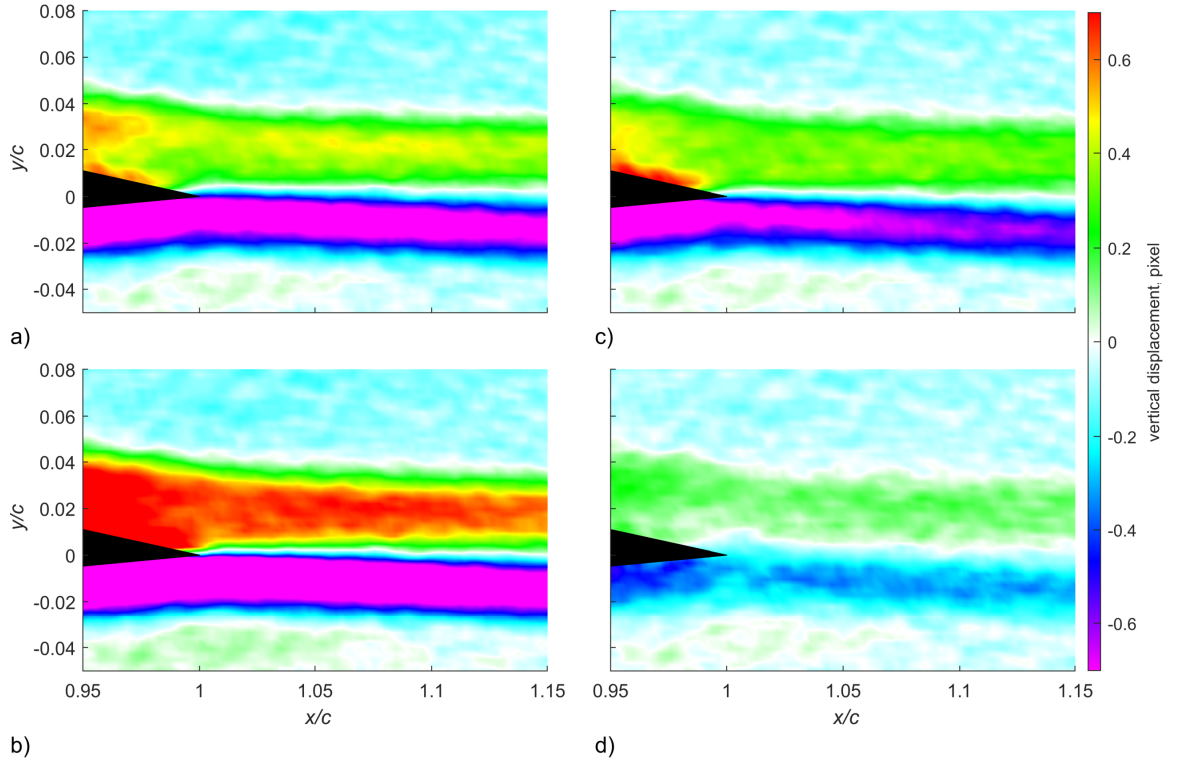


Figure 5.21: Displacement field around the surging NACA 63-015A airfoil trailing-edge at selected phase angles from BOS measurements, $\alpha = 3^\circ$, a) $\phi = 0^\circ$, b) $\phi = 90^\circ$, c) $\phi = 180^\circ$, d) $\phi = 270^\circ$.

curve up and forms a sharp bend when the freestream velocity is at the minimum ($\phi = 270^\circ$). Besides the oscillatory behavior of the position of the stagnation streakline, the amplitude of displacement in the thermal boundary layer and the wake also oscillates. Since heat transfer, viscous dissipation, and vorticity are all functions of velocity, a changing freestream velocity would affect these effects and thus the density gradient in the local flow field. For example, the flow temperature is colder when the freestream velocity is at the maximum than when it is at the minimum. Thus, the effect of

convective heat transfer is stronger when the freestream velocity is higher, leading to a greater density gradient as shown in Figure 5.20.

The movement of the trailing-edge stagnation streakline is further illustrated in Figure 5.22 for the tested angles of attack. For each phase position, the locus of points corresponding to the zero density-gradient is extracted from each phase-averaged BOS frame. The movement of the trailing-edge stagnation condition is clearly observed at these angles of attack, with the stagnation streakline near the vicinity of the trailing-edge deflected upwards when the freestream velocity decreases (from $\phi = 90^\circ$ to $\phi = 270^\circ$). At other phase positions, the stagnation streakline follows more closely the path of the steady stagnation streamline. The movement of the trailing-edge stagnation steakline, especially near the vicinity of the trailing-edge, is found to coincide with the undershoot and overshoot of the unsteady lift coefficient. To be more specific, it was found that when the stagnation streakline is deflected downward under the increasing freestream velocity the unsteady lift coefficient is growing relative to the quasi-steady value, whereas when the stagnation streakline is curved up under the decreasing freestream the unsteady lift coefficient is declining relative to the quasi-steady value.

Such relationship between the trailing-edge stagnation condition and the lift coefficient is comparable to the relationship of bound circulation and lift in the potential flow theory. For a steady airfoil in potential flow, the application of the Kutta condition fixes the stagnation point at the airfoil trailing-edge and a unique lift is produced. However, without the Kutta condition, the lift can be greater when the trailing-edge stagnation point is moved towards the lower surface or the lift can be lower when the trailing-edge stagnation point is moved towards the upper surface. The inclined

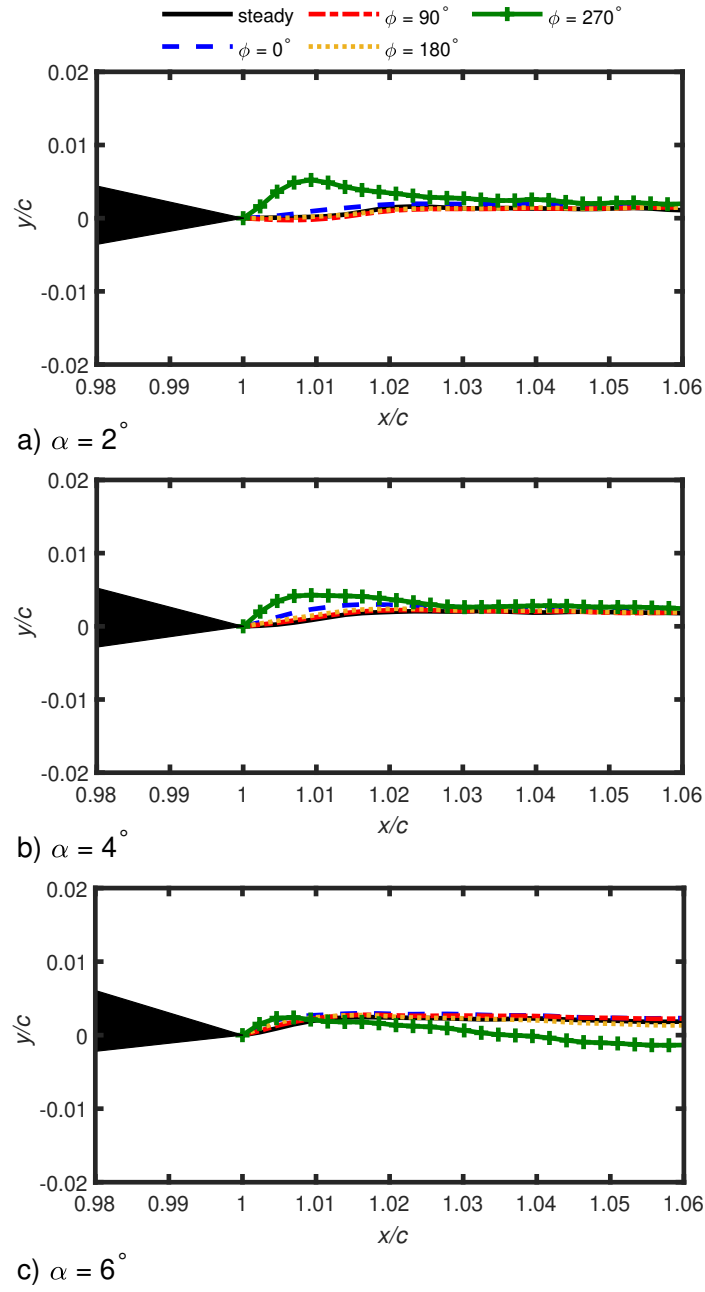


Figure 5.22: Trailing-edge stagnation streakline of the surging NACA 0018 airfoil at selected phase, a) $\alpha = 2^\circ$, b) $\alpha = 4^\circ$, c) $\alpha = 6^\circ$.

and oscillatory trailing-edge stagnation condition observed in the current study is not the first time similar phenomenon has been reported. Poling and Tellionis [36] found that the stagnation streamline of a fixed-pitch airfoil immersed in the wake of a pitching airfoil deflects in a region that is totally immersed in the viscous layers immediately downstream of the trailing-edge. Thus, it is found that the trailing-edge stagnation condition is not steady as the Kutta condition implies and can be altered in the current surging airfoil investigation, which led to the measured lift coefficient disagreeing with Isaacs' surging airfoil theory.

This violation of the classic Kutta condition in Isaacs' surging airfoil theory is deduced as a result of unsteady viscous effects at the trailing-edge [36, 37, 38]. Xia & Mohseni [37] developed a free vortex model for airfoils under pitching and plunging motion. They simulated the unsteady viscous effects at the trailing-edge from conservation of circulation and momentum and found that the momentum balance between the upper and lower shear layers at the trailing-edge dictates the stagnation condition, specifically the angle of the stagnation streamline. Based on this idea, it is hypothesized that there is also a momentum balance between the shear layers at the trailing-edge for surging airfoil. Based on the BOS measurements, it can be presumed that when freestream velocity increases the upper shear layer momentum becomes stronger than the lower shear layer momentum. Thus, the trailing-edge stagnation condition, specifically, the curvature of the stagnation streakline is deflecting downwards and a higher lift coefficient is produced. Similarly, when freestream velocity decreases, the upper shear layer momentum becomes weaker than the lower shear layer momentum (or weaker than the steady reference) and the stagnation streakline

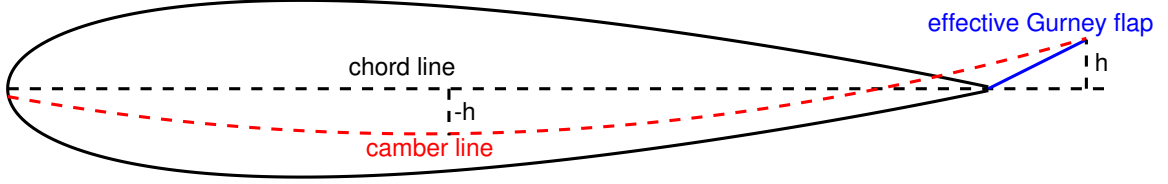


Figure 5.23: Schematic of the effective camber effect due to oscillation of the trailing-edge stagnation point.

is curved upwards for a reduced lift coefficient. Thus, there is a coupling between the oscillatory trailing-edge stagnation condition and the measured lift coefficient.

A first approximation is attempted to compensate the effect of this oscillatory trailing-edge stagnation condition. That is, the oscillatory trailing-edge stagnation streakline can be approximated as a dynamic Gurney flap [70] that extends upwards into the flow field and modifies the stagnation point. Such Gurney flap creates an effective (negative) camber to the airfoil, illustrated in Figure 5.23. In Figure 5.23, the effective camber of the airfoil is assumed to be proportional to the negative height of the effective Gurney flap, which is the deflected stagnation condition near the vicinity of the trailing-edge.

The dynamic of this effect under the oscillating freestream is plotted in Figure 5.24 based on BOS measurements. The effective Gurney flap height h is normalized by the effective height under steady freestream because similar but fixed deflection of the trailing-edge stagnation condition is also observed for the steady airfoil (see Figure 5.9). In general, results show that the effective steady airfoil has a negative camber; and the negative camber is reduced when the freestream velocity is above the mean while the negative camber is increased when the freestream velocity is below the mean.

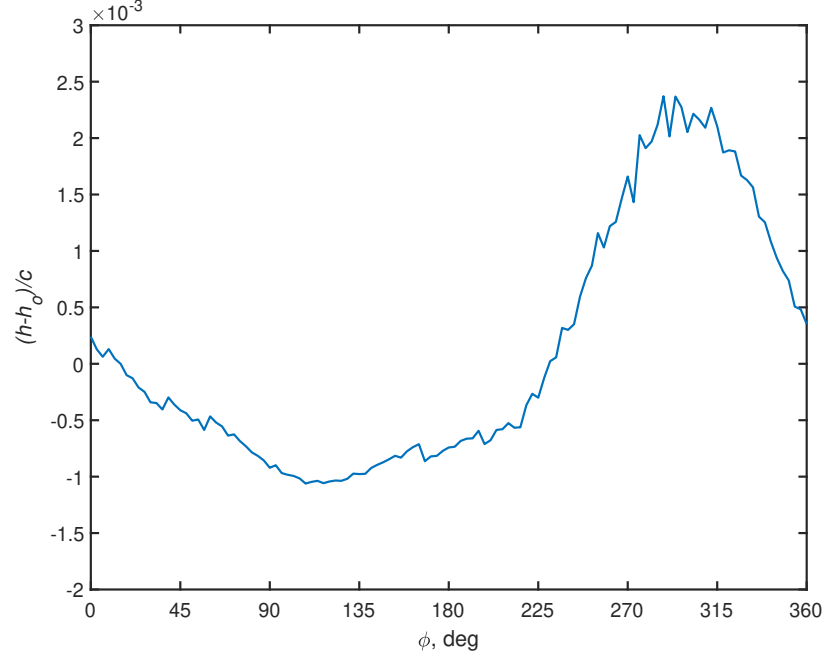


Figure 5.24: Height of the effective Gurney flap of the NACA 0018 airfoil based on trailing-edge stagnation condition measured by the BOS technique, $\alpha = 4^\circ$, $k = 0.025$.

This first approximation is compared to the measured normalized unsteady lift coefficient by combining the dynamic camber effect with thin airfoil theory. The lift coefficient of a cambered airfoil is given by thin airfoil theory as [8, 68]

$$C_l = 2\pi \sin\left(\alpha + \frac{2h}{c}\right) \quad (5.1)$$

Thus, the normalized unsteady lift coefficient due to dynamic camber effect can be approximated as

$$\frac{C_l(t)}{C_{l,qs}} = \frac{2\pi \sin\left(\alpha + \frac{-2h(t)}{c}\right)}{2\pi \sin\left(\alpha + \frac{-2h_0}{c}\right)} \quad (5.2)$$

where $-h(t)/c$ is the camber of the airfoil under the surging condition, $-h_0/c$ is the camber of the airfoil under steady freestream.

The effect of camber on the lift frequency response is shown in Figure 5.25. It is observed that the unsteady lift coefficient due to the oscillatory camber effect (an approximation of oscillatory trailing-edge stagnation condition) has a closer agreement with the experimental lift than Isaacs' theory, especially in terms of phase. This may indicate the unsteady trailing-edge stagnation condition has a more significant impact on the lift coefficient than Isaacs' theory of shed vorticity under the current flow condition. However, this approximation alone cannot perfectly explain the measured lift frequency response since there is still deficiency in the lift magnitudes. Thus, additional effects may be contributing to the deviation, such as Isaacs' circulation theory. It is also speculated that any combined effect, such as Isaacs' circulation theory and the unsteady trailing-edge stagnation condition, may not be linear.

5.5 Discussion

In the current experimental investigation of the surging NACA 0018 and NACA 63-015A airfoil, the time-history of the unsteady C_l ratio is found to be in disagreement with Isaacs' surging airfoil theory. To better understand the cause of this disagreement, it is necessary to revisit the various assumptions employed in the development of the classic unsteady airfoil theory, and evaluate the likely weak points that should be examined in closer detail.

5.5.1 Thin Airfoil

Inherent to Isaacs' theory is the assumption of thin airfoil theory. This effectively models the airfoil as a flat plate, such that thickness effects are neglected. This directly implies that the airfoil section under study should not have an impact on the ideal unsteady loading. The two airfoils in the current investigation have different

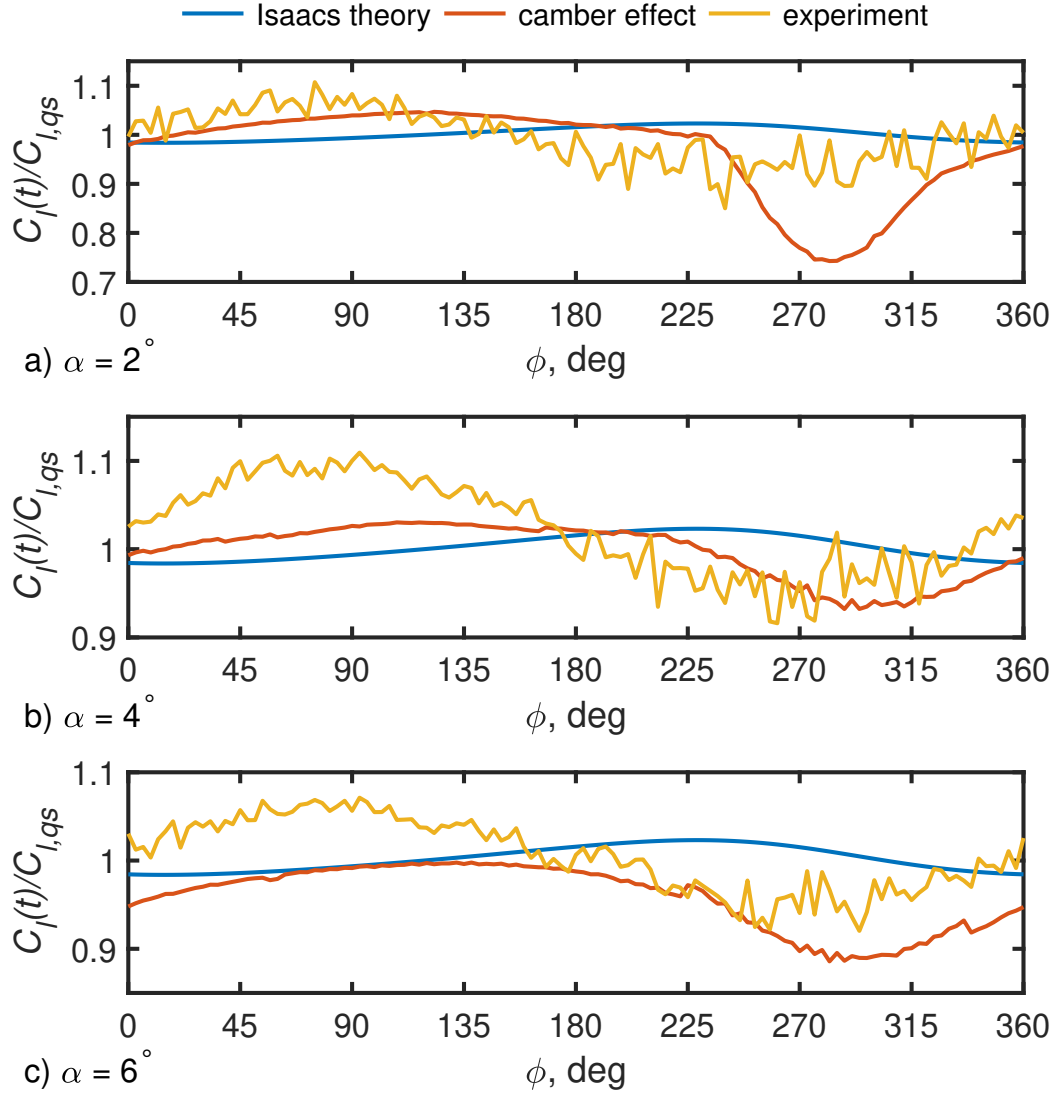


Figure 5.25: Comparison of effective camber effect and Isaacs' theory, a) $\alpha = 2^\circ$, b) $\alpha = 4^\circ$, c) $\alpha = 6^\circ$.

thickness ratio and profile. However, they both exhibited similar behavior that does not agree with the phase or amplitude predicted by Isaacs. The NACA 63-015A may be considered 'thin', while the NACA 0018 may be considered 'thick'. Thus, it is

concluded that airfoil thickness does not have a significant impact on the unsteady loading, to the extent that the flow remains attached.

5.5.2 Low Angles of Attack

Thin airfoil theory also presumes that the airfoil is at a low enough fixed angle of attack that the flow remains attached and well away from stall. This work studied a range of angles of attack, with all of the cases showing similar behavior. It is anticipated that higher angles of attack will start to depart from the collapsed curves seen in Figure 5.12, with changes in the trailing-edge separation characteristics or formation of a laminar separation bubble near the leading-edge.

5.5.3 Incompressible Flow

Isaacs' theory assumes that the flow is incompressible. The mean Mach number in this work (≈ 0.2), as well as the maximum Mach number (≈ 0.25), are well within the incompressible regime despite the velocity magnitude in the current work (≈ 68 m/s) being an order of magnitude higher than what was used in other experimental investigations [7, 15, 22]. Thus, this assumption is well satisfied. However, the impact of compressibility on the unsteady lift response for a fixed-pitch airfoil is currently unknown.

5.5.4 Zero Streamwise Pressure Gradient

The flow situation modeled by Isaacs' theory presumes an airfoil moving at an unsteady velocity through a quiescent medium. However, in most unsteady wind tunnels in use today, the unsteady freestream velocity is produced by varying a downstream

blockage area. Information on the time-varying back pressure is communicated upstream at the speed of sound, leading to longitudinal pressure gradients. However, the effects of the streamwise pressure gradient are currently unknown. A time-varying adverse pressure gradient could lead to thickening of the boundary layer during the decelerating portion of the waveform, making it more susceptible to separation or other unsteady viscous phenomena. At present it is difficult to assess the significance of this effect. The best approach to assessing the impact of streamwise pressure gradients will be found through a careful comparison with experimental data in a facility with no pressure gradient. For example, water towing tank [16] is free from streamwise pressure gradients, but have other challenges related to control of the experimental conditions.

5.5.5 Planar Wake

Isaacs' theory presumes that the shed vorticity convects downstream along the freestream direction, remaining in the same plane. The theory does not account for transverse movement of vorticity due to mutual induction of vorticity or other effects. Some degree of non-planar wake behavior is not surprising; in fact the data in Figure 5.22 clearly show that the instantaneous wake moves away from the steady wake. As a result, The stretching, shortening, and induced rotation between wake vortices are likely to have a impact on the lift coefficient. For example, if a counterclockwise vorticity is shed but travels below the plane of wake in Issacs' theory due to another counterclockwise vorticity downstream, its downwash effect on the airfoil could be diminished because of the increased radius between the airfoil and the vorticity. Thus, the lift undershoot is reduced comparing to the planar wake. Similarly, the upwash

effect of a clockwise shed vorticity in Isaacs' theory could be reduced. A free vortex model [37] may give more insight of the wake vortices analytically.

5.5.6 Shed Vorticity Convects at Local Freestream Velocity

In addition, Isaacs presumes that the shed vorticity convects at a speed equal to the local freestream velocity. The convective speed of the vortical structures is important since it dictates the contraction or dilation of the shed vorticity, vortical roll-up, and the line of action of the vorticity. During the high-velocity portions of the waveform, the vorticity is spatially dilated and swept downstream faster. This results in a reduction of the effects of the induced velocity - the accelerating freestream sheds vorticity that produces downwash, but this vorticity is more rapidly convected downstream so the magnitude of the C_l reduction is diminished. In contrast, the low-velocity portion of the waveform will contract the vorticity (slower convective speed), such that the shed vorticity that induces upwash has shorter radius to the airfoil and enhances the C_l increment. This behavior, as described by Isaacs, presumes an inviscid wake with vorticity that convects at the same velocity as the freestream. However, this assumption ignores the effects of the velocity deficit in the wake due to viscosity on the airfoil. During the high-velocity portion of the waveform there is increased dimensional drag (relative to the mean) and a larger relative velocity deficit. This serves to reduce the amount by which the vorticity is dilated during the accelerating portion of the waveform. Similarly, during the low-velocity portion of the waveform the dimensional drag is reduced relative to the mean and results in a lower relative velocity deficit. This trend leads to a relative increase in the contradiction of the shed vorticity during the decelerating portion of the waveform. Thus, the wake

velocity deficit (unaccounted for by Isaacs) is anticipated to lead to phase lead or lag, and distortion of the C_l time history relative to the ideal case. These effects are anticipated to be small, but the significance of this contribution is not yet known.

5.5.7 Inviscid Potential Flow

Shed vorticity is a well-accepted phenomenon that is known to have a first-order impact on the unsteady lift [3, 35, 37]. Any vorticity present in the fluid surrounding the airfoil will have a substantial impact on lift, and it is known that changes in circulation will result in shed vorticity. This inviscid potential flow effect, as captured by Isaacs, has been validated by comparison with an unsteady, inviscid Euler code simulation [3]. Furthermore, Strangfeld *et al.* [7] argued that a high- Re validation of Isaacs' theory should show better agreement; however, the results presented here have not validated Isaacs' theory as expected. Thus, it can be inferred that any differences between the experimental data and Isaacs' inviscid potential flow prediction must be due to viscous effects. The viscous effects could include the nonplanar wake and the wake velocity deficit mentioned above, and also the local boundary layer [36, 37, 38]. The exact nature of these viscous effects must be established to fully understand the measured unsteady lift coefficients.

5.5.8 Kutta Condition

Inherent to the thin airfoil theory is the assumption that the Kutta condition is satisfied. The Kutta condition is essentially an empirically-based closure mechanism that narrows down an infinite number of potential flow solutions to a single solution that predicts steady airfoil lift extremely well. This presumed condition conveniently works so well due to the characteristic sharp trailing-edge of most airfoils. The Kutta

condition can be further expressed as a stipulation of zero transverse pressure gradient between the upper and lower surfaces at the trailing-edge, or alternatively as a stipulation that the velocity at the trailing-edge must be finite or zero [36]. The key implication of the classic Kutta condition is that the rear stagnation point is fixed at the trailing-edge of the airfoil, and the stagnation streamline must bisect the finite angle of the trailing-edge.

However, there has been substantial controversy over the question of whether the classic Kutta condition is valid for an unsteady airfoil [34, 35, 36, 37, 38]. The observed trailing-edge flow oscillation in the current study shows clear violation of the classic Kutta condition for surging airfoil. It also opens a question about precisely what physical mechanisms actually lead to the movement of the stagnation point. The central idea is focused on the momentum balance of the shear layer at the trailing-edge [37], an unsteady viscous effect that determines the stagnation condition and thus lift coefficient. According to BOS measurements, it can be hypothesized that the momentum balance favors the upper shear layer when the freestream velocity is increasing and a greater lift coefficient is produced because the stagnation condition (*i.e.*, the stagnation point and the stagnation streakline) is moving towards the lower side; whereas the momentum balance favors the lower shear layer when the freestream velocity is decreasing and a smaller lift coefficient is produced with stagnation condition moving towards the upper side. Alternatively, it can be viewed as a total pressure imbalance at the trailing-edge. For instance, if the suction-side boundary layer has a higher momentum flux and matched static pressure of the pressure-side boundary layer at the accelerating phase, it will have a higher total pressure and will ‘overpower’ the pressure-side boundary layer and the trailing-edge stagnation

streakline is curved down. This leads to an increased circulation and lift overshoot. The opposite would happen during the decelerating phase. The decelerating phase is particularly prone to separation, which would limit the pressure recovery on the suction-side. That would make for a lower total pressure on the suction-side at the trailing-edge compared to the pressure-side. This would lead to stagnation condition curving towards the suction-side and create negative circulation and lift undershoot.

Finally, the work of Strangfeld *et al* [7] is revisited. In their work, the airfoil under study had one surface of the symmetric airfoil tripped and the opposing surface was untripped. When their ‘smooth’ surface was positioned on the suction side, their lift time-history results resembled the findings presented in the current work. However, when their tripped surface was positioned on the suction side, their lift results agreed much more closely with Isaacs’ theory. This can be explained by the presence of significantly different boundary layer characteristics of the upper and lower surfaces when the viscous layers meet at the trailing-edge. With a tripped suction-side boundary layer, the upper boundary layer may have a higher momentum at the trailing-edge and be more resistant to movement of the stagnation streamline. However, with a clean boundary layer on the suction side, the upper surface boundary layer has less momentum and will be more susceptible to the stagnation point movement.

5.6 Summary

Airfoils at fixed pitch angles under a harmonically varying freestream was experimentally investigated in order to perform a high- Re assessment of Isaacs’ surging airfoil theory. One of the primary airfoils studied in this work was a NACA 0018 model, to facilitate comparison with a similar investigation at lower Reynolds number

[7]. Airfoil surface pressures were measured to compare the unsteady lift coefficients with Isaacs' predictions. It was found that the measured unsteady lift coefficient disagrees with Isaacs' theory in terms of the magnitude and phase of the maximum lift overshoot and undershoot. For a given advance ratio and reduced frequency, experimental C_l always show significant deviations from the steady C_l at a different phase from the theory - in many cases, the measured lift data was 90 to 180 degrees out of phase with Isaacs' predictions. In fact, none of the measured high- Re data came close to matching the theoretical predictions.

Because of this discrepancy, the background-oriented schlieren method was applied to investigate the flow field near the airfoil trailing-edge. Results suggest that the trailing-edge stagnation condition of the surging airfoil does not satisfy the classic Kutta condition. Instead, viscous effects are likely to dominate the trailing-edge stagnation condition where it is hypothesized that the shear layer momentum balance at the trailing-edge leads to the oscillation of the stagnation point under the oscillatory freestream. The oscillatory trailing-edge stagnation condition is also approximated as a dynamic Gurney flap where the lift coefficient can be estimated by thin airfoil theory with camber effect.

Chapter 6: Experimental Investigation of the Unsteady Viscous Kutta Condition for a Surging Airfoil

In Chapter 5, it has been found that the normalized unsteady lift coefficients of the surging NACA 0018 and NACA 63-015A airfoil at mean $Re_c = 1,500,000$, mean $M_\infty \approx 0.21$, $\sigma \approx 0.21$, and $k = 0.025$ disagree with the prediction from Isaacs' surging airfoil theory. The drastically different lift overshoot and undershoot magnitudes and corresponding phases from the prediction are likely to be caused by flow physics not captured in the potential flow analysis. After careful examination of the surface pressure distribution and with the help of the flow field measurements, it is found that the trailing-edge stagnation condition of the surging airfoil exhibits strong unsteady characteristics. This unsteady trailing-edge stagnation condition violates the classic Kutta condition assumed in the classic unsteady airfoil theory and could be the cause of the different lift coefficient from prediction. A hypothesis is formulated that the momentum of the upper and lower shear layers at the trailing-edge dictates the unsteady stagnation condition. However, this hypothesis is only based on observations of the displacement field of the airfoil, an indirect measurement of the shear layer momentum itself. To further prove the existence of a shear layer momentum balance

at the trailing-edge and to understand its dynamic characteristics, a traversing near-wake survey experiment was designed to measure the velocity near the vicinity of the airfoil trailing-edge.

6.1 Modification of Shear Layer Momentum

The design of the traversing wake rake has been given in Chapter 2. In addition, if the trailing-edge stagnation is determined by the momentum balance of the two shear layers, a modification of the local shear layer momentum should have an impact on the momentum balance. In order to explore this effect, forced transition on the NACA 63-015A airfoil with two-dimensional boundary layer trip were investigated.

According to White [60], the introduction of a roughness with a wire height k greater than the local displacement thickness δ^* at the wall will lead to earlier transition because of the additional disturbances fed into the boundary layer. An empirical relation to determine optimal trip height for incompressible flow over a flat plate was given by Gibbings [71]

$$Re_k = \frac{Uk}{\nu} \approx 826 \quad (6.1)$$

where the Reynolds number based on effective trip wire diameter is written in terms of the local flow and boundary layer parameters (U and ν). Tani *et al.* [72] extended the flat plate correlation to include the influence of the local pressure gradient via the dimensionless pressure gradient parameter λ^* and displacement thickness δ_k^* .

$$\frac{U(x_k)k}{\nu} \approx 826e^{-0.9\lambda^*} \quad (6.2)$$

$$\lambda^* = \frac{(\delta_k^*)^2}{\nu} \frac{dU}{dx} \quad (6.3)$$

This equation was applied to predict an effective trip wire diameter k on the NACA 63-015A airfoil needed for the current experiment. The boundary layer properties

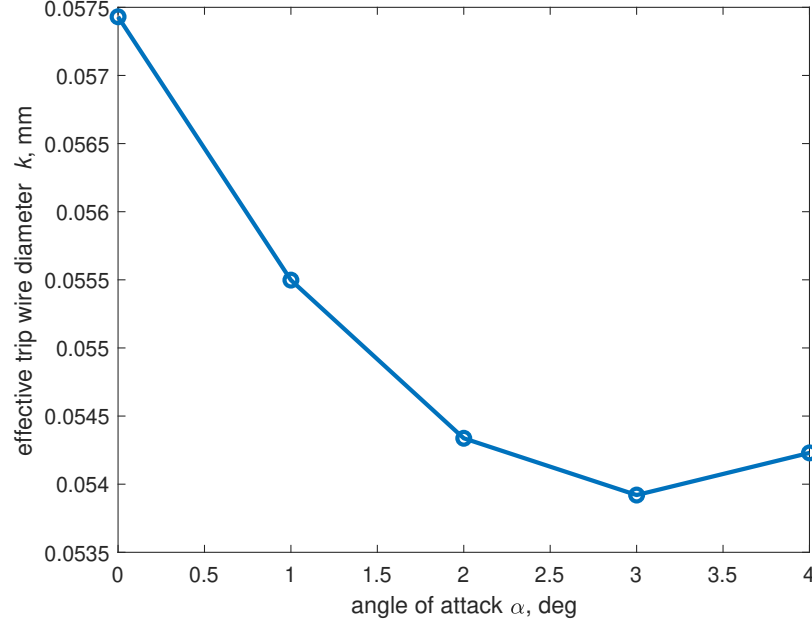


Figure 6.1: Effective trip wire diameter as a function of the angle of attack, NACA 63-015A airfoil, $x_k/c = 0.05$, $Re_c = 1.5 \times 10^6$, $M_\infty = 0.2$.

were predicted by a two-dimensional panel code [33] which solves the variables δ_k^* and dU/dx of the target airfoil shape and freestream condition. The resulting effective trip dimensions on the suction surface at $x/c = 0.05$ are plotted against the angle of attack in Figure 6.1. Higher angles of attack are not shown because natural transition would have already occurred near the leading-edge. As a result, a graphic tape with width of 1.6mm and thickness of 0.06mm was selected as the boundary layer trip. The tape was applied across the airfoil span and the trailing-edge of the tripping tape was positioned at $x/c = 0.05$. Two forced transition cases are investigated besides the clean NACA 63-015A airfoil, one only tripping the suction surface, and another only tripping the pressure surface.

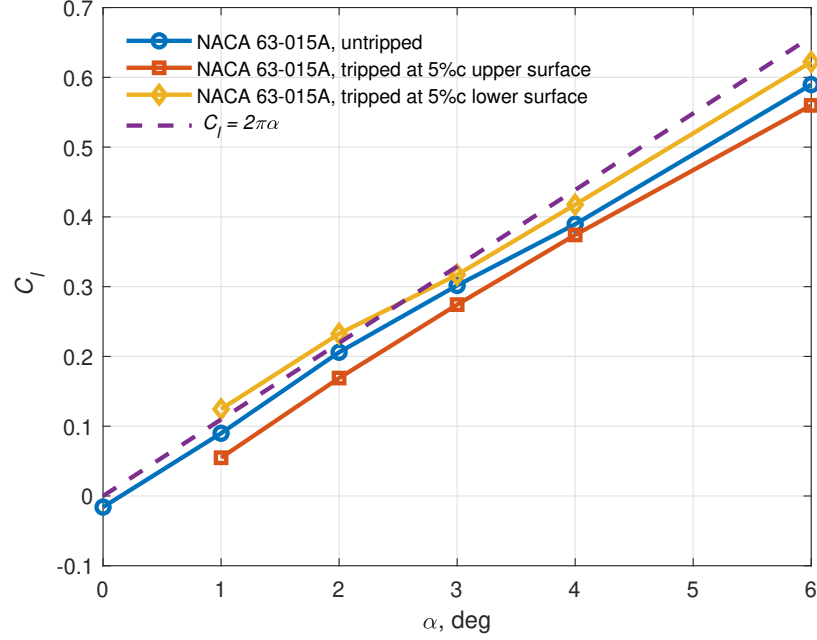


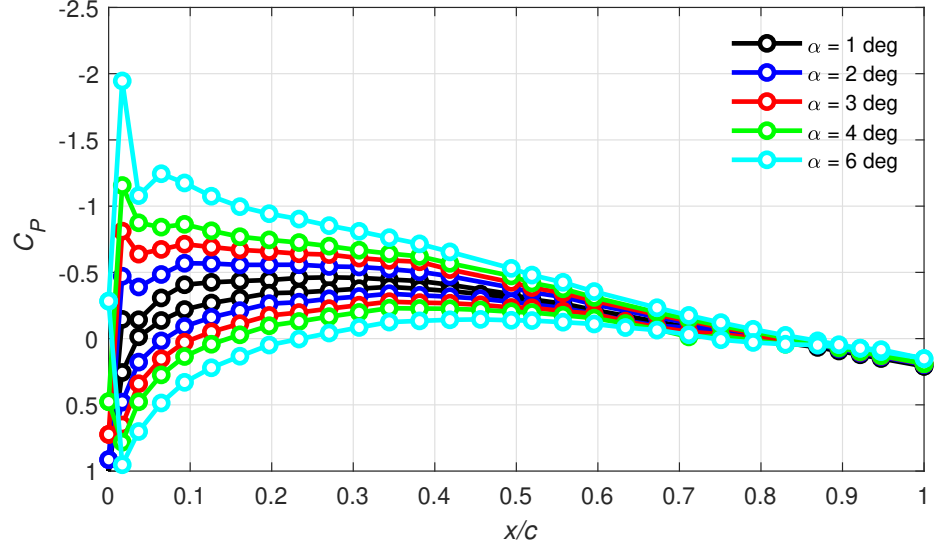
Figure 6.2: Comparison of the steady lift coefficient of the NACA 63-015A airfoil with different transition conditions.

6.2 Characteristics of the Steady Airfoil

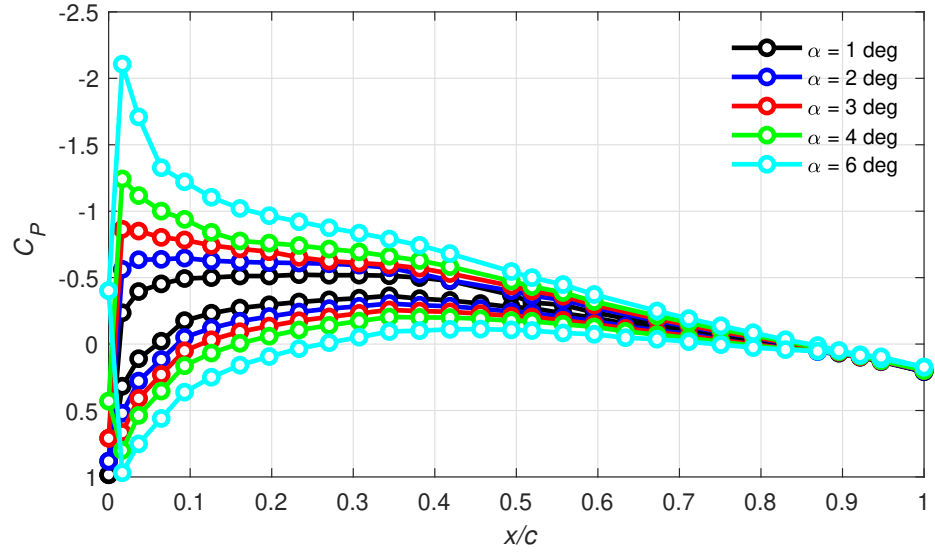
The lift characteristics of a clean NACA 63-015A airfoil at steady $Re_c = 1,500,000$ and $M_\infty = 0.2$ has been already illustrated in Figure 5.2. The effect of flow transition to steady lift characteristics is shown in Figure 6.2. The classic thin airfoil theory is included as a reference. In Figure 6.2, it is found that the lift curve slope of the three transition cases are all approximately 5.8 radians^{-1} , smaller than $2\pi \text{ radians}^{-1}$. The major difference lies in the magnitude of the lift coefficient. When only the upper surface is tripped at $0.05c$ the lift coefficient is reduced, whereas when the lower surface is tripped at $0.05c$ the lift coefficient is increased. This difference is consistent with angles of attack and clearly depends on the condition of flow transition over the airfoil.

To illustrate the effect of the forced flow transition, Figure 6.3 shows the steady C_p distribution of the tripped upper surface case and the tripped lower surface case. In Figure 6.3a, it is observed that the suction-side pressure at approximately $0.04c$ is greatly increased because of the tripping tape across $x/c = 0.04 - 0.05$. The tripping tape has a thickness on the order of the local boundary layer displacement thickness, so the local shear flow decelerates when it impinges towards the tripping tape. This tripping tape, working as a forced transition device, should create a small laminar separation bubble and turbulent reattachment downstream. Similarly, an increase in pressure can be seen on the pressure-side when the tape is placed on the pressure surface, shown in Figure 6.3b. However, the effect is less noticeable, probably due to a smaller local flow velocity and favorable pressure gradient on the pressure surface. Overall, the increased pressure due to the forced transition leads to lower C_l when transition occurs on the suction side and higher C_l when transition occurs on the pressure side.

Further comparison of the effect of flow transition to the surface pressure distribution is illustrated in Figure 6.4. The repeatability of the angle of attack in these experiments may suffer from an uncertainty of 0.1-deg, however, the C_p distribution of the three transition conditions at each angle of attack are comparable. The effect of flow transition is highlighted with the increased pressure between $x/c = 0.017$ and $x/c = 0.065$ due to the tripping tape. The panel method prediction was also compared with the experimental measurements. The comparison of C_p distribution with the panel method in Figure 6.4 emphasized again the loss of pressure information near the leading-edge in the experimental measurements, especially at higher angles of attack. Another interesting feature is observed near the airfoil trailing-edge. In both



a)



b)

Figure 6.3: Experimentally measured C_p distribution of the NACA 63-015A airfoil, a) tripped at $x/c = 0.05$ on the suction surface, b) tripped at $x/c = 0.05$ on the pressure surface.

the experimental measurements and the panel method (calculated viscous pressure), pressure on the suction surface becomes greater than pressure on the pressure-side

near the trailing-edge (at approximately $x/c = 0.90$ in experiments and $x/c = 0.95$ in the panel code, respectively). This crossover in pressure near the trailing-edge would not be captured if an inviscid analysis was made on the airfoil. Thus, it is presumed that the viscous effects further reduce the local velocity on the suction side.

Besides C_p distribution, the flow field of the steady airfoil trailing-edge at different angles of attack and flow transition conditions were measured with the BOS method. To compare the behavior of the trailing-edge stagnation streamline as a function of angles of attack and flow transition, the zero-displacement line is extracted and plotted in Figure 6.5. Similar to Chapter 5, it is observed that there is a local curvature change with respect to the entire stagnation streamline in the near wake. A strong effect of flow transition over the airfoil is observed by comparing the results. When the suction surface is tripped at $x/c = 0.05$, the airfoil trailing-edge stagnation streamline has a smaller (negative) slope, even positive at $\alpha = 1^\circ$. However, when the pressure surface is tripped at $x/c = 0.05$, the trailing-edge slope becomes greater and the positive slope at $x/c < 1.05$ is reduced. Because the forced transition alters the local shear layer momentum, this comparison further supports the hypothesis that the local shear layer momentum and its balance may affect the trailing-edge stagnation condition, specifically, the slope of the streamline emanating from the trailing-edge [37].

BOS measurements have demonstrated that the steady trailing-edge stagnation condition in the near wake has a general agreement with the Kutta condition. Very near the trailing-edge, that is within the viscous shear layer, however, the flow appears to be sensitive to the upstream transition condition and the angle of attack. As a result, the stagnation streamline was not found to bisect the sharp trailing-edge.

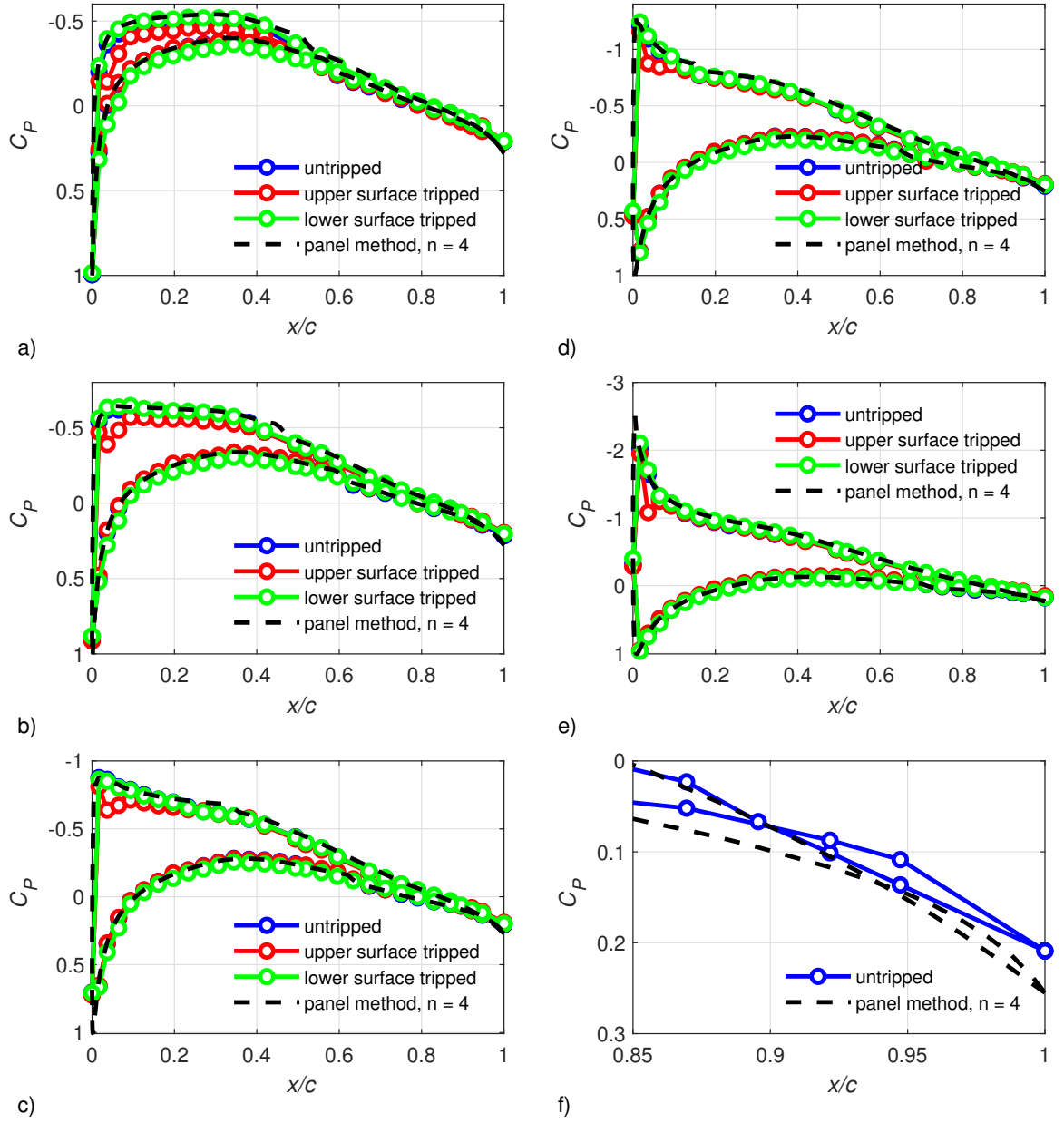


Figure 6.4: Comparison of the steady C_p distribution of the NACA 63-015A airfoil with different transition conditions, a) $\alpha = 1^\circ$, b) $\alpha = 2^\circ$, c) $\alpha = 3^\circ$, d) $\alpha = 4^\circ$, e) $\alpha = 6^\circ$, f) trailing-edge C_p , $\alpha = 4^\circ$.

This can be explained as a result of strong viscous interaction when the upper surface shear layer flow and the lower surface shear layer flow meet with each other (which

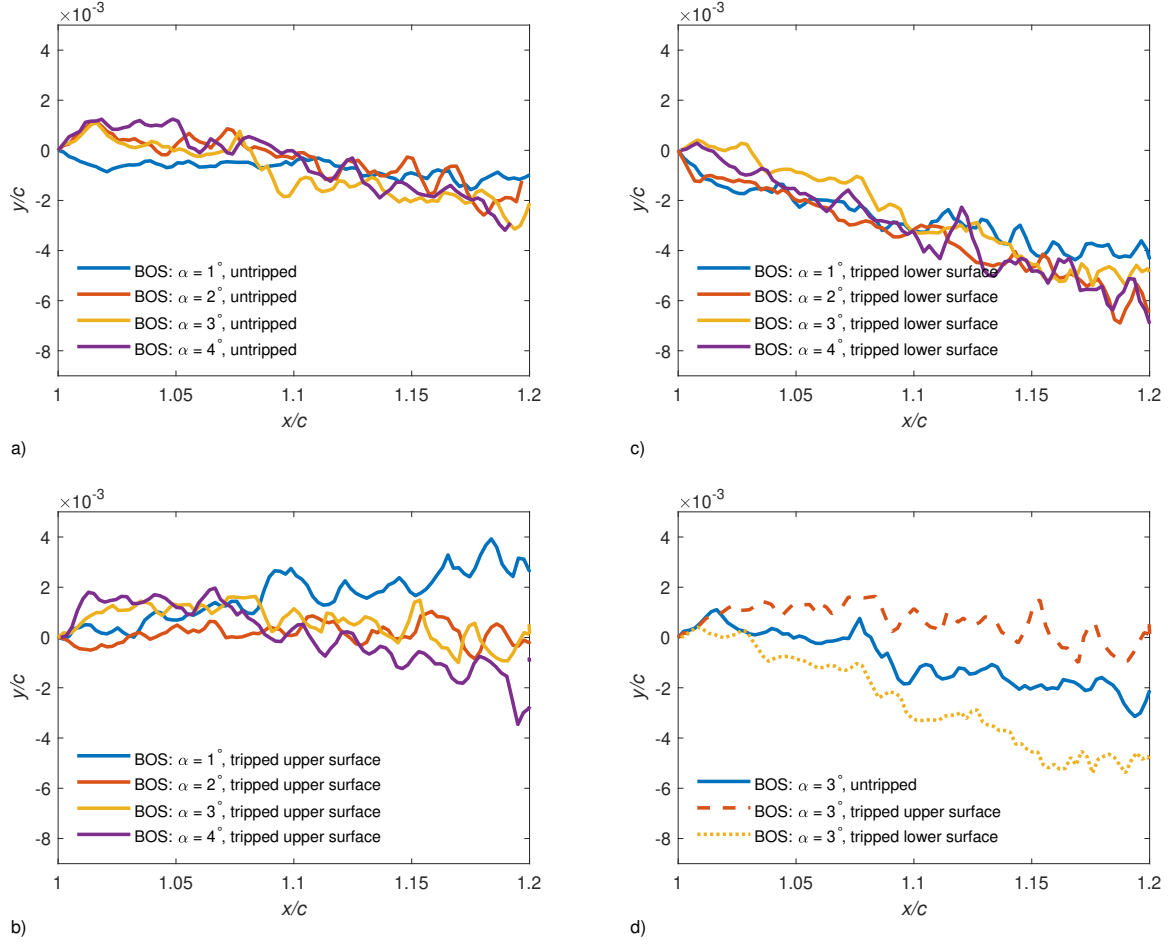


Figure 6.5: Trailing-edge stagnation streamline from BOS measurements, a) untripped airfoil, b) tripped upper surface, c) tripped lower surface, d) comparison of the three transition conditions.

is neglected by the classic Kutta condition [37, 38]). To verify this observation, near wake survey of the airfoil under steady freestream condition is also conducted.

The near wake survey includes a total of 39 sample locations with the use of three hot-film probes. The tip of the hot-film probe is about 0.3-inch downstream of the airfoil trailing-edge. The actual distance can vary slightly because of the airfoil's angle of attack (the traversing wake rake is fixed in the streamwise direction while

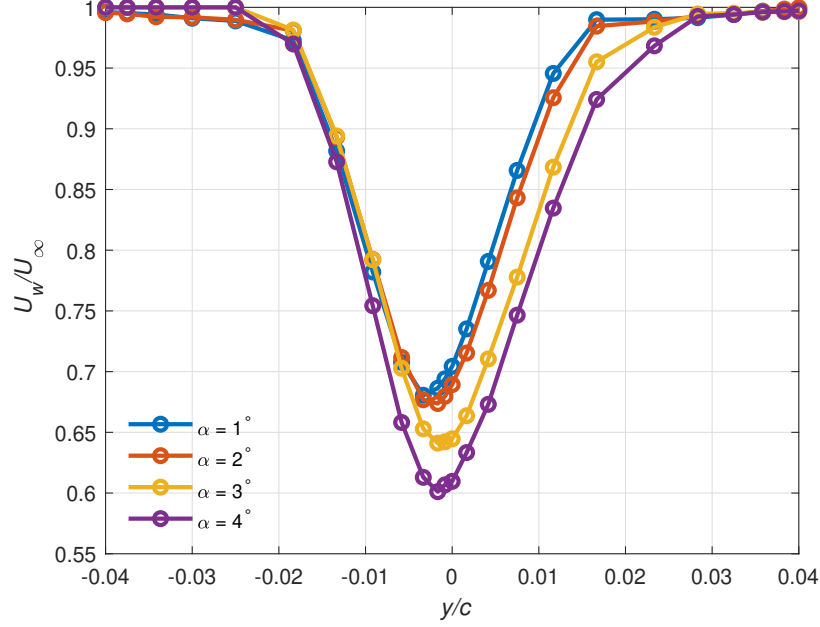


Figure 6.6: Wake profiles of the clean airfoil under steady freestream, $x/c = 1.05$.

the airfoil can be rotated with the optical window). Thus, the near wake survey is at about $0.05c$ aft of the airfoil trailing-edge. Figure 6.6 shows the wake profile of the clean airfoil at various small angles of attack. An asymmetric wake profile is clearly observed where the upper wake (wake flow from the airfoil upper surface boundary layer) has a greater thickness than the lower wake (wake flow from the airfoil lower surface boundary layer). Similar wake profiles from the NACA series can be seen in reference [73, 74]. In general, it is found that the (upper) wake thickness grows with the angle of attack because of the growing airfoil boundary layer thickness. In addition, the velocity deficit in the wake grows with the angle of attack because of the growing adverse pressure gradient along the airfoil. This is consistent with similar wake surveys reported previously [74].

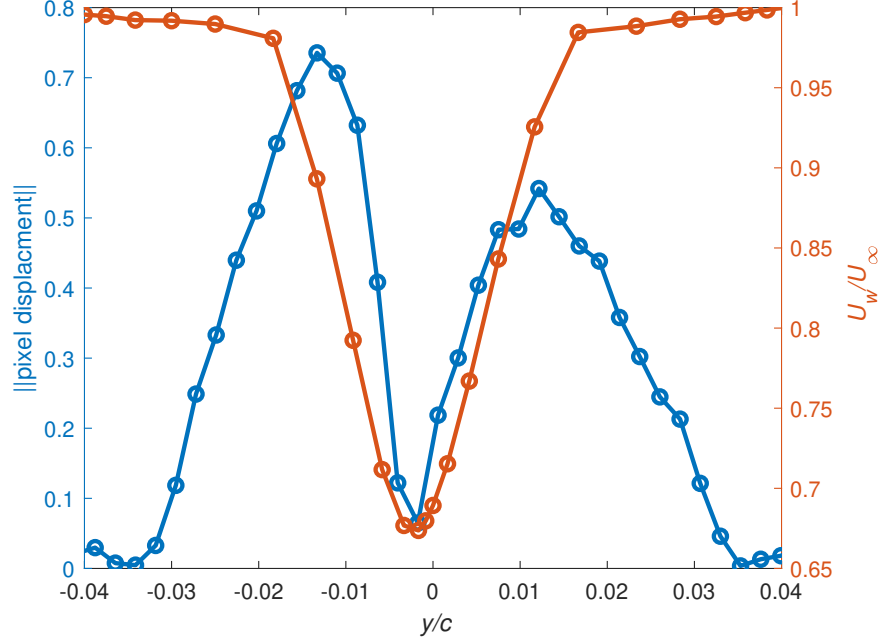


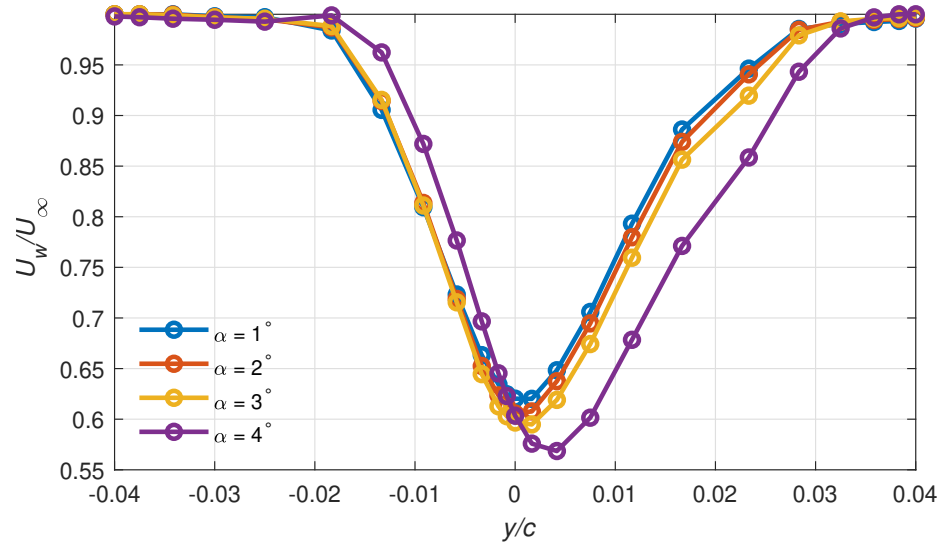
Figure 6.7: Comparison of the displacement profile from BOS measurements and velocity profile from the wake survey, $\alpha = 2^\circ$, $x/c = 1.05$.

In addition, the displacement profile from BOS measurements and the velocity wake survey at $x/c = 1.05$ are compared in Figure 6.7. It shows that the displacement wake, which equivalents to a thermal shear layer, is thicker than the velocity wake. This is because Prandtl number of the flow smaller than 1. Furthermore, the wake centerline at this streamwise location is comparable between the two measuring techniques.

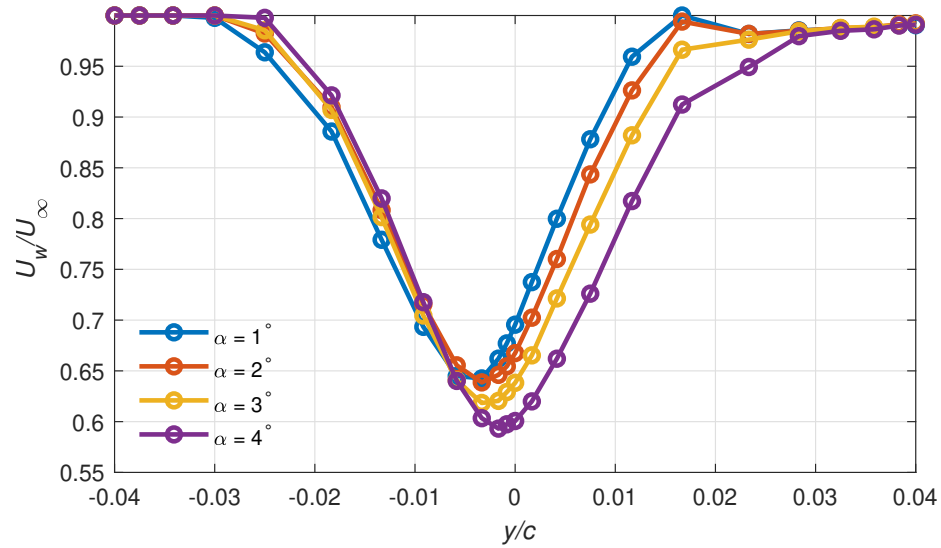
The two forced transition cases are also investigated and the steady wake profiles are shown in Figure 6.8. The effect of angle of attack to wake thickness and velocity deficit are found to be similar in these wake profiles. Additionally, the effect of forced transition to the wake profile is given in Figure 6.9. It highlights the greater shear layer thickness as a result of earlier transition to turbulent boundary layer. In

addition, the wake centerline or the location of the maximum velocity deficit can also be seen to vary among the transition conditions. At $\alpha = 4^\circ$, the wake profiles of the clean airfoil and the tripped lower surface case become similar in terms of magnitude and thickness, whereas the tripped upper surface case is still significantly different. On one hand, this is due to that the clean airfoil and tripped lower surface experience similar natural transition on the suction side. On the other hand, this is likely due to the smaller effect of tripping on the lower surface with the angle of attack where the local flow velocity is smaller and the pressure gradient is favorable. Because natural transition and forced transition at $x/c = 0.05$ become closer beyond $\alpha = 4^\circ$, no higher angles of attack were sampled in wake survey to save the experimental cost.

The main objective of the near wake survey is to quantify the shear layer momentum, however, the finite number of sampling locations may not be sufficient for accurate quantitative analysis. As a result, it is necessary to construct a robust curve fit based on measurements from the limited sampling locations. Self-similarity was discovered for wakes further downstream (far wake measurements) and far wake profiles can be plotted following a Gaussian distribution [60, 73, 75]. However, this does not apply to near wake or very near wake because the wake is still developing. In order to fit the experimental data, several curve-fitting strategies were attempted. The greatest challenge is to take into account the multiple inflection points of the wake profile. At last, it is decided to apply a second-order polynomial fit on three sample points about the measured maximum velocity deficit. This polynomial fit would interpolate the location and magnitude of the maximum velocity deficit. In addition, error function was used on each side of the wake outside of this region. The use of the error function provides the inflection point and allows a very good fit of



a)



b)

Figure 6.8: Steady wake profile, $x/c = 1.05$, a) tripped upper surface, b) tripped lower surface.

the experimental wake profile. This curve-fitting strategy is demonstrated in Figure 6.10. It can be seen that there is a discontinuity at the point when two curve fits meet with each other (intersection of the blue and the red curve). This discontinuity

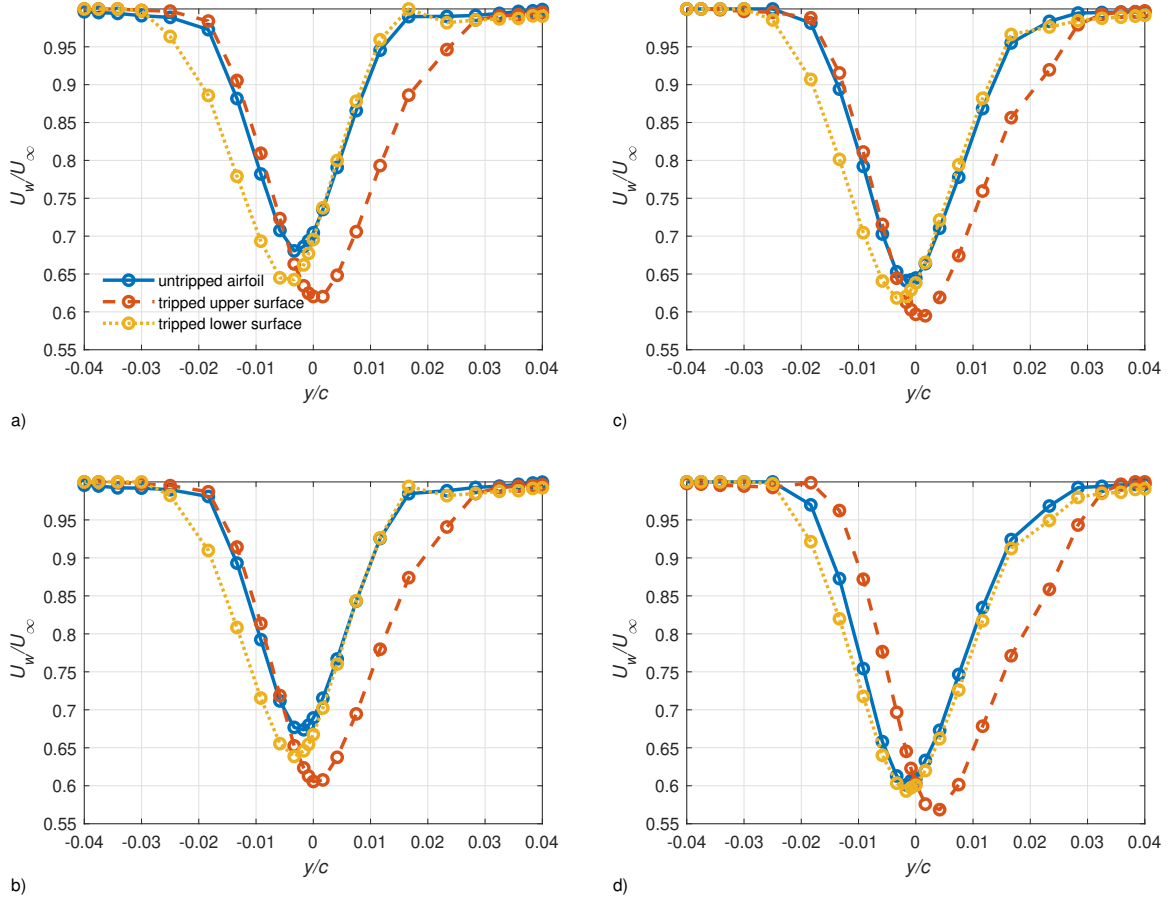


Figure 6.9: Comparison of the steady wake profile for different transition conditions, a) $\alpha = 1^\circ$, b) $\alpha = 2^\circ$, c) $\alpha = 3^\circ$, d) $\alpha = 4^\circ$.

cannot be avoided but the difference is one to two orders of magnitude smaller than the local velocity magnitude. Thus, the discontinuity is considered acceptable and the curve-fitting is used to integrate the wake profile in the following sections.

In Figure 6.8 and 6.9, it has been observed that the wake centerline (*i.e.*, the point of the maximum velocity deficit) shifts as a function of the angle of attack. With the help of curve-fitting, this movement is further investigated and compared to BOS measurements. Note that the wake survey and the BOS measurements are separate

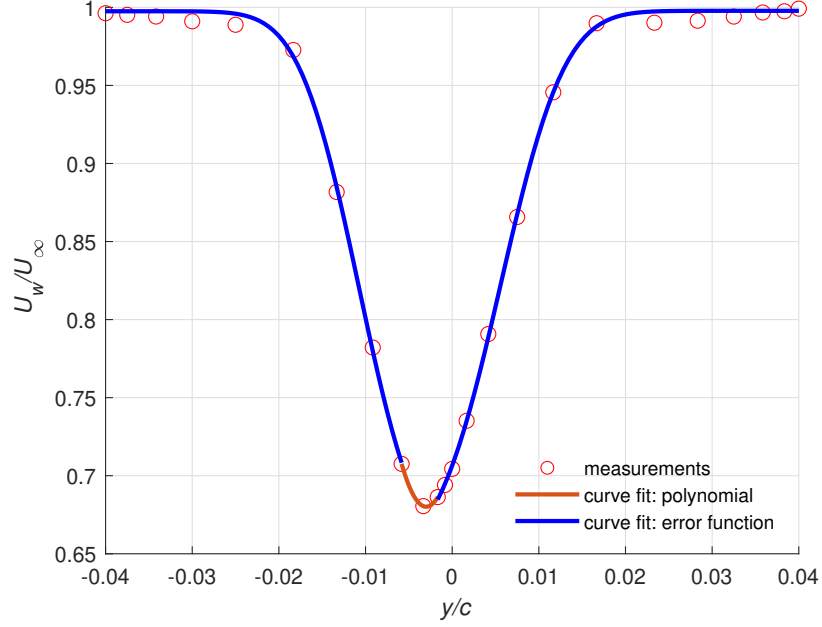


Figure 6.10: Illustration of curve-fitting strategy for the experimentally measured wake data.

experiments so there are uncertainties of the repeatability of the airfoil's angle of attack (on the order of 0.1-degree) and the added blockage of the traversing wake rake may alter the local flow field. In the previous section, the steady trailing-edge stagnation streamline of various flow conditions have been shown. This is compared with near wake survey in Figure 6.11. Despite the absolute location of the wake centerline at $x/c = 1.05$ is not exactly the same, the effects of the angle of attack and the flow transition to the wake centerline are found to be similar between the two different measurements. As a result, the near wake survey verified the BOS measurements about the stagnation streamline of the steady NACA 63-015A airfoil.

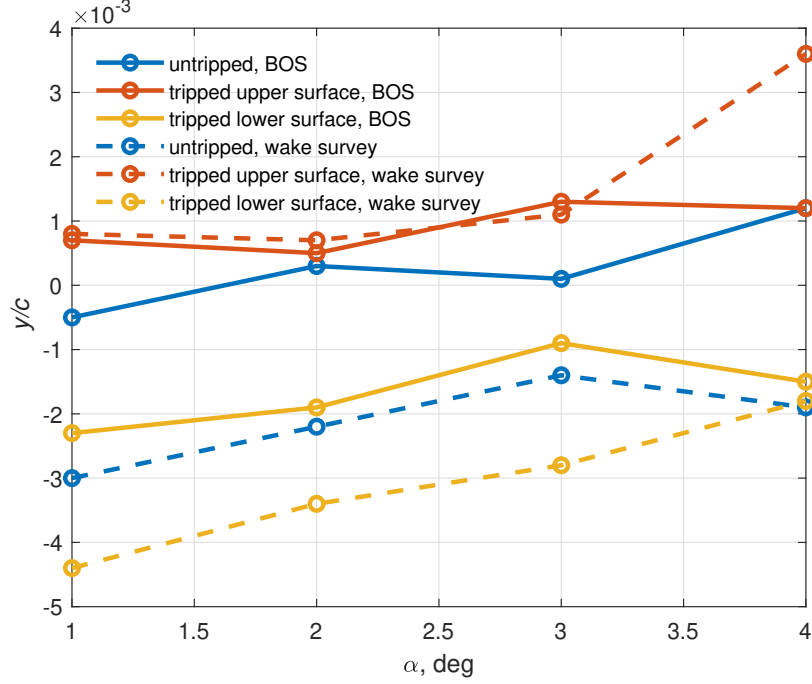


Figure 6.11: Comparison of the steady wake centerline at $x/c = 1.05$.

In Chapter 5, it has been hypothesized that the viscous shear layer has a strong effect on the trailing-edge stagnation condition. The viscous Kutta condition is proposed to be affected by the local momentum balance between the upper and the lower shear layer. As a result, the momentum balance of the near wake (assuming constant density) is evaluated as

$$net \ momentum \ flux = \int_0^{\delta/c} \left(\frac{u_w}{u_\infty} \right)_{upper}^2 dy - \int_0^{\delta/c} \left(\frac{u_w}{u_\infty} \right)_{lower}^2 dy \quad (6.4)$$

which is based on the momentum flux defined in reference [37]

$$J = \int_0^\delta u_w^2 dy \quad (6.5)$$

with the modification to non-dimensionalize the term.

The net momentum flux of the steady airfoil wake at $x/c = 1.05$ is plotted in Figure 6.12. Three transition conditions are compared and the effect of the angle of attack is also demonstrated. Firstly, it is shown that the net momentum flux decreases with the investigated angle of attack. This is because of stronger adverse pressure gradient on the airfoil suction surface with the growing angle of attack. As a result, the upper boundary layer flow suffers more momentum loss relative to the lower boundary layer flow. However, this trend is not likely to be monotonic. For a symmetric airfoil at zero angle of attack, the net momentum flux at the trailing-edge is supposed to be zero because of symmetry. As the angle of attack increases, the asymmetry is supposed to produce a positive net momentum flux at the trailing-edge because the shear layer on the suction surface is thicker than the shear layer on the pressure surface. Thus, it is believed that the net momentum flux should increase from zero as the formation of asymmetric boundary layer and wake, then decreases as a result of stronger adverse pressure gradient on the suction side, which eventually would cause flow separation.

Secondly, it is found that the magnitude of the net shear layer momentum flux of the clean airfoil falls between the two forced transition cases. In Figure 6.12, the case of the tripped upper surface always has a smaller net momentum flux at the near wake whereas the case of the tripped lower surface always possesses a stronger net momentum flux. This is believed to be caused by the forced transition where a laminar separation bubble and reattachment cause additional momentum loss that affects the momentum balance between the two shear layers aft the trailing-edge. As a result, the tripped shear layer at the trailing-edge and near wake possesses smaller momentum and the momentum balance favors the other side. Such observed net

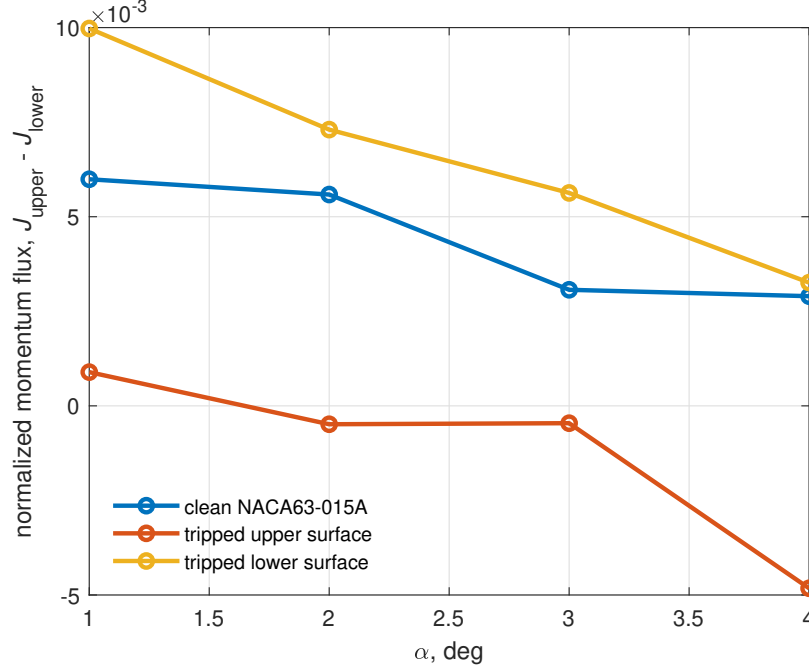


Figure 6.12: Momentum balance of the shear layer near the steady airfoil trailing-edge.

momentum flux in Figure 6.12 supports the BOS measurements and the hypothesis in Chapter 5 that the trailing-edge stagnation condition is affected by the momentum balance within the shear layer.

6.3 Characteristics of the Surging Airfoil

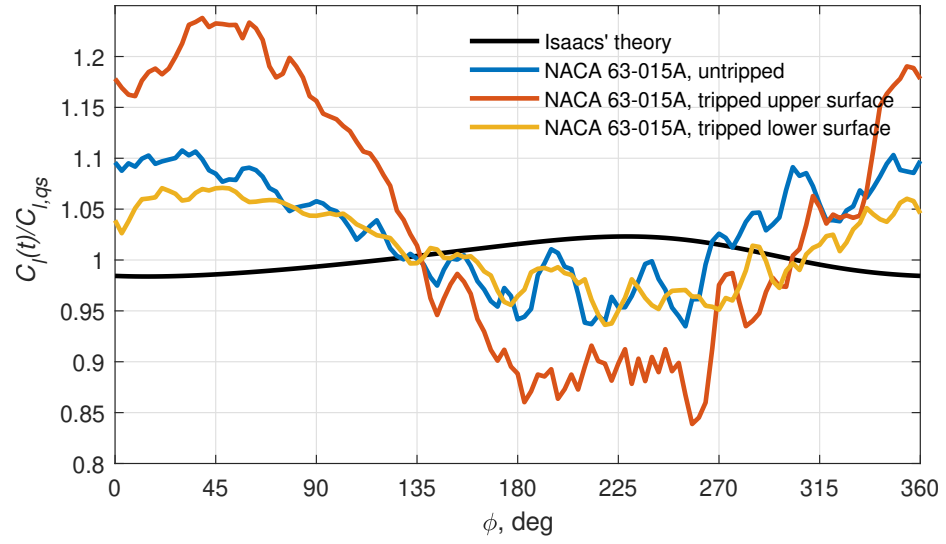
The unsteady lift frequency response of the NACA 63-015A airfoil with different flow transition conditions is presented in Figures 6.13 and 6.14. Similar to the clean airfoil data, the maximum lift overshoot and undershoot and the corresponding phases are in disagreement with Isaacs' theory for the forced transition cases. That is, when the freestream velocity is increased from the mean to the maximum (from $\phi = 0^\circ$ to

$\phi = 90^\circ$) the unsteady lift coefficient is generally increasing and is above the quasi-steady value, whereas when velocity is decreasing (from $\phi = 90^\circ$ to $\phi = 270^\circ$) the unsteady lift coefficient is decreasing and reaches maximum lift undershoot. The phase of the maximum lift overshoot and undershoot also indicate that there is a phase delay between the freestream velocity and the lift coefficient.

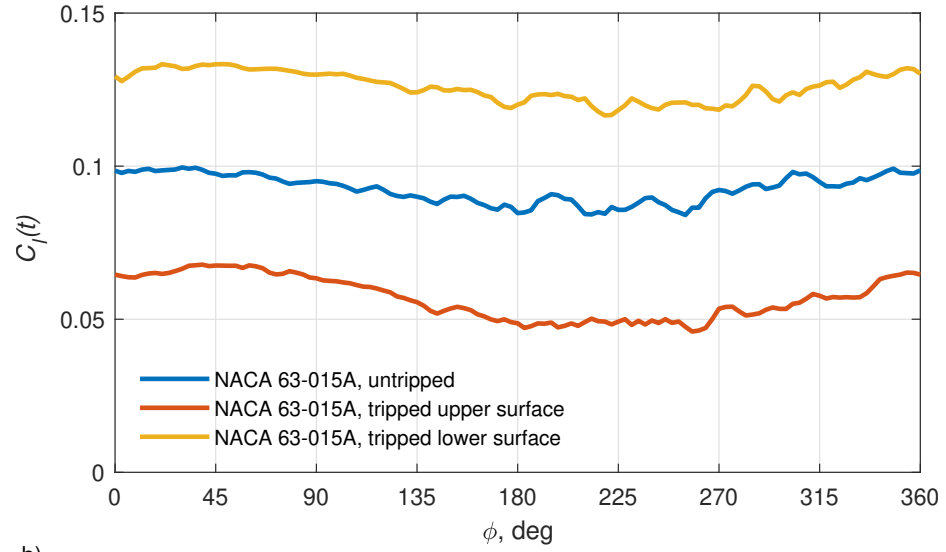
A few interesting features are observed in Figures 6.13 and 6.14. First, the difference among the three transition conditions are greatest at $\alpha = 1^\circ$. However, this is found to be related to the percentage difference being greater when the quasi-steady C_l is smaller at lower angles of attack. An illustration is shown in Figure 6.13b where the unsteady C_l of the three transition conditions are compared. Similar to the comparison of the steady C_l , the case of only tripping the upper surface possesses a smaller lift magnitude and the case of only tripping the lower surface has a greater lift magnitude. As a result, when these unsteady C_l are normalized by their corresponding quasi-steady values, the normalized magnitude is different in Figure 6.13a. This shows that flow transition can affect both C_l and $C_l(t)/C_{l,qs}$ profiles.

Second, the effect of the angle of attack is found to influence the phase angle of the maximum lift overshoot. The phase angle changed from $\phi \approx 45^\circ$ when $\alpha = 1^\circ$ to $\phi \approx 90^\circ$ when $\alpha = 3^\circ$ and beyond. However, the maximum lift undershoot stays relatively constant between $\phi = 225^\circ$ and $\phi = 270^\circ$ with higher-frequency oscillations. Such effect of the angle of attack is absent in Isaacs' theory but was also observed by Strangfeld *et al* [7]. This effect may be related to the boundary layer development on the airfoil surface.

Finally, when the angle of attack grows to $\alpha = 6^\circ$ and beyond, the natural transition on the suction surface will occur near or upstream of the tripping tape. Thus,



a)



b)

Figure 6.13: Comparison of lift frequency response of the NACA 63-015A airfoil with different transition conditions, $\sigma = 0.21, k = 0.025, \alpha = 1^\circ$ a) normalized unsteady lift coefficient, b) unsteady lift coefficient.

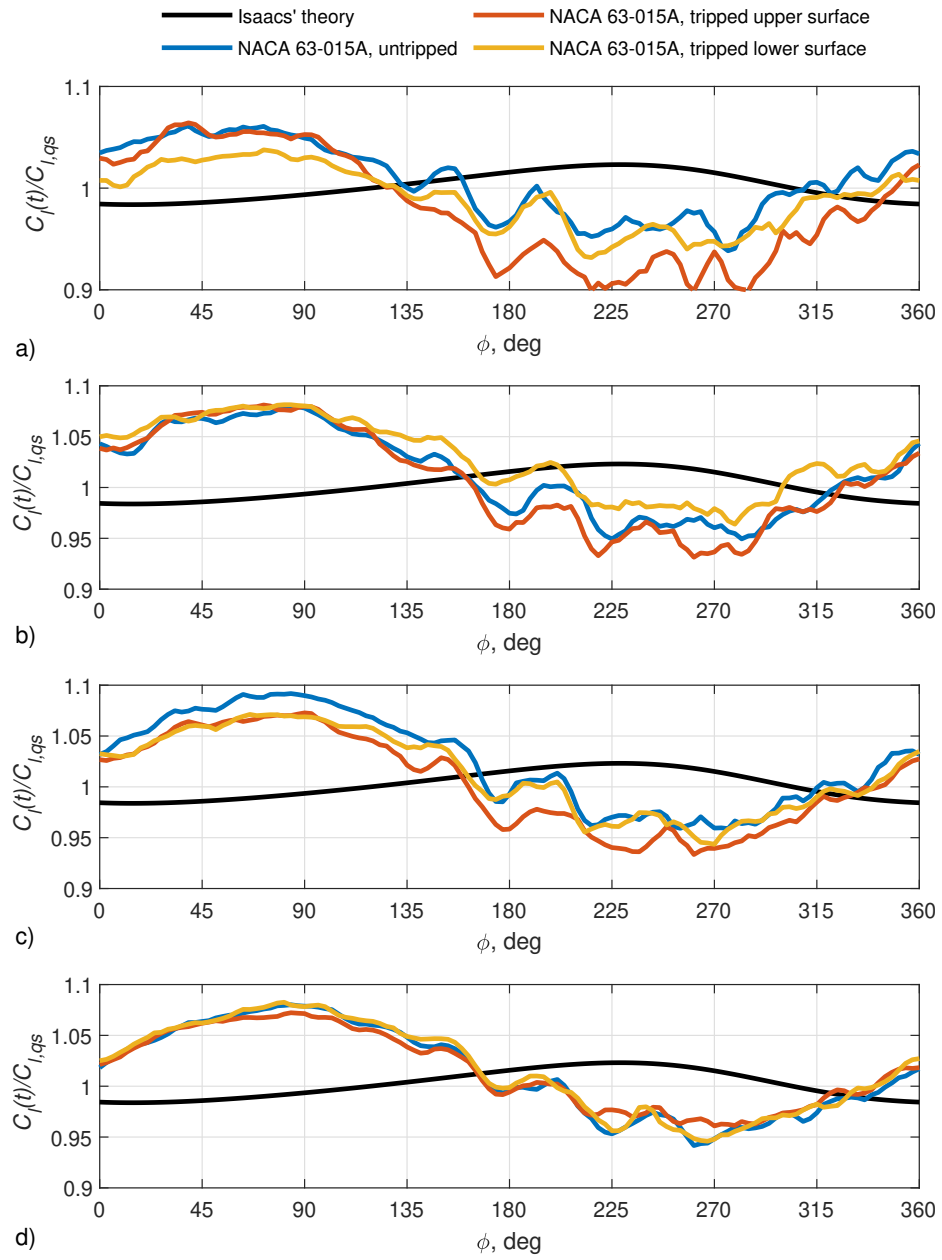


Figure 6.14: Comparison of the normalized unsteady lift coefficient with different transition conditions, $\sigma = 0.21, k = 0.025$, a) $\alpha = 2^\circ$, b) $\alpha = 3^\circ$, c) $\alpha = 4^\circ$, d) $\alpha = 6^\circ$.

one may assume the performance of the three transition cases will become similar. This seems to be true in Figure 6.14. However, the unsteady C_l continues to show the similar magnitude difference at these angles of attack, see Figure 6.15. This is inferred that the tripping tape still has a negative effect on the local shear layer development even though the natural transition has occurred. For the tripping tape on the lower surface, it continues to force transition at these angles of attack since the natural transition is always further downstream on the pressure side (this can be verified with negative angles of attack in Table 5.3).

To summarize, flow transition on the airfoil does produce different unsteady lift performance. The direct impact is the measured local surface pressure distribution. In addition, the momentum loss due to the laminar separation bubble, turbulent reattachment, and mixing is present. Its effect is likely different between the natural and the forced transition. This effect may play some role on the final pressure and lift data and will be explored further in the following sections.

Furthermore, the trailing-edge stagnation streakline under the surging condition is examined. The dynamics of the stagnation streakline is demonstrated in Figure 6.16, where it can be seen that the curvature of the very near wake ($x/c < 1.02$) and the position of the streakline oscillate with the freestream velocity. Again, the slope of the streakline near the vicinity of the airfoil trailing-edge is positive and its position with respect to the airfoil trailing-edge is lower when the freestream velocity is at the maximum, whereas the slope near the vicinity of the airfoil trailing-edge can become negative and the position with respect to the airfoil trailing-edge is higher. While the exact location of the airfoil trailing-edge is ambiguous due to the limitation of this technique (*e.g.*, due to image blur near the airfoil surface), the altered streakline

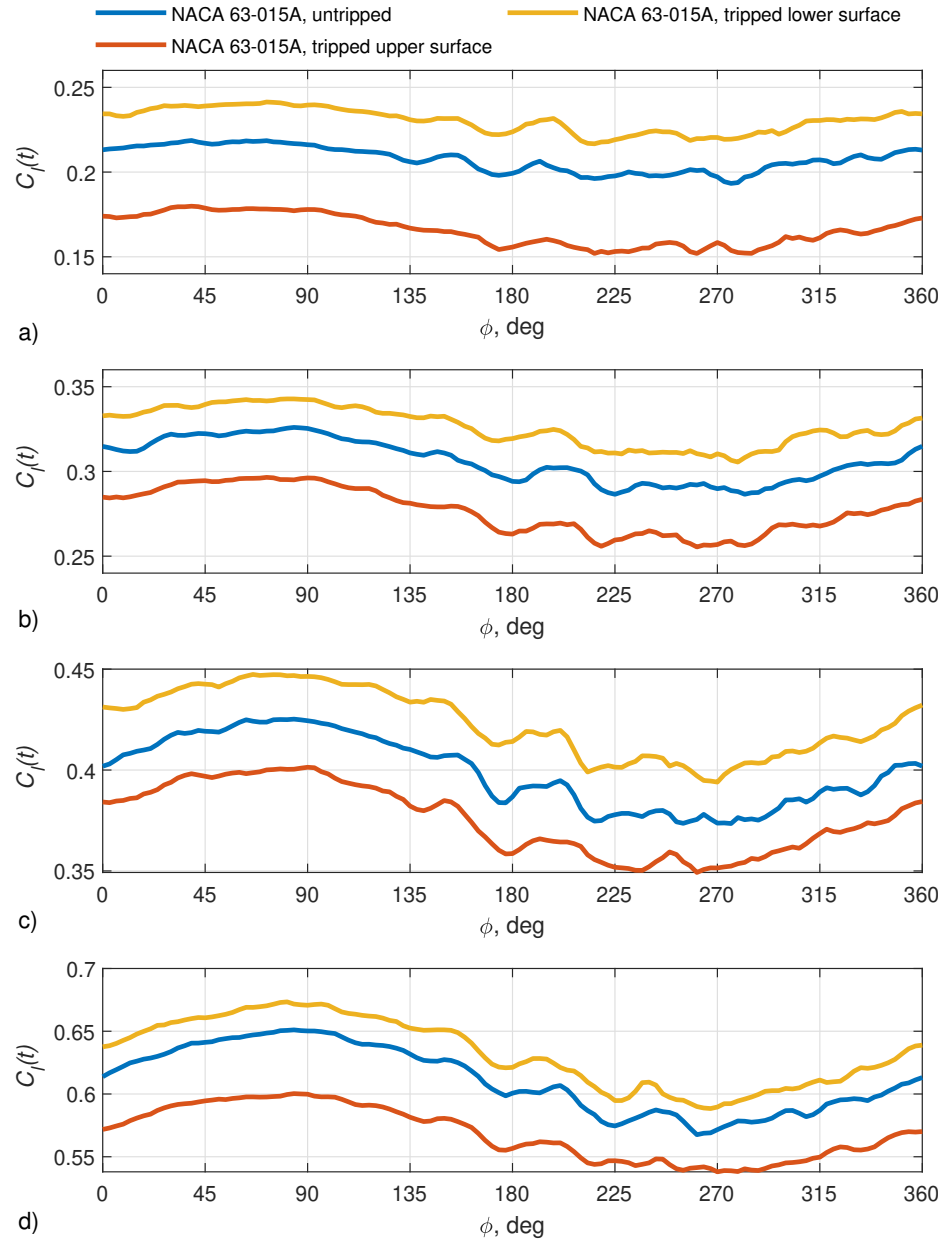


Figure 6.15: Comparison of the unsteady lift coefficient with different transition conditions, $\sigma = 0.21$, $k = 0.025$, a) $\alpha = 2^\circ$, b) $\alpha = 3^\circ$, c) $\alpha = 4^\circ$, d) $\alpha = 6^\circ$.

position may indicate a small region of flow separation right upstream of the trailing-edge when the freestream velocity is near the minimum.

To verify the unsteady stagnation streakline and the hypothesis of shear layer momentum balance at the trailing-edge, the wake survey is applied to the surging NACA 63-015A airfoil. Figure 6.17 demonstrates the dynamics of the near wake flow at selected phases of the surging cycle. It is found that the upper wake velocity increases in both the absolute magnitude and in the normalized format (u_w/u_∞) when the freestream velocity increases. Similarly, the velocity magnitude of the upper wake decreases when the freestream velocity decreases. Although the lower wake velocity profile does not show the same level of change (analogous to the more stable pressure-side C_p), it is found that the maximum wake deficit and the wake centerline location vary with the freestream velocity. Despite the fact that the wake survey was only conducted at one fixed streamwise location, the oscillation of the wake center at this near wake location was captured and is shown in Figure 6.18. It is clearly shown that the wake center oscillates in a sinusoidal fashion similar to that from BOS measurements, where the wake center moves towards the $-y/c$ when the freestream velocity is increasing and moves towards the $+y/c$ when the freestream velocity is decreasing. The relative position with respect to the trailing-edge ($y/c = 0$) and the effect of different flow transition conditions are also consistent with the BOS measurements.

At this point, both the near wake survey and the BOS measurements have confirmed the existence of the unsteady trailing-edge flow and the classic Kutta condition for an airfoil under steady freestream is clearly violated in Isaacs' surging airfoil theory. It is also hypothesized that the cause of this unsteadiness is the momentum

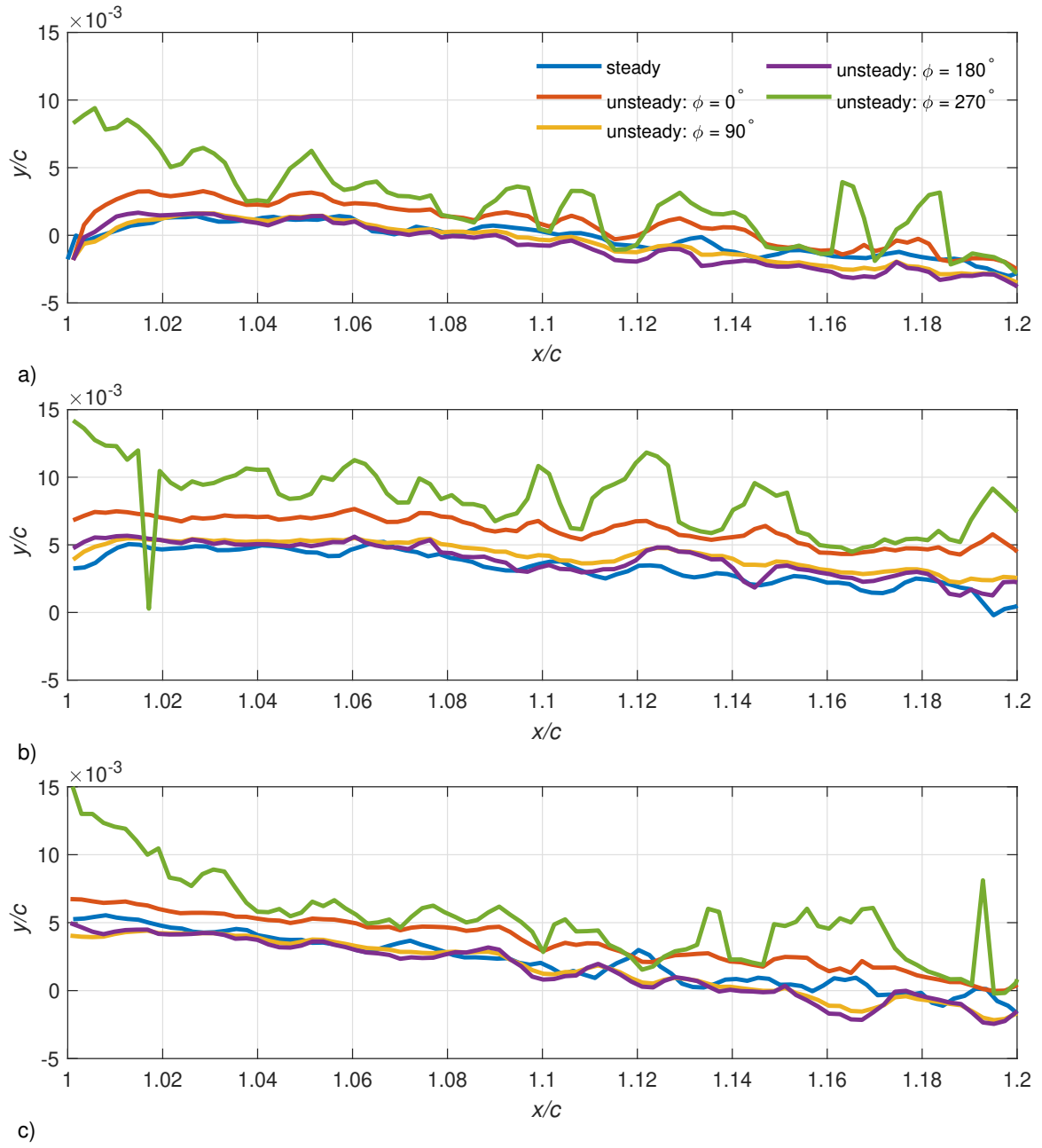


Figure 6.16: Comparison of the surging airfoil trailing-edge stagnation condition at $\alpha = 4^\circ$, a) untripped airfoil, b) tripped upper surface, c) tripped lower surface.

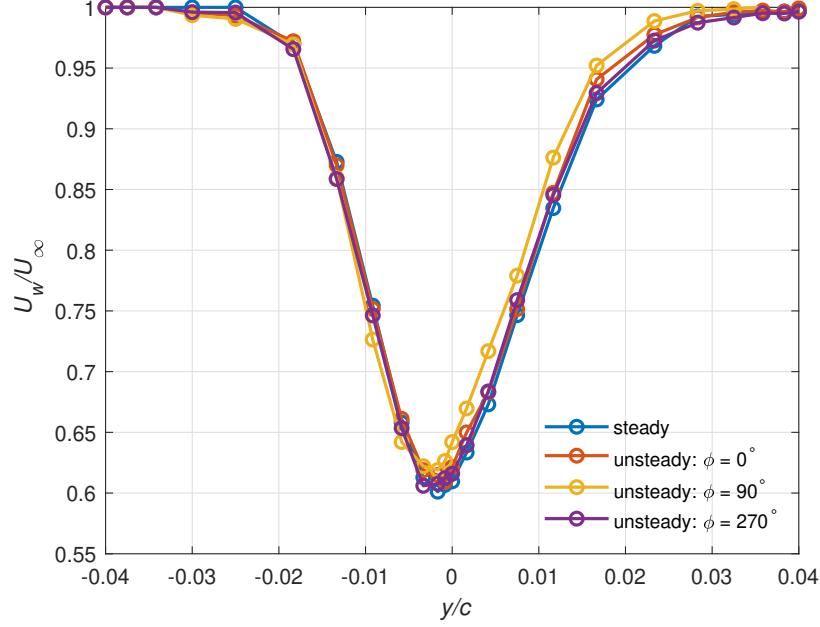


Figure 6.17: Wake profile of the NACA 63-015A airfoil during the surging freestream condition, $x/c = 1.05$.

balance of the two shear layers at the trailing-edge. With the near wake survey, the dynamics of the shear layer momentum balance can be characterized with Equation 6.5. The results are shown in Figure 6.19. All angles of attack and flow transition conditions are plotted over the phases of the freestream oscillation.

The net momentum flux is found to increase when the freestream velocity increases ($\phi = 0 - 90^\circ$) and decrease when the freestream velocity decreases ($\phi = 90 - 270^\circ$). This indicates when the freestream velocity increases, the momentum of the upper shear layer out-weighs the momentum of the lower shear layer. Thus, the balance of the two shear layers favors the upper side and the local equilibrium point/line is tilting towards the lower side. In other words, the equilibrium condition of the two mixing shear layers is moving towards $-y/c$, or more precisely, the curvature of

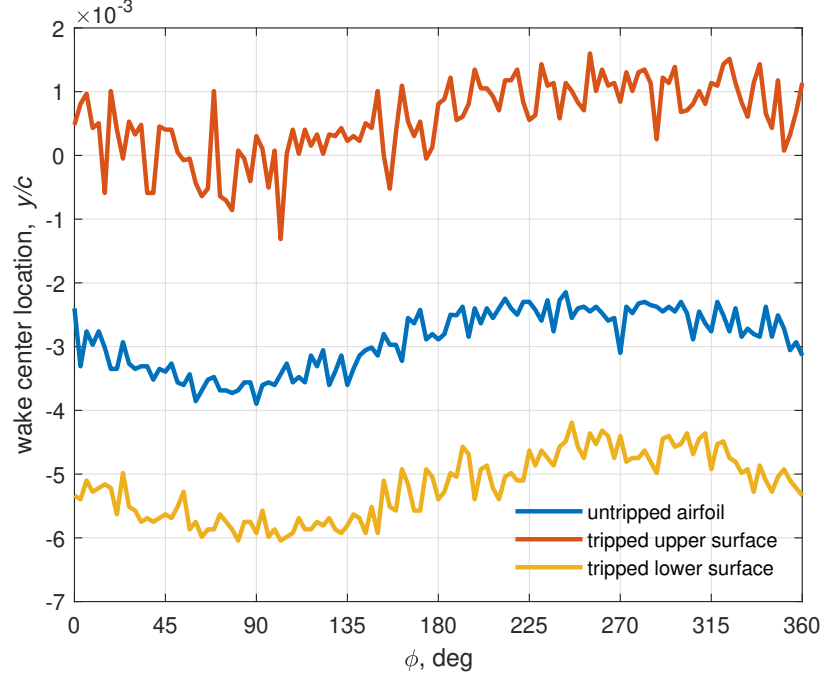


Figure 6.18: Dynamics of the unsteady wake centerline during the surging freestream, $\alpha = 1^\circ$, $x/c = 1.05$.

the stagnation streamline is tilting towards the lower side. Similarly, when the net momentum flux is decreasing this balance is tilting towards the upper side. Such results agree with the BOS measurements.

Another interesting observation is that for the clean airfoil and the tripped lower surface case, the net momentum flux stays positive during the velocity cycle, meaning the upper shear layer always possesses a stronger momentum than the lower shear layer. However, when the upper surface is tripped, it is found that the net momentum flux can become negative, especially for $\alpha = 4^\circ$. This can be explained as the tripping tape affects the suction and the pressure side differently. On the suction side, the flow along the surface has a stronger acceleration initially and quickly encounters

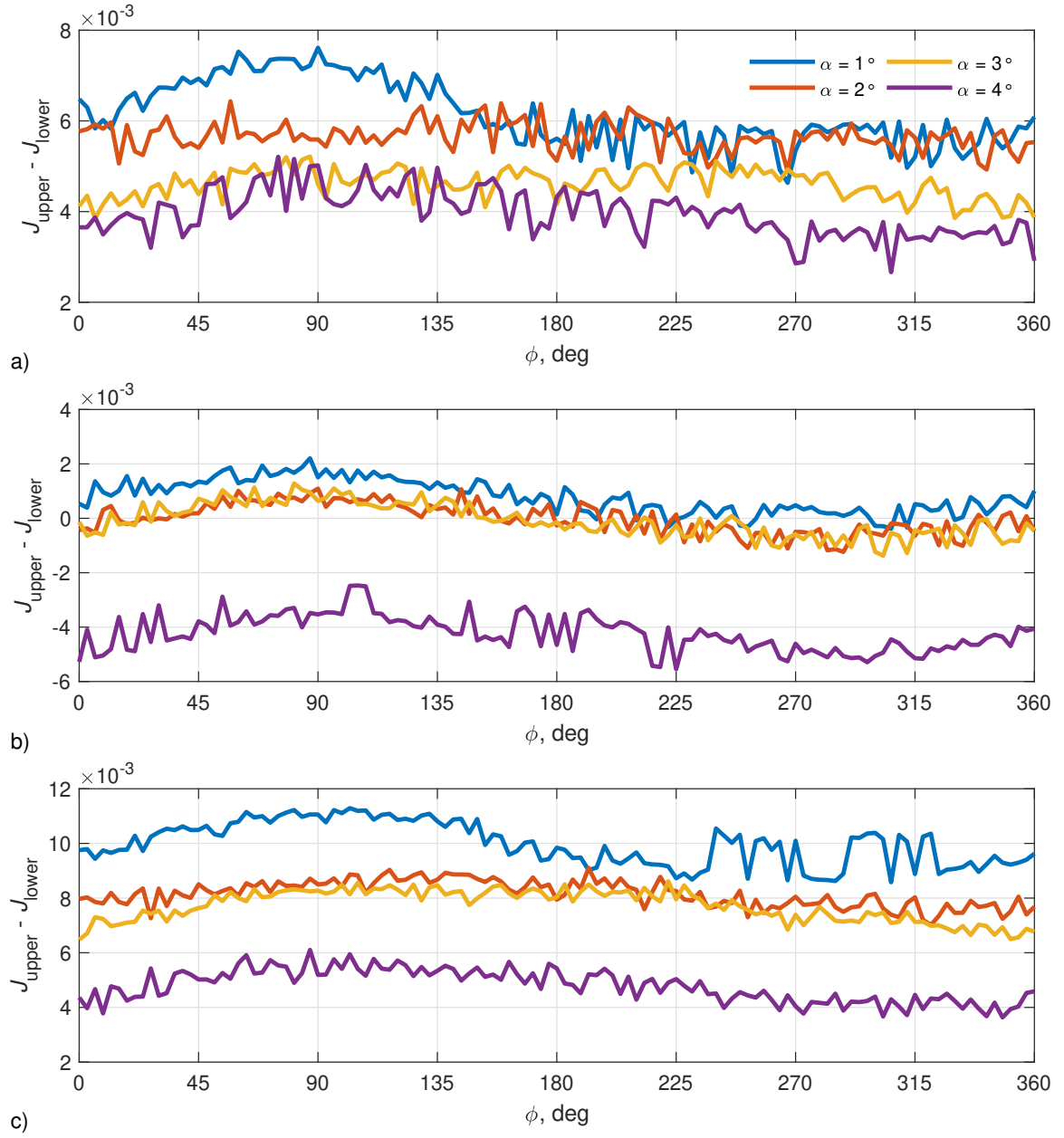


Figure 6.19: Momentum balance of the shear layer at the near wake as a function of the angle of attack, a) untripped airfoil, b) tripped upper surface, c) tripped lower surface.

the (stronger) adverse pressure gradient. Thus, the forced transition is believed to have a stronger negative effect on the local shear layer momentum. As such the net momentum flux is reduced when the two shear layers meet aft of the trailing-edge.

The effect of the flow transition to the local shear layer momentum balance is further illustrated in Figure 6.20. As described above, the tripped upper surface is believed to suffer a greater momentum loss on the suction side and the tripped lower surface suffers a greater momentum loss on the pressure side. At $\alpha = 4^\circ$, the clean airfoil and the case of tripped lower surface have a similar net momentum flux near the trailing-edge. This is believed as a result of flow transition on the suction surface being similar between the two cases and the tripping effect on the pressure surface becomes less significant. By examining Figure 6.15, it is found that the lift coefficient of the clean airfoil and the tripped lower surface case also becomes closer, agreeing with the closer net momentum flux measured in the near wake.

Finally, the effect of the unsteady trailing-edge stagnation condition is correlated to the unsteady lift response assuming the effect is equivalent to a dynamic cambering effect of a Gurney flap at the trailing-edge. The result is illustrated in Figure 6.21. It is observed that the first approximation produces a lift response closer to the measured lift coefficient, especially the corresponding phases of maximum lift overshoot and undershoot. This is an evidence that the oscillatory trailing-edge stagnation condition can have a strong effect on the measured lift coefficient, at least under the current surging flow condition. The remaining discrepancy may lie in the fundamental formulation of this approximation. In addition, it may indicate there are other effects contributing to the lift coefficient, such as the dynamics of shed vorticity in the

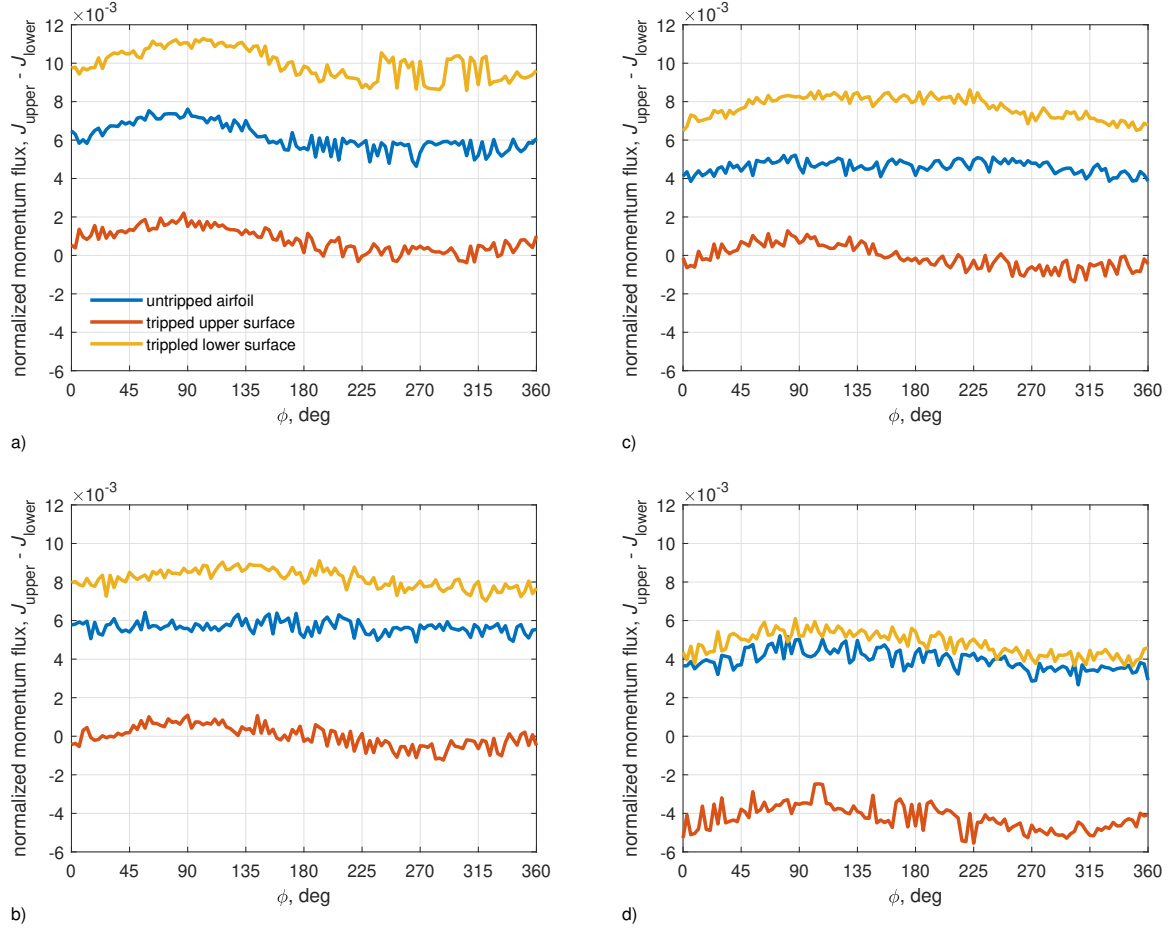


Figure 6.20: Momentum balance of the shear layer at the near wake as a function of flow transition, a) $\alpha = 1^\circ$, b) $\alpha = 2^\circ$, c) $\alpha = 3^\circ$, d) $\alpha = 4^\circ$.

wake from Isaacs' theory. The coupling of multiple effects may not be easily resolved if the effects are not linear.

6.4 Summary

In this chapter a natural-laminar-flow airfoil NACA 63-015A with forced transition was investigated under the surging flow condition. Its lift characteristics, specifically, unsteady responses were compared with Isaacs' surging airfoil theory. Similar to

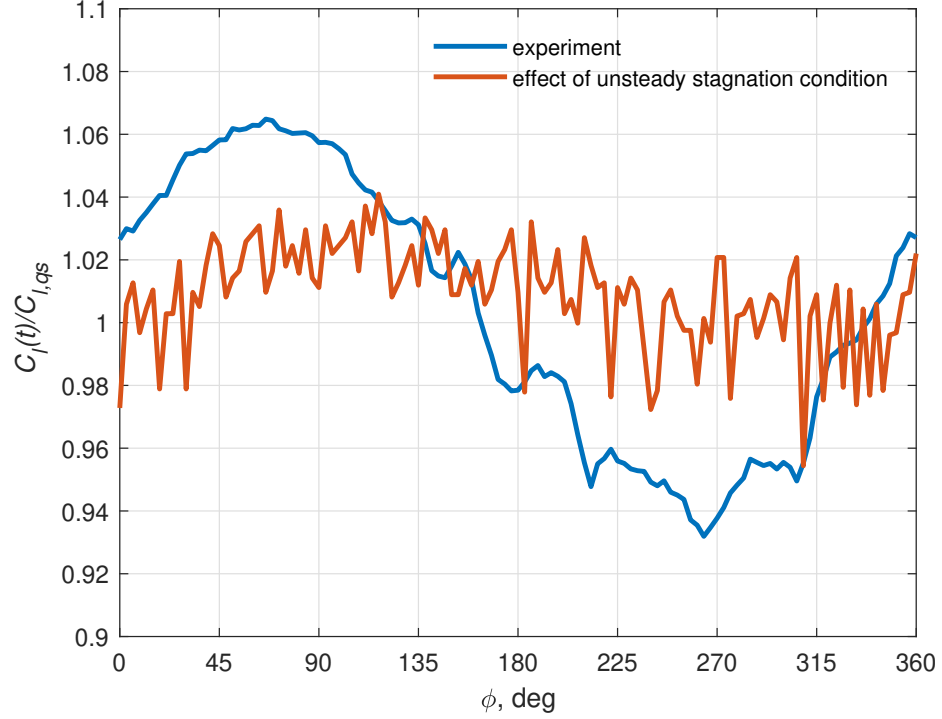


Figure 6.21: Effect of the unsteady trailing-edge stagnation condition to the unsteady lift response, clean NACA 63-015A airfoil, $\alpha = 4^\circ$.

surging airfoil investigation in Chapter 5, it was found that its lift overshoot and undershoot varies from the prediction. In addition, it is found that tripping on the upper surface lead to a reduced lift coefficient whereas tripping on the lower surface lead to an amplified lift coefficient. Optical measurements utilizing the BOS technique were also applied to characterize the trailing-edge flow field. Results also indicate an oscillatory trailing-edge stagnation condition. Furthermore, a near wake survey of the airfoil was conducted with a traversing wake rake to characterize the trailing-edge flow quantitatively. The wake survey showed a velocity wake profile and its dynamics during the surging flow, a further evidence of an unsteady trailing-edge stagnation condition. Specifically, it is observed that the wake centerline oscillates

in time which agrees with the BOS measurements, and the shear layer momentum balance oscillates in support of the hypothesis that the stagnation condition is dictated by the shear layer momentum. The shear layer momentum balance was calculated using the normalized momentum flux equation and showed a correlation with the unsteady lift coefficient. It was found that a lift overshoot is produced when the upper shear layer momentum is stronger than the lower shear layer momentum (or the balance favors the upper shear layer with respect to the steady case), whereas a lift undershoot is produced when the upper shear layer momentum is weaker than the lower shear layer momentum (or the balance favors the lower shear layer with respect to the steady case). Such findings help describe a modified Kutta condition that includes unsteady and viscous effects.

Chapter 7: Conclusions

This investigation is intended to validate Isaacs' surging airfoil theory experimentally and evaluate the key parameters in the unsteady motion, the flow physics, and the validity of the assumptions. The detailed work includes modifying the wind tunnel to generate a sinusoidally-oscillating freestream in the test section. This is achieved with an inverse design process assuming the flow in the test section is quasi-steady and inviscid. As a result, rotating elliptical choke-vanes downstream of the test section are designed. With the surging freestream condition, this modified unsteady wind tunnel was characterized with various intrusive measurements and analytical and numerical tools. It is found that the unsteady response of the facility is affected by the wave propagation caused by the fundamental forcing frequency, along with acoustic resonance and shedding of the choke vanes. As a result, the sinusoidal velocity amplitude is reduced and distorted as a function of the forcing frequency, whereas the phase delay along the test section increases. This significantly limits the available sinusoidal freestream in the test section. Thus, it is recommended that an elliptic sprocket or a feedback control on the motor speed is applied in the future to change the RPM of the rotating elliptic choke vanes to compensate the observed distortions in the freestream velocity. In addition, CFD can also be introduced in the choke vane

design process to investigate viscous effects. However, this can be costly for every target freestream condition.

After the characterization of the unsteady response of the wind tunnel, two NACA airfoils (0018 and 63-015A) were tested for incompressible surging flow conditions with a Reynolds number on the order of 10^6 . Surprisingly, the measured lift frequency response exhibit significant disagreement with Isaacs' surging airfoil theory in terms of the magnitudes and phases of the lift coefficient. While no apparent flow separation is observed on the airfoil, optical measurements of the flow field indicate a dynamic trailing-edge stagnation condition that oscillates with the freestream. This oscillatory trailing-edge stagnation condition clearly violates the classic Kutta condition assumed in the unsteady airfoil theory. Data suggest a strong viscous effects that cannot be neglected. A hypothesis is formulated that the balance of the local shear layer momentum dominates the state of the trailing-edge stagnation condition. Such oscillatory trailing-edge stagnation condition is approximated as a Gurney flap and the effect is shown to influence the resulting lift frequency response.

Finally, a near wake velocity survey with hot-film probes was conducted on the NACA 63-015A airfoil with different flow transition to prove the hypothesis of shear layer momentum balance at the trailing-edge. Results from the near wake survey support the oscillatory behavior of the trailing-edge stagnation condition. In addition, it proves a dynamic balance of the shear layer momentum near the trailing-edge that oscillates with the freestream and can be corroborated to the measured lift frequency response. This finding provides additional information to construct an unsteady viscous Kutta condition for surging airfoil.

Overall, the lift frequency response of a surging airfoil at high Reynolds number, low Mach number, moderate advance ratio, and low reduced frequency is experimentally investigated. The results do not validate Isaacs' theory and an unsteady Kutta condition due to viscous effects is found for the airfoil under surging freestream. However, there are still some discrepancies to analytically predict the lift frequency response as a result of the unsteady Kutta condition. A free vortex model [37] for surging airfoil may provide more insight of the development of the wake analytically and provide a more robust solution than a first approximation of the dynamic camber effect. In addition, it is unknown if the oscillatory trailing-edge stagnation condition still exists and dominates the lift frequency response at higher Mach number, advance ratio, or reduced frequency because of the narrow surging flow parameters in the current investigation. Future experimental work should include a greater range of advance ratio and reduced frequency without sacrificing the sinusoidal freestream condition. CFD simulations may help answer this question since such flow condition is challenging to produce in the laboratory. To compare with existing investigations and mask out the effects of dynamic pressure, unsteady load should always be normalized as shown in this work.

Appendix A: Location of the Surface Pressure Tap

Table A.1: Pressure tap location on NACA 0018.

Tap No.	x/c	y/c	Tap No.	x/c	y/c
1	0.0000	0.0000	26	0.6300	0.0646
2	0.0100	0.0255	27	0.6767	0.0583
3	0.0300	0.0426	28	0.7200	0.0520
4	0.0500	0.0533	29	0.7683	0.0445
5	0.0683	0.0607	30	0.8150	0.0368
6	0.0883	0.0671	31	0.8600	0.0290
7	0.1083	0.0722	32	0.9067	0.0205
8	0.1283	0.0765	33	0.0100	-0.0255
9	0.1483	0.0799	34	0.0300	-0.0426
10	0.1667	0.0825	35	0.0500	-0.0533
11	0.1867	0.0848	36	0.0683	-0.0607
12	0.2133	0.0871	37	0.0883	-0.0671
13	0.2400	0.0887	38	0.1083	-0.0722
14	0.2650	0.0896	39	0.1483	-0.0799
15	0.2917	0.0900	40	0.1867	-0.0848
16	0.3183	0.0899	41	0.2267	-0.0880
17	0.3450	0.0894	42	0.2650	-0.0896
18	0.3700	0.0885	43	0.3050	-0.0900
19	0.3967	0.0872	44	0.3450	-0.0894
20	0.4233	0.0856	45	0.4267	-0.0854
21	0.4500	0.0837	46	0.5083	-0.0786
22	0.4850	0.0808	47	0.5900	-0.0697
23	0.5217	0.0773	48	0.6817	-0.0576
24	0.5583	0.0734	49	0.7683	-0.0445
25	0.5933	0.0693	50	0.8550	-0.0297

Table A.2: Pressure tap location on NACA 63-015A.

Tap No.	x/c	y/c	Tap No.	x/c	y/c
1	0.0000	0.0000	30	0.0167	-0.0212
2	0.0167	0.0212	31	0.0368	-0.0312
3	0.0368	0.0312	32	0.0645	-0.0408
4	0.0645	0.0408	33	0.0931	-0.0484
5	0.0931	0.0484	34	0.1260	-0.0553
6	0.1260	0.0553	35	0.1610	-0.0611
7	0.1610	0.0611	36	0.1969	-0.0658
8	0.1969	0.0658	37	0.2335	-0.0696
9	0.2335	0.0696	38	0.2704	-0.0723
10	0.2704	0.0723	39	0.3072	-0.0741
11	0.3072	0.0741	40	0.3442	-0.0749
12	0.3442	0.0749	41	0.3813	-0.0748
13	0.3813	0.0748	42	0.4185	-0.0737
14	0.4185	0.0737	43	0.4560	-0.0718
15	0.4560	0.0718	44	0.4938	-0.0691
16	0.4938	0.0691	45	0.5191	-0.0669
17	0.5191	0.0669	46	0.5572	-0.0631
18	0.5572	0.0631	47	0.5954	-0.0588
19	0.5954	0.0588	48	0.6338	-0.0539
20	0.6338	0.0539	49	0.6725	-0.0486
21	0.6725	0.0486	50	0.7115	-0.0430
22	0.7115	0.0430	51	0.7508	-0.0372
23	0.7508	0.0372	52	0.7904	-0.0313
24	0.7904	0.0313	53	0.8300	-0.0255
25	0.8300	0.0255	54	0.8695	-0.0196
26	0.8695	0.0196	55	0.8956	-0.0158
27	0.8956	0.0158	56	0.9217	-0.0119
28	0.9217	0.0119	57	0.9473	-0.0081
29	0.9473	0.0081			

Appendix B: MATLAB Code for the Rotating Choke-Vane Design

```
% Wenbo's elliptical choke vane design for the 6" x 22"
% unsteady transonic wind tunnel
% constant rotation
% determines optimal choke vane geometry for a given ideal
% Mach history
% date of last revision 2018-12-26
% version 1.3
clear
close all
clc

%% desired Mach history
M_mean = 0.2;
sigma = 0.05/0.2;
omega_t = linspace(0,4*pi,360*4+1); % omega*t
M = M_mean*(1+sigma*sin(omega_t));
figure;
plot(omega_t*180/pi,M)
xlabel('\theta, degree')
ylabel('Mach_number')
xlim([0,360])
title('desired_Mach_number_history')

%% tunnel geometry
h_ts = 22; % test section height, in.
w_ts = 6; % test section width, in.
h_model = 0; % model height, neglected for now as a clean
% wind tunnel
h_b = 1.25; % brick height in the choke vane location
w_b = 6; % brick width in the choke vane location
```

```

N_b = 2; % no. of bricks
N_cv = 4; % no. of choke vane
A_ts = h_ts*w_ts; % test-section area;
A_throat = 26*6-h_b*w_b*N_b; % throat section minus brick
% area, keep choke vane area
w_cv = 5.73; % width of each choke vane, it needs some space
% to allow rotation
g = 1.4;
A_As = (((g+1)*0.5)^(-0.5*(g+1)/(g-1)))*((1+0.5*(g-1)*M.^2)
        .^(0.5*(g+1)/(g-1)))./M; % A/Astar

figure
plot(omega_t*180/pi,A_As)
xlabel('\theta, degree')
ylabel('A/A^{*}')
title('A/A^{*} from desired Mach number history')
As = A_ts./A_As; % required throat area for Mach history
figure
plot(omega_t*180/pi,As)
xlabel('\theta, degree')
ylabel('A^{*}, inch^{2}')
title('A^{*} from desired Mach number history')
A_cv = (A_throat-As)/N_cv; % area of each choke vane
% projected area corresponding to Mach history
h_cv = A_cv/w_cv; % remember this height is TWO TIMES of
% the length on half choke vane piece
figure
plot(omega_t*180/pi,h_cv)
xlabel('\theta, degree')
ylabel('choke vane projected height, inch')
title('choke vane projected height from desired Mach
        number history')
a_cv = max(h_cv)/2; % choke vane semi-major axis
b_cv = min(h_cv)/2; % choke vane semi-minor axis

%% elliptical choke vane optimization
a = a_cv;
b = b_cv;
x = linspace(-a,a,720+1);
m = 2;
n = 2;
yp = (1-abs((x/a)).^m).^(1/n)*b;
yn = -(1-abs((x/a)).^m).^(1/n)*b;
x1 = [x -x]; % allocate x

```

```

y1 = [yp yn]; % allocate y

% no-optimization
figure
plot(x1,y1);
axis image
legend('m=_2, _n=_2')
xlabel('x, _inch')
ylabel('y, _inch')
title('pre-optimized _choke _vane _geometry')

% opitimization
figure
plot(omega_t*180/pi, circshift(M,-90*2))
xlabel('\theta, _degree')
ylabel('Mach _number')
title('desired _Mach _number _history _starting _at
      _ _ _ _ _ minimum _blockage')
figure
plot(omega_t*180/pi, circshift(h_cv,-90*2))
xlabel('\theta, _degree')
ylabel('h _cv, _inch')
title('desired _choke _vane _projected _height _starting
      _ _ _ _ _ at _minimum _blockage')
h_cv2 = circshift(h_cv,-90*2);
j = 1;
for m = 1.8:0.004:2.2 % or use 0.002 for 2.5 mins calculation
    for n = 1.8:0.004:2.2
        i = 1;
        x = linspace(-a,a,720+1);
        yp = (1-abs((x/a)).^m).^(1/n)*b;
        yn = -(1-abs((x/a)).^m).^(1/n)*b;
        x1 = [x -x]; % allocate x
        y1 = [yp yn]; % allocate y
        for alpha = 0:0.25:180 % cv rotate half circle is
            % one Mach wave
            x2 = x1*cosd(alpha)-y1*sind(alpha);
            y2 = x1*sind(alpha)+y1*cosd(alpha);
            hp = max(y2);
            h(i) = 2*hp;
            erms1(i) = h(i)-h_cv2(i);
            i = i+1;
        end
    end
end

```

```

        erms2(j) = sqrt(sum(erms1.^2)); % rms error
        m2(j) = m;
        n2(j) = n;
        j = j+1;
    end
end

%% find optimized m,n
[val,idx] = min(erms2);
m_op = m2(idx); % optimal m
n_op = n2(idx); % optimal n

%% plot optimized choke vane
x = linspace(-a,a,720+1);
yp = (1-abs((x/a)).^m_op).^(1/n_op)*b;
yn = -(1-abs((x/a)).^m_op).^(1/n_op)*b;
x_op = [x -x]; % allocate x
y_op = [yp yn]; % allocate y
figure
plot(x_op,y_op,'linewidth',2)
axis image
legend(['{\it m}_= ', num2str(m_op), ', ',
        '{\it n}_= ', num2str(n_op)])
legend boxoff
xlabel('{\it x}_dimension, inch')
ylabel('{\it y}_dimension, inch')
grid on
xlim([-3 3])
ylim([-2 2])
set(gca, 'fontsize',10, 'LineWidth',2)
% title('optimized elliptical choke vane')

%% plot Mach history error
omega_t = linspace(0,4*pi*2,360*8+1); % omega*t
M = M_mean*(1+sigma*sin(omega_t));
M = circshift(M,-90*2);

% Mach history from choke vane rotation
i = 1;
for alpha = 0:0.25:360*2
    x3 = x_op*cosd(alpha)-y_op*sind(alpha);
    y3 = x_op*sind(alpha)+y_op*cosd(alpha);
    h_op(i) = 2*max(y3);

```

```

As_op = 26*6-h_b*w_b*N_b-N_cv*h_op(i)*w_cv;
A_As_op = A_ts/As_op;
M_guess = M_mean;
f = @(M) ( (((g+1)/2)^(-(g+1)/(2*g-2)))*((1+(g-1)*M^2/2)
          ^((g+1)/(2*g-2)))/M - A_As_op );
Ma(i) = fzero(f,M_guess);
i = i+1;
end
M_cv = Ma; % Mach number from choke vane geometry
figure
plot(h_cv2)
hold on
plot(h_op)
ylabel('h')
title('compare_h')

% percentage error
pe = (M_cv-M)./M*100;
lw = 1.5;
figure
plot(omega_t*180/pi,M,'Linewidth',1.5)
hold on
plot(omega_t*180/pi,M_cv,'—','Linewidth',1.5)
hold on
yyaxis left
xlabel('choke_vane_phase_angle\theta, degree')
ylabel('Mach_number_{it M}')
ylim([0.12 0.28])
yyaxis right
plot(omega_t*180/pi,pe,':','Linewidth',1.5)
legend('Mach_number:target','Mach_number:choke_vane',
        '%_error','location','southeast')
legend boxoff
ylabel('%_error')
ylim([-0.3 0.3])
xlim([0,720])
xticks([0 180 360 540 720])
xticklabels({'0','90','180','270','360'})
set(gca,'fontsize',10,'LineWidth',lw)

```

Appendix C: Characterization of the Frequency Response of the Pitot-Static Probe in an Acoustic Resonance Cavity

Here describes the details of an benchtop experiment to characterize the frequency response of the pitot-static probe using an acoustic box. This acoustic box had been used to characterize the dynamic response of a pressure sensitive paint formulation previously [76]. In this experiment, the box was slightly modified to help characterize the dynamic response of the pitot-static probe. The schematic of the experiment is shown in Figure C.1. The box was sealed during the experiment where a speaker (AS100N) was attached to the sidewall and was configured to generate sound waves into the box. The speaker was driven by a function generator (Agilent 33220A) which the signal was amplified by a dual-channel amplifier (Crown XLS1000). In addition, a Kulite pressure transducer powered by a signal conditioner with a gain of 100 was installed on the other side and the signal was fed to a computer. Right next to the pressure transducer, the pitot-static probe was also installed. Thus, both the pressure transducer and the pitot-static probe were facing the speaker at the same location. The pneumatic tubing behind the pitot-static probe had the same dimension (diameter and length) between the acoustic box experiment and the wind tunnel experiment. The full dimension of the pressure channel is listed in Table C.1 where the channel is divided into three sections starting from the probe tip to

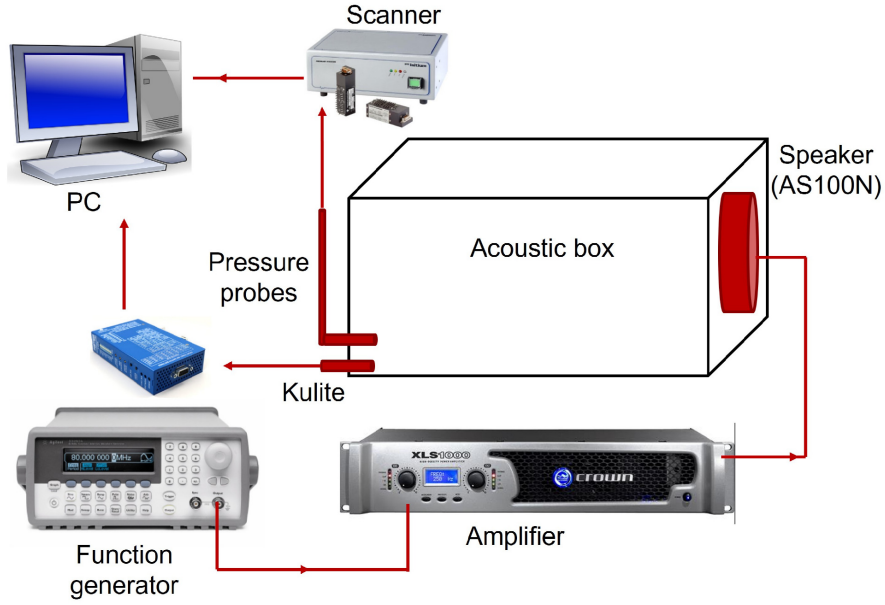


Figure C.1: Schematic of the acoustic box experiment.

the pressure scanner. The length, inner diameter, and volume are based on the definition in reference [63]. During this experiment, various sine waves with different amplitudes and frequency were generated by the function generator, and the signals from the pressure transducer and the pitot-static probe were compared to generate the transfer function in Figure 3.12. This transfer function was compared with the B-T model which showed reasonable agreement. As a result, the B-T model was proved to be a useful tool on the dynamic compensation of the pressure measuring system. Finally, the dynamic response of the pressure measuring system during the wind tunnel experimental condition was examined. This transfer function is shown in Figure C.2, where the amplitude and phase delay are similar to Figure 3.12.

Table C.1: Dimension of the pitot-static probe and the pneumatic line.

Pressure Channel	L_1 , inch	D_1 , inch	V_1 , inch ³	L_2 , inch	D_2 , inch	V_2 , inch ³	L_3 , inch	D_3 , inch	V_3 , inch ³
P_t	10	0.065	0	2	0.03	0	31	0.054	0.000809
P_s	10	0.12	0	1	0.03	0	28	0.054	0.000809

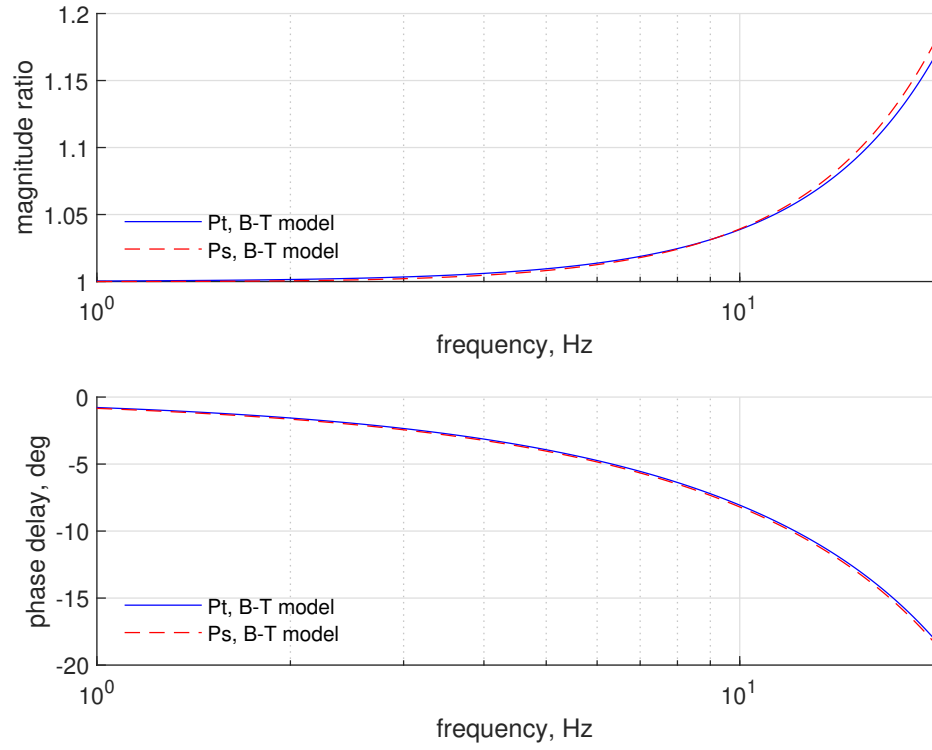


Figure C.2: Transfer function of the pressure measuring system predicted by the B-T model under the tunnel operating condition, $M = 0.2$, $Re_c = 1,500,000$.

Appendix D: Analytic Model for the Unsteady Response of the Wind Tunnel

An analytical model is derived to predict the unsteady tunnel response of this wind tunnel, similar to Reference [11]. The method utilizes the conservation of momentum.

$$\frac{d}{dt} \oint_{\mathcal{V}} \rho \vec{u} d\mathcal{V} + \oint_A \rho \vec{u} (\vec{u} \cdot \hat{n}) dA = - \oint_A P d\vec{S} + \vec{F}_{viscous} \quad (\text{D.1})$$

The mass flow rate inside the wind tunnel is assumed to be constant throughout the tunnel but can vary with time

$$\dot{m} = \rho(x, t) u(x, t) A(x, t) = \rho_p(t) u_p(t) A_p = \rho_{ts}(t) u_{ts}(t) A_{ts} = \rho_e(t) u_e(t) A_e(t) \quad (\text{D.2})$$

where lower case p refers to plenum, lower case ts refers to test section, and lower case e refers to the tunnel throat area.

As a result, the first term on the left hand side of the momentum equation C.1 is substituted as

$$\frac{d}{dt} \oint_{\mathcal{V}} \rho \vec{u} d\mathcal{V} = \frac{d}{dt} \int_p^e \rho(x, t) u(x, t) A(x, t) dx \quad (\text{D.3})$$

the second term on the left hand side of the momentum equation can be rewritten as

$$\oint_A \rho \vec{u} (\vec{u} \cdot \hat{n}) dA = -\rho_p u_p^2 A_p + \rho_e u_e^2 A_e = \rho_{ts}^2 u_{ts}^2 A_{ts}^2 \left(\frac{1}{\rho_e A_e} - \frac{1}{\rho_p A_p} \right) \quad (\text{D.4})$$

and the first term on the right hand side of the momentum equation is reduced to

$$- \oint_A P d\vec{S} = P_p A_p - P_e A_e \quad (\text{D.5})$$

Because the mass flow rate is assumed to be constant along the tunnel at every instant in time, Equation C.3 can be derived in terms of any sections along the tunnel, such as from the plenum to the exit or from the test section to the exit

$$\frac{d}{dt} \int_p^e \rho(x, t) u(x, t) A(x, t) dx = \frac{d}{dt} \int_p^e \rho_{ts}(t) u_{ts}(t) A_{ts} dx = A_{ts} L_{pe} \frac{d}{dt} [\rho_{ts}(t) u_{ts}(t)] \quad (D.6)$$

After substitution, the momentum equation becomes

$$A_{ts} L_{pe} \frac{d}{dt} [\rho_{ts}(t) u_{ts}(t)] + \rho_{ts}^2 u_{ts}^2 A_{ts}^2 \left(\frac{1}{\rho_e A_e} - \frac{1}{\rho_p A_p} \right) = P_p A_p - P_e A_e + \vec{F}_{viscous} \quad (D.7)$$

Rearranging terms, Equation C.7 becomes

$$\frac{d(u_{ts}(t))}{dt} + \frac{\rho u_{ts}^2 A_{ts}}{L_{pe}} \left(\frac{1}{A_e} - \frac{1}{A_p} \right) = \frac{P_p A_p - P_e A_e}{\rho A_{ts} L_{pe}} + viscous\ loss\ term \quad (D.8)$$

With a small perturbation assumption, the flow velocity, pressure, and area change at the exit (*i.e.*, the choke-vane) are decomposed into the steady and unsteady components, namely

$$u_{ts} = \bar{u}_{ts} + \tilde{u}_{ts}, A_e = \bar{A}_e + \tilde{A}_e, P_p = \bar{P}_p + \tilde{P}_p, P_e = \bar{P}_e + \tilde{P}_e, \quad (D.9)$$

In the steady case, Equation C.8 reduces to

$$\frac{\rho \bar{u}_{ts}^2 A_{ts}}{L_{pe}} \left(1 - \frac{\bar{A}_e}{A_p} \right) = \frac{\bar{A}_e (\bar{P}_p A_p - \bar{P}_e \bar{A}_e)}{\rho A_{ts} L_{pe}} + viscous\ loss\ term \quad (D.10)$$

whereas in the unsteady case it reduces to

$$\begin{aligned} & (\bar{A}_e + \tilde{A}_e) \frac{d}{dt} (\bar{u}_{ts} + \tilde{u}_{ts}) + \frac{(\bar{u}_{ts} + \tilde{u}_{ts})^2 A_{ts}}{L_{pe}} \left(1 - \frac{\bar{A}_e + \tilde{A}_e}{A_p} \right) = \\ & \frac{(\bar{A}_e + \tilde{A}_e) [(\bar{P}_p + \tilde{P}_p) A_p - (\bar{P}_e + \tilde{P}_e) (\bar{A}_e + \tilde{A}_e)]}{\rho A_{ts} L_{pe}} + viscous\ loss\ term \end{aligned}$$

Rearranging terms in Equation C.10, it becomes

$$\frac{d}{dt} (\tilde{u}_{ts}) + \frac{2 A_{ts}}{L_{pe} \bar{A}_e} \frac{\tilde{u}_{ts}}{\bar{u}_{ts}} \left(1 - \frac{\bar{A}_e}{A_p} \right) = \frac{1}{\rho A_{ts} L_{pe} \bar{u}_{ts}^2} [\dots] + viscous\ loss\ term \quad (D.11)$$

where the first term on the right hand side is lengthy and is neglected for brevity.

Assuming a new variable $\varepsilon(t) = \frac{\tilde{u}_{ts}}{\bar{u}_{ts}}$, Equation C.11 is rewritten as

$$\frac{1}{\bar{u}_{ts}} \frac{d\varepsilon}{dt} + \frac{2A_{ts}}{L_{pe}\bar{A}_e} \left(1 - \frac{\bar{A}_e}{A_p}\right) \varepsilon = [...] + \text{viscous loss term} \quad (\text{D.12})$$

which is in the form of a forced first-order linear equation

$$\tau \frac{d\varepsilon(t)}{dt} + \varepsilon(t) = g(t) \quad (\text{D.13})$$

where

$$\tau = \frac{L_{pe}\bar{A}_e A_p}{2\bar{u}_{ts} A_{ts}} \frac{1}{A_p - \bar{A}_e} \quad (\text{D.14})$$

is the time constant of the system

The solution for this first-order linear equation is

$$\varepsilon(t) = B(\omega) \sin(\omega t + \varphi) = \frac{G_{max}}{\sqrt{1 + (\omega\tau)^2}} \sin(\omega t + \varphi) \quad (\text{D.15})$$

where the amplitude ratio of the unsteady response is

$$M(\omega) = \frac{B(\omega)}{G_{max}} = \frac{1}{\sqrt{1 + (\omega\tau)^2}} \quad (\text{D.16})$$

and the time constant becomes

$$\tau = \frac{L_{ce}\bar{A}_e}{2\bar{u}_{ts}(A_{ts} - \bar{A}_e)} \quad (\text{D.17})$$

and

$$\varphi = \frac{\omega L}{a_{ts} - u_{ts}} \quad (\text{D.18})$$

Bibliography

- [1] John Anderson. *Principles of Helicopter Aerodynamics*. Cambridge University Press, Second edition, 2006.
- [2] T. Theodorsen. “General Theory of Aerodynamic Instability and the Mechanism of Flutter”. *Tech. Rep. 496, NACA Technical Report*, 1935.
- [3] B.G. van der Wall and L.G. Leishman. “On the Influence of Time-Varying Flow Velocity on Unsteady Aerodynamics”. *Journal of American Helicopter Society*, 39(4):25–36, 1994.
- [4] R. Isaacs. “Airfoil Theory for Flows of Variable Velocity”. *Journal of Aeronautical Sciences*, 12(1):113–117, 1945.
- [5] R. Isaacs. “Airfoil Theory for Rotary Wing Aircraft”. *Journal of Aeronautical Sciences*, 13(4):218–220, 1946.
- [6] J.M. Greenberg. “Airfoil in Sinusoidal Motion in a Pulsating Stream”. *Tech. Rep. 1326, NACA Technical Note*, 1947.
- [7] C. Strangfeld, H. Mueller-Vahl, C.N. Nayeri, C.O. Paschereit, and D. Greenblatt. “Airfoil in a High Amplitude Oscillating Stream”. *Journal of Fluid Mechanics*, 793:79–108, 2016.
- [8] John Anderson. *Fundamentals of Aerodynamics*. McGraw-Hill, Fifth edition, 2009.
- [9] L.S. Saxena, A.A. Fejer, and M.V. Morkovin. “Features of Unsteady Flow over Airfoils”. *AGARD conference proceedings*, 1977.
- [10] K. Granlund, B. Monnier, M. Ol, and D. Williams. “Airfoil Longitudinal Gust Response in Separated vs. Attached Flows”. *Phys. Fluids*, 26:1–14, 2014.
- [11] D. Greenblatt. “Unsteady Low-Speed Wind Tunnel”. *AIAA Journal*, 54(6):1817–1830, June 2016.

- [12] J. Farnsworth, D. Sinner, D. Gloutak, L. Droste, and D. Bateman. “Design and qualification of an unsteady low-speed wind tunnel with upstream louver system”. *Exp. Fluids*, 61, 2020.
- [13] H. Babinsky and R.M. Fernie. “NACA0012 Aerofoil in an Oscillating Transonic Free-Stream”. *AIAA Paper 2002-0115*, January 2002.
- [14] K. Gompertz, C. Jensen, P. Kumar, D. Peng, J.W. Gregory, and J.P. Bons. “Modification of Transonic Blowdown Wind Tunnel to Produce Oscillating Freestream Mach Number”. *AIAA Journal*, 49(11):2555–2563, November 2011.
- [15] K.O. Granlund, M.V. Ol, and A.R. Jones. “Streamwise Oscillation of Airfoils into Reverse Flow”. *AIAA Journal*, 54(5):1628–1636, January 2016.
- [16] M.P. Jentzsch, H. Schmidt, R. Wosidl, C. Nayeri, and C.O. Paschereit. “Development of a Setup and Measurement Procedure for Unsteady Model Velocities in a Large Water Towing Tank”. *AIAA Scitech Forum*, January 2019.
- [17] W. Zhu, B. Harter, J.W. Gregory, and J.P. Bons. “Characterizing Wave Propagation in an Unsteady Transonic Wind Tunnel”. *Aerodynamic Measurement Technology and Ground Testing Conference*, June 2018.
- [18] A.A. Fejer. “Visual Study of Oscillating Flow over a Stationary Airfoil”. *Proceedings of the SQUID Workshop*, 1977.
- [19] J.M.R. Graham. “Unsteady Flow Over Aerofoils with Separation”. *NA*, 1982.
- [20] E. Krause and W. Schweitzer. “The effect of an oscillatory freestream-flow on a NACA-4412 profile at large relative amplitudes and low Reynolds-numbers”. *Experiments in Fluids*, 9:159–166, 1990.
- [21] J. Choi, T. Colonius, and D.R. Williams. “Surging and Plunging Oscillations of an Airfoil at Low Reynolds Number”. *Journal of Fluid Mechanics*, 763:237–253, 2015.
- [22] D.A. Gloutak and A. Farnsworth. “Characteristic Wing Measurements of a NACA 0015 in Steady and Unsteady Surging Wind Tunnel Flow”. *AIAA Scitech Forum*, January 2020.
- [23] G.A. Pierce, D.L. Kunz, and J.B. Malone. “The Effect of Varying Freestream Velocity on Airfoil Dynamic Stall Characteristics”. *Journal of the American Helicopter Society*, 23(2):27–33, 1978.
- [24] D. Favier, J. Rebont, and C. Maresca. “Large-Amplitude Fluctuations of Velocity and Incidence of an Oscillating Airfoil”. *AIAA Journal*, 17(11):1265–1267, 1979.

- [25] C.A. Maresca, D.J. Favier, and J.M. Rebont. “Unsteady Aerodynamics of an Aerofoil at High Angle of Incidence Performing Various Linear Oscillations in a Uniform Stream”. *Journal of the American Helicopter Society*, 26(2):40–45, 1981.
- [26] K. Hird, M.W. Frankhouser, J.W. Gregory, and J.P. Bons. “Compressible Dynamic Stall of an SSC-A09 Airfoil Subjected to Coupled Pitch and Freestream Mach Oscillations”. *Proceedings of the 70th AHS Forum*, 2014.
- [27] K. Hird, M.W. Frankhouser, S. Naigle, J.W. Gregory, and J.P. Bons. “Study of an SSC-A09 Airfoil in Compressible Dynamic Stall with Freestream Mach Oscillations”. *Proceedings of the 71st AHS Forum*, 2015.
- [28] S.C. Naigle, M.W. Frankhouser, J.W. Gregory, and J.P. Bons. “Effects of Time-Varying Flow Velocity on Steady Blowing Flow Control for a Pitching Airfoil”. *Proceedings of the 72nd AHS Forum - Vertical Flight Society*, pages 575–586, 2016.
- [29] A.I. Jose, J.G. Leishman, and J.D. Baeder. “Unsteady Aerodynamic Modeling with Time-Varying Free-Stream Mach Numbers”. *Journal of the American Helicopter Society*, 51(4):299–318, 2006.
- [30] C. Strangfeld, C.L. Rumsey, H. Mueller-Vahl, D. Greenblatt, C. Nayeri, and C.O. Paschereit. “Unsteady Thick Airfoil Aerodynamics: Experiments, Computation, and Theory”. *45th AIAA Fluid Dynamics Conference*, 2015.
- [31] Q. Wang and Q. Zhao. “Unsteady Aerodynamic Characteristics Investigation of Rotor Airfoil under Variational Freestream Velocity”. *Aerospace Science and Technology*, 58:82–91, 2016.
- [32] A. Kocher, R. Cummings, S.A. Tran, and O. Sahni. “Large Eddy Simulation of Surging Airfoils with Moderate to Large Streamwise Oscillations”. *55th AIAA Aerospace Sciences Meeting*, 2017.
- [33] M. Drela. Xfoil: An analysis and design system for low reynolds number airfoils. In T.J. Mueller, editor, *Low Reynolds Number Aerodynamics*. Springer, Berlin, Heidelberg, pp 1-12, 1989.
- [34] B.C. Basu and G.J. Hancock. “The unsteady motion of a two-dimensional aerofoil in incompressible inviscid flow”. *Journal of Fluid Mechanics*, 87:159–178, 1978.
- [35] D.G. Crighton. “The Kutta condition in unsteady flow”. *Annual Review of Fluid Mechanics*, 17:411–445, 1985.

- [36] D.R. Poling and D.P. Telionis. “The response of airfoils to periodic disturbances - the unsteady Kutta condition”. *AIAA Journal*, 24:193–199, 1986.
- [37] X. Xia and K. Mohseni. “Unsteady aerodynamics and vortex-sheet formation of a two-dimensional airfoil”. *J. Fluid Mech.*, 830:439–478, 2017.
- [38] H. Taha and A.S. Razaei. “Viscous extension of potential-flow unsteady aerodynamics: the lift frequency response problem”. *Journal of Fluid Mechanics*, 868:141–175, 2019.
- [39] J.P. Giesing. “Vorticity and Kutta condition for unsteady multienergy flows”. *Trans. ASME J. Appl. Mech.*, 36:608–613, 1969.
- [40] A.F. Messiter. “Boundary-layer flow near the trailing edge of a flat plate”. *SIAM J. Appl. Maths*, 18(1):241–257, 1970.
- [41] W.J. McCroskey and J.J. Philippe. “Unsteady viscous flow on oscillating airfoils”. *AIAA Journal*, 13:71–79, 1975.
- [42] B. Satyanarayana and S. Davis. “Experimental studies of unsteady trailing-edge conditions”. *AIAA Journal*, 16:125–129, 1978.
- [43] X.L. Liu, A. Wo, and E.E. Covert. “Unsteady streamlines near the trailing edge of a NACA 0012 airfoil at Reynolds number of 125,000”. *AIAA Journal*, 28:169–170, 1990.
- [44] G.L. Commerford and F.O. Carta. “Unsteady aerodynamic response of a two-dimensional airfoil at high reduced frequency”. *AIAA Journal*, 12:43–48, 1974.
- [45] F.S. Archibald. “Unsteady Kutta condition at high values of the reduced frequency parameter”. *AIAA Journal*, 12:545–550, 1975.
- [46] R.A. Kadlec and S.S. Davis. “Visualization of quasiperiodic flows”. *AIAA Journal*, 17:1164–1169, 1979.
- [47] S. Fleeter. “Trailing edge conditions for unsteady flows at high reduced frequency”. *AIAA Journal*, 18:497–503, 1980.
- [48] C.M. Ho and S.H. Chen. “Unsteady Kutta condition of a plunging airfoil”. *Unsteady Turbulent Shear Flows*, pages 197–206, 1981.
- [49] S.H. Chen and C.M. Ho. “Near wake of an unsteady symmetric airfoil”. *Journal of Fluid Structure*, 1:151–164, 1987.
- [50] E.A. Lurie, D.P. Keenan, and J.E. Kerwin. “Experimental study of an unsteady separating boundary layer”. *AIAA Journal*, 36:565–570, 1998.

- [51] W.J. McCroskey. “Unsteady airfoils”. *Annu. Rev. Fluid Mech.*, 14:285–311, 1982.
- [52] W. Zhu, M. McCrink, J. Bons, and J. Gregory. “The unsteady Kutta condition on an airfoil in a surging flow”. *J. Fluid Mech.*, 893, 2020.
- [53] M. Raffel, H. Richard, and G. Meier. On the applicability of background oriented optical tomography for large scale aerodynamic investigations. *Exp Fluids*, 28:477–481, 2000.
- [54] M. Raffel. Background-oriented schlieren (bos) techniques. *Exp Fluids*, 56(60), 2015.
- [55] H. Richard and M. Raffel. Principle and applications of the background oriented schlieren (bos) method. *Meas. Sci. Technol.*, 12(9):1576–1585, 2001.
- [56] K. Gompertz, J. Bons, and J. Gregory. Passive trip evaluation using infrared thermography. In *50th AIAA Aerospace Sciences Meeting including the New Horizons Forum and Aerospace Exposition*, 2012.
- [57] E.M. Winkler. “Stagnation Temperature Probes for Use at High Supersonic Speeds and Elevated Temperatures”. *NAVORD Report 3834*, October 1954.
- [58] Stavros Tavoularis. *Measurement in Fluid Mechanics*. Cambridge University Press, First edition, 2005.
- [59] D. Collis and M. Williams. “Two-dimensional convection from heated wires at low Reynolds numbers”. *Journal of Fluid Mechanics*, 6(3):357–384, March 1959.
- [60] Frank White. *Viscous Fluid Flow*. McGraw-Hill, Third edition, 2006.
- [61] L.V. King. “On the convection of heat from small cylinders in a stream of fluid: Determination of convection constants of small platinum wires with application to hot-wire anemometry”. *Phil. Trans.*, A(214):373–432, September 1914.
- [62] John Anderson. *Computational Fluid Dynamics: The Basics with Applications*. McGraw-Hill, First edition, 1995.
- [63] H. Bergh and H. Tijdeman. “Theoretical and experimental results for the dynamic response of pressure measuring systems”. *Report NLR-TR F.238*, January 1965.
- [64] H.H. Fernholtz and J.M.R. Graham. “A Wind Tunnel for Unsteady Turbulent Shear Flows: Design and Calculation”. *Zeitschrift für Flugwissenschaften und Weltraumforschung*, 6:408–416, 1982.

- [65] P. Baj, P.J.K. Bruce, and O.R.H. Buxton. “The triple decomposition of a fluctuating velocity field in a multiscale flow”. *Physics of Fluids*, 27(7), June 2015.
- [66] T. von Karman. The analogy between fluid friction and heat transfer. *ASME Trans.*, 61:705–710, 1939.
- [67] B.A. Kader. Temperature and concentration profiles in fully turbulent boundary layers. *Int. J. Heat Mass Transfer*, 24(9):1541–1544, 1981.
- [68] Frank White. *Fluid Mechanics*. McGraw-Hill, Seventh edition, 2009.
- [69] Ira H. Abbott and Albert E. von Doenhoff. *Theory of Wing Sections*. Dover, 1981.
- [70] R.H. Liebeck. “Design of Subsonic Airfoils for High Lift”. *J. Aircraft*, 15(9):547–561, 1978.
- [71] J.C. Gibbings. “On Boundary-Layer Transition Wires”. *Aeronautical Research Council, CP-462*, 1959.
- [72] I. Tani, M. Iuchi, and K. Yamamoto. “Further Experiments on the Effect of a Single Roughness Element on Boundary-Layer Transition”. *Rep. Inst. Sci. Tech., Tokyo Univ.*, 8, 1953.
- [73] B. Satyanarayana. “Unsteady Wake Measurements of Airfoils and Cascades”. *AIAA Journal*, 15, 1977.
- [74] C. Hah and B. Lakshminarayana. “Measurement and prediction of mean velocity and turbulence structure in the near wake of an airfoil”. *J. Fluid Mech.*, 115:251–282, 1982.
- [75] Stephen Pope. *Turbulent Flows*. Cambridge University Press, First edition, 2000.
- [76] M.Y. Ali, A. Pandey, and J.W. Gregory. “Dynamic Mode Decomposition of Fast Pressure Sensitive Paint Data”. *Sensors*, 16(2), 2016.