

CRANFIELD UNIVERSITY

L. S. ROBERTS

BOUNDARY-LAYER TRANSITION ON WINGS IN
GROUND EFFECT

CENTRE FOR DEFENCE ENGINEERING
Aeromechanical Systems Group

PhD

Academic Year: 2016–2017

Supervisors:

Dr. M. V. Finnis

Prof. K. Knowles

March 2017

CRANFIELD UNIVERSITY

CENTRE FOR DEFENCE ENGINEERING
Aeromechanical Systems Group

PhD

Academic Year: 2016–2017

L. S. ROBERTS

Boundary-Layer Transition on Wings in Ground Effect

Supervisors:

Dr. M. V. Finnis

Prof. K. Knowles

March 2017

© Cranfield University 2017. All rights reserved. No part of this publication may be reproduced without the written permission of the copyright owner.

Abstract

The competitiveness of a high-performance racing car is extremely reliant on aerodynamics. Due to the current economic climate, track testing is often forsaken and the majority of aerodynamic development carried out using sub-scale wind tunnel testing and computational simulations. It is important, therefore, that experimental and computational approaches represent real-world conditions as closely as possible.

Although racing cars travel at much higher speeds than typical passenger cars, in comparison to aircrafts they still operate at relatively low Reynolds numbers and, consequently, laminar and transitional phenomena are evident. Despite this, the bulk of relevant literature available for racing-car aerodynamics is undertaken with little regard to the influence of Reynolds number, and in the case of computational studies, the omission of laminar and transitional phenomena all together.

The present work has demonstrated, using a super-scale two-dimensional wind-tunnel model, that laminar and transition flow phenomenon are important at Reynolds numbers equivalent to a full-scale racing car. Moreover, the influence of these aspects increased as the wing's ground clearance reduced; meaning that in ground effect they are even more important. Further experiments with three-dimensional models of varying complexity, from a simple single-element wing to a highly complex F1-specification wing, showed that laminar phenomena are important for F1 applications as well as for lower-downforce capability racing cars.

A transition-sensitive eddy-viscosity turbulence model, $k - k_L - \omega$, was used to simulate inverted wings operating in ground effect. It was shown that that laminar and transitional flow states could be simulated easily inside a commercial solver, and that the model offered a substantial improvement over the classical fully-turbulent $k - \omega$ SST in terms of both force coefficient prediction and surface-flow structures.

This experiments and computational simulations described in this thesis show the Reynolds-number sensitivity of, and importance of laminar phenomenon on, wings operating in ground effect. It has been shown that laminar boundary layers are an important aspect of the flow characteristics of wings in ground effect, at both full-scale and model-scale Reynolds numbers. As such, it is recommended that future studies incorporate laminar and transitional phenomena.

Keywords

Turbulence; Laminar; CFD; Reynolds number; Separation bubble; Aerofoil; Racing car.

Contents

Abstract	v
Table of Contents	vii
List of Figures	xiii
List of Tables	xxv
Nomenclature	xxvii
Acknowledgements	xxxiii
1 Introduction	1
1.1 The Aerodynamic Requirement	2
1.2 The Front Wing	3
1.3 Aerodynamic Development	6
1.4 Aims & Objectives	7
2 Literature Review	11
2.1 Laminar-Turbulent Transition	11
2.1.1 The Reynolds Number	12
2.1.2 Natural Transition	14

2.1.3	Separation-Induced Transition	16
2.1.4	Bypass Transition	19
2.1.5	Relaminarisation	20
2.1.6	Influential Parameters	20
2.1.7	A Transition-Sensitive Eddy Viscosity Closure Model	26
2.2	Transition on Aerofoils	30
2.2.1	Use of the $k - k_L - \omega$ Closure Model	34
2.3	Wings in Ground Effect	37
2.3.1	Computational Methods	47
2.4	Summary	50
3	Description of Experiments	57
3.1	Test Facility	57
3.2	Test Conditions	59
3.3	Test Models	60
3.3.1	Two-Dimensional Wing	61
3.3.2	Three-Dimensional Wing Configurations	68
3.4	Experimental Methods & Uncertainties	78
3.4.1	Force Acquisition	78
3.4.2	Hot-Film Anemometry	79
3.4.3	Laser Doppler Anemometry	81
3.4.4	Static Pressure Measurements	86
3.4.5	Surface-Flow Visualisation	88
4	Description of Computations	101
4.1	Two-Dimensional Euler Simulation	101
4.1.1	Overview of MSES	101

4.1.2	Grid Generation	103
4.1.3	Solving	103
4.2	Three-Dimensional RANS	104
4.2.1	Facilities & Software	106
4.2.2	Geometry Preparation	106
4.2.3	Grid Generation	106
4.2.4	Boundary Conditions	108
4.2.5	Numerical Method	109
5	Results - Two-Dimensional Wing in Free Transition	113
5.1	Experimental	114
5.1.1	Static Pressure Distribution	114
5.1.2	Surface-Flow Visualisation	118
5.1.3	Hot-Film Anemometry	120
5.1.4	LDA Velocity Profiles	128
5.2	Euler Simulations - MSES	131
5.2.1	Influence of Reynolds number at Zero Incidence	131
5.2.2	Effect of Incidence	135
5.2.3	Effect of Ground Proximity	137
5.2.4	Effect of Freestream Turbulence	138
5.2.5	Geometry Dependency	140
5.3	Summary & Discussion	142
6	Results - Three-Dimensional Wings in Free Transition	163
6.1	Single-Element Wing (CSW-SE)	163
6.1.1	Force Measurements	163
6.1.2	Surface-Flow Visualisation	166

6.1.3	RANS Simulations	170
6.2	Multi-Element Wing (CSW-ME)	180
6.2.1	Force Measurements	180
6.2.2	Surface-Flow Visualisation	182
6.2.3	RANS Simulations	187
6.3	F1 Wing	194
6.3.1	Force Measurements	194
6.3.2	Surface-Flow Visualisation	196
6.4	Summary & Discussion	198
7	Results - Forced Transition	223
7.1	Two-Dimensional Wing	224
7.1.1	Pressure Measurements	225
7.1.2	Surface-Flow Visualisation	227
7.1.3	Hot-Film Anemometry	228
7.1.4	LDA Velocity Profiles	230
7.2	Three-Dimensional Wings	231
7.2.1	Roughness Trip	231
7.2.2	Serrated Trip	240
7.3	Summary & Discussion	245
8	Conclusions	269
8.1	Characteristics of Transition at Varying Reynolds Number, Ground Clearance and Incidence	270
8.2	Two- and Three-Dimensional Effects	272
8.3	Numerical Modelling of Transition	274
8.4	The Influence of Laminar and Turbulent Boundary-Layer Extent	276

8.5	Importance of Transition for Wings in Ground Effect	280
9	Recommendations for Future Work	283
9.1	Freestream turbulence intensity	283
9.2	Further Measurements on Three-Dimensional Wings	283
9.3	High Resolution Hot-Film Measurements	284
9.4	LES Simulation	284
A	Appendices	287
A.1	Force Coefficient Uncertainty	287
A.2	Evaluation of Appropriate Inlet Conditions for $k - k_L - \omega$	289
A.3	CFD Parameters	295
	References	297

List of Figures

1.1	Scuderia Toro Rosso STR11 Formula One Car front wing	9
2.1	Schematic of natural transition over a flat plate [101]	52
2.2	Curves of neutral stability for two-dimensional incompressible perturbations in a) plane boundary layer, and b) boundary layer under various pressure gradients	52
2.3	Oscillograms of hot-wire anemometer signals showing fluctuations in the boundary layer of a flat plate at set distances from the leading edge (in metres) for a fixed freestream velocity [5]	53
2.4	Schematic of a laminar separation bubble [15]	53
2.5	Velocity vectors of a laminar separation bubble a) averaged result b-d) instantaneous results [13]	54
2.6	Static Pressure distribution about an aerofoil with a laminar separation bubble [49]	54
2.7	Regions of a turbulent boundary layer	55
2.8	Comparison of predicted and measured skin friction for $k - k_L - \omega$ in the test case of Walters & Cokljat [46]	55
2.9	Downforce curve for a double-element wing [72]	56
2.10	Vortex system behind a McLaren F1 front wing a) laser-smoke visualisation (looking upstream from behind wing) and b) schematic of vortex system showing vortex formation due to pressure gradients [77]	56
3.1	Schematic of DS Houghton Wind Tunnel	90
3.2	CAD illustration of 2D wing in wind tunnel test section	90

3.3	Diagram of 2D wing section, showing upper and lower sections fitted together (left), and lower section (right) in isolation. Letters indicate cable access hole (A), large-brass pin slot (B), small-brass pin slot (C), and fixing bolt holes (D)	91
3.4	Photograph showing the instrumentation inside the 2D wing	91
3.5	Schematic of endplate geometry (2D wing)	92
3.6	Top view (looking vertically downwards) of wing fitted with wool tufts with tunnel running at $25ms^{-1}$ (2D wing, flow moving top to bottom) . .	92
3.7	Photograph of setup for spanwise pressure tests (2D wing, flow from left to right)	93
3.8	a) dynamic pressure, b) total pressure and c) static pressure at various spanwise locations measured with Pitot tube ($y/c = 0$ is wind tunnel centreline, 2D wing)	93
3.9	Diagram of single-element wing (CSW-SE)	94
3.10	Diagram of multi-element wing (CSW-ME)	94
3.11	Photograph of Tatuus F4-T014 car as used in 2015 ADAC Formel 4 series [118]	94
3.12	Photograph of Dallara T12 car as used in 2015 Formula V8 3.5 series. Copyright Formula V8 3.5 [119]	95
3.13	Diagram of F1 Wing	95
3.14	Aerofoil sections used at the a) central region, b) mid-semi span (taken at 225 mm from centreline) and c) wing tip of the F1 wing	95
3.15	Dimensions of serrated trip	96
3.16	Quasi-shear stress distribution on the suction surface of 2D wing with error bars showing uncertainty at each measurement location for varying Reynolds number	96
3.17	Photograph of three-dimensional wing (single-element configuration) installation	97
3.18	Diagram of hot-film sensor locations	97
3.19	Photograph of the LDA system in use	98
3.20	LDA measurement planes	98

3.21	Static-pressure tap locations	99
4.1	MSES grid generation	111
4.2	Slice of computational grid at symmetry plane for single-element wing . .	112
4.3	Slice of computational grid at symmetry plane for multi-element wing . .	112
5.1	Suction-surface static pressure distribution at varying Reynolds number, a-c) $\alpha = 0^\circ$, d-f) $\alpha = 3^\circ$, g-i) $\alpha = 5^\circ$, and varying ground clearance of $h/c = 0.5$ (left column), $h/c = 0.3$ (centre column), and $h/c = 0.1$ (right column) (Experimental)	145
5.2	Suction-surface static pressure distribution at varying incidence $0^\circ \leq \alpha \leq$ 15° at a) $Re = 3.59 \times 10^5$ and b) $Re = 4.79 \times 10^5$ (Experimental)	146
5.3	Suction-surface static pressure distribution at varying ground clearance $0.1 \leq h/c \leq 0.9$ at a) $\alpha = 0^\circ$, b) $\alpha = 3^\circ$, and c) $\alpha = 5^\circ$ at $Re = 4.79 \times 10^5$ (Experimental)	146
5.4	Suction-surface static pressure distribution with flow visualisation to show key flow features (Experimental flow visualisation and computational static pressure distribution, $Re = 7.19 \times 10^5$, $\alpha = 0^\circ$, $h/c = 0.3$, flow moving left to right)	147
5.5	Suction-surface flow visualisation at a) $Re = 3.59 \times 10^5$, b) $Re = 4.79 \times$ 10^5 , c) $Re = 5.99 \times 10^5$ and d) $Re = 7.19 \times 10^5$ (Experimental, $\alpha = 0^\circ$, $h/c = 0.3$, flow moving top to bottom, blue line showing reattachment) .	147
5.6	Suction-surface flow visualisation at a) $\alpha = 0^\circ$, b) $\alpha = 3^\circ$, and c) $\alpha = 5^\circ$ (Experimental, $Re = 4.79 \times 10^5$, $h/c = 0.3$, flow moving top to bottom) .	148
5.7	Suction-surface flow visualisation at $\alpha = 15^\circ$ (Experimental, $Re = 4.79 \times$ 10^5 , $h/c = 0.3$, flow moving top to bottom) - close up view of leading edge on the right	148
5.8	a) τ_Q and b) CV of hot-film gauges at varying Reynolds number (Experi- mental, $\alpha = 0^\circ$, $h/c = 0.3$)	149
5.9	a) τ_Q and b) CV of hot-film gauges at varying incidence (Experimental, $h/c = 0.3$, $Re = 5.99 \times 10^5$)	149
5.10	a) τ_Q and b) CV of hot-film gauges at varying Reynolds number (Experi- mental, $\alpha = 0^\circ$, $h/c = 0.1$)	150

5.11 a) τ_Q and b) CV of hot-film gauges at varying incidence (Experimental, $h/c = 0.1$, $Re = 4.79 \times 10^5$)	150
5.12 PSD of hot-film gauges for varying Reynolds number (Experimental, $\alpha = 0^\circ$, $h/c = 0.3$)	151
5.13 Diagram of separation bubbles of varying length due to growth of instabilities in the pre-bubble boundary layer	152
5.14 PSD of hot-film gauges for varying incidence (Experimental, $h/c = 0.3$, $Re = 5.99 \times 10^5$)	153
5.15 PSD of hot-film gauges for varying ground clearance and varying Reynolds number (Experimental, $\alpha = 0^\circ$)	154
5.16 LDA wake survey of x-velocity at varying Reynolds number at a) $x/c = 1.5$, b) $x/c = 2.0$, and c) $x/c = 3.0$ (Experimental, $h/c = 0.3$, $\alpha = 0^\circ$)	155
5.17 LDA wake survey of x-velocity at varying Reynolds number at a) $x/c = 1.5$, b) $x/c = 2.0$, and c) $x/c = 3.0$ (Experimental, $h/c = 0.3$, $\alpha = 3^\circ$)	155
5.18 LDA wake survey of x-velocity at varying Reynolds number at a) $x/c = 1.5$, b) $x/c = 2.0$, and c) $x/c = 3.0$ (Experimental, $h/c = 0.3$, $\alpha = 5^\circ$)	156
5.19 LDA survey of x-velocity under the wing at varying Reynolds number at $x/c = 0.375$ for $\alpha = 0^\circ$ (Experimental, $h/c = 0.3$)	156
5.20 Variation of location of transition (x_{Trans}), downforce coefficient and drag coefficient with Reynolds number a) $\alpha = 0^\circ$, b) $\alpha = 3^\circ$, and c) $\alpha = 5^\circ$ (MSES, $h/c = 0.3$)	157
5.21 Chordwise distributions of pressure coefficient, boundary-layer displacement thickness and friction coefficient at varying Reynolds number and incidence a-c) $\alpha = 0^\circ$, d-f) $\alpha = 3^\circ$, and g-i) $\alpha = 5^\circ$ for GA(W)-1 aerofoil (MSES, $h/c = 0.3$)	158
5.22 Comparison of pressure coefficient distributions from MSES (plain lines) and experiments (dotted lines) at a) $\alpha = 0^\circ$, b) $\alpha = 3^\circ$, and c) $\alpha = 5^\circ$ for GA(W)-1 aerofoil ($h/c = 0.3125$, $Re = 2.44 \times 10^5$)	159
5.23 Diagram of measurement area for Table 5.2, red lines show the start and end of bubble (defined by $C_F = 0$)	159
5.24 Chordwise distributions of pressure coefficient, boundary-layer displacement thickness and friction coefficient at various ground clearances for a-c) $Re = 2.85 \times 10^5$, d-f) $Re = 5.94 \times 10^5$, and g-i) $Re = 8.32 \times 10^5$ for GA(W)-1 aerofoil (MSES, $\alpha = 0^\circ$)	160

5.25	Chordwise distributions of pressure coefficient, boundary-layer displacement thickness, and friction coefficient at varying turbulence intensity for GA(W)-1 aerofoil (MSES, $Re = 5.94 \times 10^5$, $\alpha = 0^\circ$, $h/c = 0.3$)	161
5.26	Chordwise distributions of pressure coefficient, boundary-layer displacement thickness, and friction coefficient at varying turbulence intensity for Tyrell-026 aerofoil (MSES, $h/c = 0.3$, $\alpha = 0^\circ$)	161
6.1	Curves of a) downforce and b) rate of change of downforce change with non-dimensional ground clearance at varying Reynolds number (Experimental, CSW-SE)	201
6.2	Variation of drag coefficient with ground clearance for varying Reynolds number (Experimental, CSW-SE)	201
6.3	Reynolds-number dependency at constant ground clearances (Experimental, CSW-SE)	202
6.4	Suction-surface flow visualisation (Experimental, CSW-SE, $h/c = 0.3125$, $Re = 2.44 \times 10^5$, flow moving top to bottom)	203
6.5	Suction-surface flow visualisation at Reynolds numbers a) 1.63×10^5 b) 2.04×10^5 and c) 2.44×10^5 (Experimental, CSW-SE, $h/c = 0.3125$, flow moving top to bottom)	203
6.6	Suction-surface flow visualisation at Reynolds numbers a) 1.63×10^5 b) 2.04×10^5 and c) 2.44×10^5 (Experimental, CSW-SE, $h/c = 0.125$, flow moving top to bottom)	204
6.7	Suction-surface flow visualisation at the wing tip at Reynolds numbers a) 1.63×10^5 b) 2.04×10^5 and c) 2.44×10^5 (Experimental, CSW-SE, $h/c = 0.3125$, flow moving top to bottom)	204
6.8	Suction-surface flow visualisation at normalised ground clearances of a) $h/c = 0.125$ b) $h/c = 0.225$ and c) $h/c = 0.3175$ (Experimental, CSW-SE, $Re = 2.44 \times 10^5$, flow moving top to bottom)	205
6.9	Streaklines on the suction surface of the single-element wing from a) wind-tunnel testing and b) RANS $k - k_L - \omega$ CFD (CSW-SE, $h/c = 0.3125$, $Re = 2.44 \times 10^5$, flow moving top to bottom)	205
6.10	Static pressure distribution at centreline ($z/s = 0$) and part-semi span ($k - k_L - \omega$ RANS, CSW-SE, $h/c = 0.3125$, $Re = 2.44 \times 10^5$)	206

6.11	a) separation bubble shown by streamlines generated by velocity, and b) shear stress and turbulent kinetic energy on the lower surface ($z/s = 0$, $k - k_L - \omega$ RANS, CSW-SE, $h/c = 0.3125$, $Re = 2.44 \times 10^5$, flow field distorted for clarity ¹)	206
6.12	Comparison of experimental suction-surface flow visualisation (top left in each case) and computational surface streamlines generated by shear stress (bottom left in each case), and contours of turbulent kinetic energy with streamlines generated by velocity (right in each case) at a) $Re = 1.63 \times 10^5$, b) $Re = 2.04 \times 10^5$, and c) $Re = 2.44 \times 10^5$ ($k - k_L - \omega$ RANS, CSW-SE, $h/c = 0.3125$, flow moving left to right, cross-section of flow field distorted for clarity ¹)	207
6.13	Comparison of surface streamlines generated by $k - k_L - \omega$ (left) and $k - \omega$ SST (right) (RANS, CSW-SE, $h/c = 0.3125$, $Re = 2.44 \times 10^5$, flow moving top to bottom)	208
6.14	Comparison of closure models at main-plane centre line: contours of a) ΔC_p , b) ΔP_0 loss, and c) Δk (RANS, CSW-SE, $h/c = 0.3125$, $Re = 2.44 \times 10^5$)	209
6.15	Variation of wing forces with ground clearance for varying Reynolds numbers a) downforce b) drag (Experimental, CSW-ME)	210
6.16	Percentage loss of a) downforce and b) drag of each Reynolds number compared to $Re = 2.44 \times 10^5$ (Experimental, CSW-ME)	210
6.17	Flow visualisation of a) suction surfaces and b) pressure surfaces (Experimental, CSW-ME, $h/c = 0.3125$, $Re = 2.44 \times 10^5$, flow moving top to bottom)	211
6.18	Suction-surface flow visualisation of a) flow regimes at main-plane centre and b) close-up of turbulent reattachment at main-plane centre (Experimental, CSW-ME, $h/c = 0.3125$, $Re = 2.44 \times 10^5$, flow moving top to bottom)	211
6.19	Suction-surface flow visualisation for $h/c = 0.3125$ at Reynolds numbers a) 1.63×10^5 , b) 2.04×10^5 , and c) 2.44×10^5 (Experimental, CSW-ME, flow moving top to bottom)	212
6.20	Suction-surface flow visualisation for $h/c = 0.225$ at Reynolds numbers a) 1.63×10^5 , b) 2.04×10^5 , and c) 2.44×10^5 (Experimental, CSW-ME, flow moving top to bottom)	212

6.21	Suction-surface flow visualisation for $h/c = 0.125$ at Reynolds numbers a) 1.63×10^5 , b) 2.04×10^5 , and c) 2.44×10^5 (Experimental, CSW-ME, flow moving top to bottom)	213
6.22	Flow visualisation of inner endplate surface at a) $h/c = 0.3125$, and b) $h/c = 0.125$ (Experimental, CSW-ME, $Re = 2.44 \times 10^5$, flow moving right to left)	213
6.23	Comparison of experimental suction-surface flow visualisation and CFD surface streamlines on suction surface (CSW-ME, $Re = 2.44 \times 10^5$, $h/c =$ 0.3125 , flow moving top to bottom)	213
6.24	Static pressure and x-component of shear stress distributions on the suc- tion surface of multi-element wing main-plane at $z/s = 0$ and $z/s = 2/3$ ($k - k_L - \omega$ RANS, CSW-ME, $h/c = 0.3125$, $Re = 2.44 \times 10^5$)	214
6.25	Contours of ΔC_P on the pressure (left) and suction (right) wing surfaces, and C_P distribution at $z/s = 2/3$ (plane A-A) ($k - k_L - \omega$ RANS, CSW- ME, $h/c = 0.3125$, $Re = 2.44 \times 10^5$)	215
6.26	Contours of total-pressure coefficient loss in the flow-field behind the wing at a) $x/c = 2.25$, b) $x/c = 2.5$, and c) $x/c = 3$ ($k - k_L - \omega$ RANS, CSW-ME, $h/c = 0.3125$, $Re = 2.44 \times 10^5$)	216
6.27	Contours of turbulent kinetic energy and streamlines of velocity at $z/s =$ $2/3$, and graph of x-component of shear stress and turbulent kinetic en- ergy on suction surfaces ($k - k_L - \omega$ RANS, CSW-ME, $h/c = 0.3125$, $Re = 2.44 \times 10^5$, flow-field distorted for clarity - see footnote ¹ on p200) .	217
6.28	Comparison of experimental flow visualisation CFD surface streamlines at a) $Re = 2.44 \times 10^5$ and b) $Re = 1.63 \times 10^5$ ($k - k_L - \omega$ RANS, CSW- ME, $h/c = 0.3125$, flow moving top to bottom)	218
6.29	Comparison of surface streamlines (flow moving top to bottom) generated by $k - k_L - \omega$ (left) and $k - \omega$ SST (right)(RANS, CSW-ME, $Re = 2.44 \times$ 10^5 , $h/c = 0.3125$)	218
6.30	Comparison of closure models at main-plane centre line: contours of a) ΔC_P , b) ΔP_0 loss, and c) Δk (RANS ($(k - k_L - \omega) - (k - \omega$ SST)), CSW- ME, $h/c = 0.3125$, $Re = 2.44 \times 10^5$)	219
6.31	Variation of force coefficient with ground clearance for varying Reynolds number a) downforce b) drag (Experimental, F1 wing)	220

6.32	Suction-surface flow visualisation on F1 wing for varying Reynolds numbers a) $Re = 1.63 \times 10^5$ and b) $Re = 2.44 \times 10^5$ (Experimental, F1 wing, $h/c = 0.3125$, flow moving top to bottom)	220
6.33	Flow visualisation of F1 wing suction-surfaces (top) and inner endplate (bottom) at a) $h/c = 0.125$ and b) $h/c = 0.3125$ (Experimental, $Re = 2.44 \times 10^5$, flow moving top to bottom)	221
7.1	Suction-surface pressure distributions for free- and forced-transition at varying wing incidence a) $\alpha = 0^\circ$, b) $\alpha = 3^\circ$, and c) $\alpha = 5^\circ$ (Experimental, 2D wing, points for tappings covered by trip are removed, $Re = 5.94 \times 10^5$, $h/c = 0.3$)	247
7.2	Suction-surface pressure distribution for varying Reynolds number with a) free-transition and b) transition forced at $x/c = 0.1$ (Experimental, 2D wing, points for tappings covered by trip are removed, $h/c = 0.3$, $Re = 5.94 \times 10^5$)	247
7.3	Suction-surface flow visualisation for free- and forced-transition cases at incidence a) $\alpha = 0^\circ$, b) $\alpha = 3^\circ$, and c) $\alpha = 5^\circ$ (Experimental, 2D wing, flow moving from top to bottom, $h/c = 0.3$, $Re = 6 \times 10^5$)	248
7.4	Quasi-wall-shear stress for free- and forced-transition cases at incidence a) $\alpha = 0^\circ$, b) $\alpha = 3^\circ$, and c) $\alpha = 5^\circ$ (Experimental, 2D wing, $h/c = 0.3$, $Re = 6 \times 10^5$)	248
7.5	CV for free- and forced-transition cases at incidence a) $\alpha = 0^\circ$, b) $\alpha = 3^\circ$, and c) $\alpha = 5^\circ$ (Experimental, 2D wing, $h/c = 0.3$, $Re = 6 \times 10^5$)	249
7.6	PSD of hot-film gauge outputs for free- and forced-transition cases at $\alpha = 0^\circ$ (Experimental, 2D wing, $h/c = 0.3$, $Re = 6 \times 10^5$)	250
7.7	Wake surveys for free- and forced-transition cases at $\alpha = 0^\circ$ for a) $x/c = 1.5$, b) $x/c = 2$, and c) $x/c = 3$ (Experimental, 2D wing, $h/c = 0.3$, $Re = 6 \times 10^5$)	251
7.8	Surveys for free- and forced-transition cases at $x/c = 0.375$ for a) $\alpha = 0^\circ$, b) $\alpha = 3^\circ$, and c) $\alpha = 5^\circ$ (Experimental, 2D wing, $h/c = 0.3$, $Re = 6 \times 10^5$)	251
7.9	Downforce measurements of CSW-SE wing at varying Reynolds number, ground clearance and trip location a) clean wing, b) tripped at $x/c = 0.1$, c) tripped at $x/c = 0.2$, d) tripped at $x/c = 0.3$, e) tripped at $x/c = 0.4$ and f) tripped at $x/c = 0.5$ (Experimental, CSW-SE, roughness trip)	252

7.10	Drag measurements of CSW-SE wing at varying Reynolds number, ground clearance and trip location a) clean wing, b) tripped at $x/c = 0.1$, c) tripped at $x/c = 0.2$, d) tripped at $x/c = 0.3$, e) tripped at $x/c = 0.4$ and f) tripped at $x/c = 0.5$ (Experimental, CSW-SE, roughness trip)	253
7.11	Suction-surface flow visualisation for varying trip location at $h/c = 0.125$ (left), $h/c = 0.225$ (middle), and $h/c = 0.3125$ (right) for a) clean wing, b) tripped at $x/c = 0.1$, c) tripped at $x/c = 0.3$ and d) tripped at $x/c = 0.5$ (Experimental, CSW-SE, roughness trip, $Re = 2.44 \times 10^5$, flow moving from top to bottom)	254
7.12	Suction-surface flow visualisation with boundary-layer trips at $x/c = 0.1$ for varying Reynolds numbers $Re = 1.63 \times 10^5$ (left), $Re = 2.03 \times 10^5$ (middle), and $Re = 2.44 \times 10^5$ (right) at a) $h/c = 0.3125$, b) $h/c = 0.225$, and c) $h/c = 0.125$ (Experimental, CSW-SE, roughness trip, $Re = 2.44 \times 10^5$, flow moving from top to bottom)	255
7.13	Experimental downforce measurements of CSW-ME wing at varying Reynolds number, ground clearance and trip location a) clean wing, b) tripped at $x/c = 0.1$, c) tripped at $x/c = 0.2$, d) tripped at $x/c = 0.3$, e) tripped at $x/c = 0.4$ and f) tripped at $x/c = 0.5$ (Experimental, CSW-ME, roughness trip)	256
7.14	Experimental drag measurements of CSW-ME wing at varying Reynolds number, ground clearance and trip location a) clean wing, b) tripped at $x/c = 0.1$, c) tripped at $x/c = 0.2$, d) tripped at $x/c = 0.3$, e) tripped at $x/c = 0.4$ and f) tripped at $x/c = 0.5$ (Experimental, CSW-ME, roughness trip)	257
7.15	Suction-surface flow visualisation on multi-element wing for a) clean wing, b) tripped at $x/c = 0.1$, c) tripped at $x/c = 0.2$, d) tripped at $x/c = 0.3$ (Experimental, CSW-ME, roughness trip, $Re = 2.44 \times 10^5$, $h/c = 0.3125$, flow moving from top to bottom)	258
7.16	Suction-surface flow visualisation on multi-element wing with trip at $x/c = 0.1$ for a) $Re = 1.63 \times 10^5$, b) $Re = 2.03 \times 10^5$, and c) $Re = 2.44 \times 10^5$ (Experimental, CSW-ME, roughness trip, $h/c = 0.3125$, flow moving from top to bottom)	258
7.17	Suction-surface flow visualisation on multi-element wing with trip at $x/c = 0.1$ for a) $h/c = 0.125$, b) $h/c = 0.225$, and c) $h/c = 0.3125$ (Experimental, CSW-ME, roughness trip, $Re = 2.44 \times 10^5$, flow moving from top to bottom)	258

- 7.18 Downforce measurements of CSW-SE wing at varying Reynolds number, ground clearance and trip location a) clean wing, b) tripped at $x/c = 0.1$, c) tripped at $x/c = 0.2$, d) tripped at $x/c = 0.3$, e) tripped at $x/c = 0.4$ and f) tripped at $x/c = 0.5$ (Experimental, CSW-SE, serrated trip) 259
- 7.19 Drag measurements of CSW-SE wing at varying Reynolds number, ground clearance and trip location a) clean wing, b) tripped at $x/c = 0.1$, c) tripped at $x/c = 0.2$, d) tripped at $x/c = 0.3$, e) tripped at $x/c = 0.4$ and f) tripped at $x/c = 0.5$ (Experimental, CSW-SE, roughness trip) 260
- 7.20 Downforce measurements of CSW-SE wing at varying Reynolds number and ground clearance for a) clean wing, b) tripped at $x/c = 0.1$, c) tripped at $x/c = 0.2$, d) tripped at $x/c = 0.3$, e) tripped at $x/c = 0.4$ and f) tripped at $x/c = 0.5$ with serrated trip (solid line) and roughness trip (dashed line) 261
- 7.21 Drag measurements of CSW-SE wing at varying Reynolds number and ground clearance for a) clean wing, b) tripped at $x/c = 0.1$, c) tripped at $x/c = 0.2$, d) tripped at $x/c = 0.3$, e) tripped at $x/c = 0.4$ and f) tripped at $x/c = 0.5$ with serrated trip (solid line) and roughness trip (dashed line) 262
- 7.22 Suction-surface flow visualisation for a) free-transition and serrated trip placed at b) $x/c = 0.1$, c) $x/c = 0.3$, and d) $x/c = 0.5$ (Experimental, CSW-SE, $h/c = 0.3125$, $Re = 2.44 \times 10^5$, serrated trip, flow moving top to bottom) 263
- 7.23 Suction-surface flow visualisation for serrated trip placed at $x/c = 0.1$ for a) $h/c = 0.3125$, b) $h/c = 0.225$, and c) $h/c = 0.125$ at $Re = 1.63 \times 10^5$ (left), $Re = 2.03 \times 10^5$ (middle), and $Re = 2.44 \times 10^5$ (right) (Experimental, CSW-SE, serrated trip, flow moving from top to bottom) 264
- 7.24 Comparison of free transition (left) and forced transition at $x/c = 0.3$ (right) for $h/c = 0.125$ at $Re = 1.63 \times 10^5$ (Experimental, CSW-SE, serrated trip, flow moving from top to bottom) 265
- 7.25 Downforce measurements of CSW-ME wing at varying Reynolds number, ground clearance and trip location a) clean wing, b) tripped at $x/c = 0.1$, c) tripped at $x/c = 0.2$, d) tripped at $x/c = 0.3$, e) tripped at $x/c = 0.4$ and f) tripped at $x/c = 0.5$ (Experimental, CSW-ME, serrated trip) 266
- 7.26 Drag measurements of CSW-ME wing at varying Reynolds number, ground clearance and trip location a) clean wing, b) tripped at $x/c = 0.1$, c) tripped at $x/c = 0.2$, d) tripped at $x/c = 0.3$, e) tripped at $x/c = 0.4$ and f) tripped at $x/c = 0.5$ (Experimental, CSW-ME, serrated trip) 267

- 7.27 Suction-surface flow visualisation for a) free-transition and forced-transition wing with serrated trip at b) $x/c = 0.1$, c) $x/c = 0.3$, and d) $x/c = 0.5$ (Experimental, CSW-ME, serrated trip, $h/c = 0.3125$, $Re = 1.63 \times 10^5$, flow moving top to bottom) 268
- A.1 Comparison of a) experimental flow visualisation to CFD surface streamlines, generated by shear stress, with inlet turbulence intensity of b) $Tu = 0.5\%$, c) $Tu = 3\%$, and d) $Tu = 5\%$ ($Re = 2.44 \times 10^5$, $h/c = 0.3125$, flow from top to bottom, CSW-SE (top) and CSW-ME (bottom) in each case) . 293
- A.2 x-component of shear stress on suction surface of a) single-element wing, and b) multi-element wing at $z/s = 0$ and $z/s = 2/3$ for varying inlet turbulence intensity ($Re = 2.44 \times 10^5$, $h/c = 0.3125$, RANS $k - k_L - \omega$) . 294

List of Tables

3.1	Ground clearance increase due to change of incidence	63
3.2	Test Reynolds numbers (chord-based) for 2D wing experiments	66
3.3	Maximum blockage ratio for test models (including strut)	72
3.4	Test Reynolds numbers (chord-based) for 3D wing experiments	74
3.5	CSW wind-tunnel test cases	76
3.6	F1 wing wind-tunnel test cases	77
3.7	Average Reynolds number and temperature change recorded by diagnostics for LDA wake profiles	86
5.1	Magnitude and location of velocity deficit at varying Reynolds number and location behind wing	129
5.2	Integral of static pressure distribution where $C_F = 0$ i.e. inside separation bubble (MSES, $h/c = 0.3125$)	134
5.3	Integral of friction coefficient distribution ($h/c = 0.3$, $\alpha = 0^\circ$)	136
6.1	Separation bubble size and location at the centreline of the wing at varying Reynolds number (Experimental, CSW-SE, $h/c = 0.3125$)	167
6.2	Separation bubble size and location at centre at varying ground clearance (Experimental, CSW-SE, $Re = 2.44 \times 10^5$)	170
6.3	Separation bubble size and location at the wing's centreline (Experimental, CSW-SE, $Re = 2.44 \times 10^5$)	171

6.4	Comparison of experimental and computational downforce and drag coefficients, and errors as a percentage of full scale for CSW-SE at varying Reynolds numbers ($h/c = 0.3125$)	175
6.5	Comparison of experimental and computational downforce and drag coefficients for CSW-SE for fully-turbulent and transition-sensitive turbulence models ($h/c = 0.3125$)	177
6.6	Separation bubble size and location at the wing's main-plane centreline for varying Reynolds number (Experimental, CSW-ME, $h/c = 0.3125$) . .	185
6.7	Comparison of downforce and drag forces for experiments and computations for CSW-ME ($Re = 2.44 \times 10^5$, $h/c = 0.3125$)	188
6.8	Comparison of downforce and drag coefficients from experiments and $k - k_L - \omega$ RANS computations for CSW-ME at varying Reynolds numbers ($h/c = 0.3125$)	191
6.9	Downforce and drag coefficients generated by each closure model compared to the experimental result ($Re = 2.44 \times 10^5$, $h/c = 0.3125$)	193
7.1	Percentage difference of the total area bounded by the velocity deficit between the forced-transition cases and the free-transition case	231
7.2	Comparison of the ground clearances at which the force-reduction phenomenon begins for varying Reynolds numbers and trip location, '-' denotes that no force reduction has occurred and thus maximum downforce is at the lowest tested ground clearance	233
A.1	AeroTech force balance uncertainty in measured coefficient	287
A.2	Contribution to total uncertainty of force coefficients	288
A.3	Variation of each variable between two tests and uncertainty of variable measurement	288
A.4	Comparison of downforce and drag generated by varying inlet turbulence intensity ($k - K_L - \omega$ RANS, $Re = 2.44 \times 10^5$, $h/c = 0.3125$)	291
A.5	IcemCFD mesh settings	295
A.6	IcemCFD part settings	295
A.7	Fluent solution methods	296
A.8	Fluent under-relaxation factors for $k - k_L - \omega$ simulations	296

Nomenclature

c	Wing main-plane chord, m
C_D	Drag coefficient ($\frac{D}{q_\infty S}$)
C_F	Friction coefficient
C_L	Downforce coefficient ($\frac{L}{q_\infty S}$)
C_P	Pressure coefficient ($\frac{\Delta P}{q_\infty}$)
CV	Coefficient of variation
CFD	Computational fluid dynamics
CAD	Computer-aided design
DES	Detached-eddy simulation
D	Drag force, N
D_L	Anisotropic (near-wall) dissipation term for k_L
D_T	Anisotropic (near-wall) dissipation term for k
E	Hot-film output voltage, V

E_0	Hot-film zero-shear stress voltage, V
f	Frequency, Hz
f_w	Inviscid near-wall damping function
f_ω	Boundary layer wake term damping function in ω equation
h	Ground clearance, m
I	Turbulence intensity, %
k	Turbulent kinetic energy, m^2s^{-1}
k_L	Laminar kinetic energy, m^2s^{-1}
l	Turbulence length scale, m
l_m	Total model length, m
l_x	Reference length, m
L	Downforce, N
LES	Large-eddy simulation
M	Mach number
N	Critical amplification factor
P	Static pressure, Nm^{-2}
P_0	Total pressure, Nm^{-2}
P_{k_L}	Production of k_L by mean strain
P_{k_T}	Production of k by mean strain

PSD	Power spectral density, $(\text{Nm}^{-2})^2$
q	Dynamic pressure, Nm^{-2}
q_∞	Freestream dynamic pressure, Nm^{-2}
RANS	Reynolds-averaged Navier Stokes
Re	Chord-based Reynolds number $\left(\frac{Vc}{\nu}\right)$
Re_θ	Momentum-thickness-based Reynolds number $\left(\frac{V\theta}{\nu}\right)$
R_{BP}	Bypass transition production term
R_{NAT}	Natural transition production term
s	Wing semi-span, m
S	Wing main-plane planform area, m^2
U_*	Shear velocity, ms^{-1}
V_∞	Freestream velocity, ms^{-1}
V_x	x-component of velocity, ms^{-1}
x_j	Position vector
x, y, z	Cartesian coordinates (x—streamwise, y—ground normal, z—spanwise)
x_S/c	Normalised chordwise bubble separation point
x_R/c	Normalised chordwise bubble reattachment point
x_L/c	Normalised chordwise bubble length
x_T/c	Normalised chordwise transition location

y_f/c	Local, normalised flow-field y coordinate
Y^+	Dimensionless wall distance $\left(\frac{U_* Y}{\nu}\right)$
Y	Distance to nearest wall, m
α	Angle of incidence, $^\circ$
α_T	Effective diffusivity for turbulence-dependent variables
δ^*	Boundary-layer displacement thickness, m
ΔC_D	Percentage change in C_D between max and min Re cases $\left(\frac{C_{D_{Re_{Min}}} - C_{D_{Re_{Max}}}}{C_{D_{Re_{Min}}}}\right)$
ΔC_L	Percentage change in C_L between max and min Re cases $\left(\frac{C_{L_{Re_{Min}}} - C_{L_{Re_{Max}}}}{C_{L_{Re_{Min}}}}\right)$
Δk	Difference in k between turbulence models $((k - k_L - \omega) - (k - \omega_{SST}))$, m^2s^{-1}
ΔP	Local static pressure relative to freestream, Nm^{-2}
ΔP_0	Difference in P_0 between turbulence models $((k - k_L - \omega) - (k - \omega_{SST}))$, Nm^{-2}
θ	Boundary-layer momentum thickness, m
ν	Kinematic viscosity, m^2s^{-1}
ρ	Density, kgm^{-3}
σ	Standard deviation of quasi-wall-shear stress, Nm^{-2}
τ	Shear stress, Nm^{-2}
τ_x	x-component of shear stress, Nm^{-2}

τ_Q	Quasi-wall-shear stress, $(\frac{E^2 - E_0^2}{E_0^2})^3$
$\overline{\tau_Q}$	Mean quasi-wall-shear stress, Nm^{-2}
ω	Specific dissipation rate, s^{-1}

$k - k_L - \omega$ model constants

$C_{\omega 1} = 0.44$	$C_{\omega R} = 1.4$	$C_{\omega 2} = 0.92$	$C_{\omega 3} = 0.3$	$\sigma_k = 1$	$\sigma_\omega = 1.17$
-----------------------	----------------------	-----------------------	----------------------	----------------	------------------------

Acknowledgements

I would like to extend my sincerest gratitude to both my supervisors: Dr. Mark Finnis and Prof. Kevin Knowles. Not only have they provided me with the knowledge and confidence to accomplish many things over the past three years, including presenting my work at conferences and being published in leading academic journals, but they also paved the way for me to achieve my goal of working in Formula One.

I also acknowledge the support of numerous other members of staff and students with whom I have had the privilege of working during my time at Cranfield University. Particularly, I would like to thank Dr. Alistair Saddington for providing guidance on my work, Keith Brook for the design and manufacture of the hot-film sensor apparatus, Brian Duguid for the manufacture of the wind tunnel models, and Nick Lawson for his help with LDA measurements.

I am extremely appreciative of the funding awarded to me by Cranfield Defence & Security, without which I would not have been able to undertake the work contained in this document.

I am grateful to my industrial colleagues from my internship programme at Scuderia Toro Rosso Formula One team. In particular I would like to thank: Brendan Gilhome, Ben Mallock, Franck Sanchez, Peter Coysh, Jerome Perrin, Colin Wooding, David Barrera,

and David Smith. The knowledge and experience of both aerodynamics and vehicle dynamics that they shared with me were not only of immense benefit when writing this thesis, but also improved me as an aerodynamics engineer.

Thanks to Fortec Motorsport, who provided me the opportunity to understand the real-world application of racing-car aerodynamics. I am grateful to Trevor Foster, Russell Dixon and Paul Anderson for sharing their years of experience so openly with me. To Adam Dannan and the numerous other mechanics and engineers at Fortec; it has been a pleasure to work with them all.

Last, and far from least, thanks to all my family and friends. I am especially grateful to my parents and better half, Sarah Ward, for their support and encouragement, which has enabled me to fulfil my goals and get to where I am today.

Chapter 1

Introduction

The work presented in this thesis aspires to contribute towards the field of racing car aerodynamics from a perspective that has been mostly ignored, despite the fact that its fundamental importance has been observed in the aeronautical sector for decades. The boundary layer is arguably the most important aerodynamic characteristic of a wall-bounded flow. Its importance is even more significant for the low Reynolds number operating conditions of racing cars.

Aerodynamics has become the primary performance differentiator in the majority of monoposto racing car series. Whilst it is widely known that aerodynamics defines the performance of a Formula One car, the same is true even for lower formulae where significant levels of aerodynamic downforce are still produced.

The first chapter of this thesis describes the fundamentals of boundary-layer transition, and of wings operating in close proximity to the ground through a review of the significant published works of the fields. This will put into context the motivation behind the work and thus allow aims and objectives to be clearly defined.

This thesis aims to provide an understanding of boundary-layer transition and its effect on the aerodynamic characteristics of wings of varying complexity; from two-dimensional aerofoil sections to highly-complex F1-specification front wings. The entire array of tools available to the modern aerodynamicist has been employed in this study, including experimental and computational techniques.

The results are divided into sections based on geometry. The first section observes purely two-dimensional characteristics of a single-element aerofoil. This aerofoil section is then transformed into a three-dimensional single-element wing, followed by a simple multi-element, and finally a highly complex (F1 specification) multi-element wing. Discussions are presented at the end of each section before being drawn together to give the overall conclusions of the work.

1.1 The Aerodynamic Requirement

The design of a competitive racing car in any contemporary form of motorsport, be it Formula One (F1), Indycar or Sports cars, relies upon a well-designed aerodynamic system which provides maximum downforce and minimum drag in a stable and predictable manner. By increasing the vertical load on the tyres through the creation of aerodynamic downforce the lateral and longitudinal force capabilities of a tyre are increased, and as a consequence, higher cornering speeds and better traction/braking are enabled. Further information on the effect of downforce on car handling can be found in Smith [1]. Overall, increasing downforce reduces the braking distance, allows the car to navigate corners at a higher speed and transmit greater thrust force without wheel spin under acceleration. As a result, a modern Formula One car is able to generate up to 5G under braking and cornering, and 3G under acceleration (limited by the engine output). Therefore, al-

though aerodynamic efficiency (ratio of downforce to drag) is also important, an increase in downforce will outweigh any drag penalty on most occasions.

The production of downforce relies on three components, the front wing, the rear wing and the underbody, working in tandem. The flow field around a modern racing car involves highly complex and sensitive flow physics including:

- Large pressure gradients and very low pressures.
- Separation as a normal feature.
- Varying flow states and transition mechanisms.
- Wake interactions.
- Force enhancing vortices which wander, dissipate and burst.
- Unsteady flows.
- Aerodynamic hysteresis.

1.2 The Front Wing

The front wing of an F1 car produces approximately 25-30% [2] of the car's total downforce, however its design is dictated primarily by the way in which it influences flow to downstream components rather than to maximise downforce production. Principal functions of the front wing, other than generating downforce, are to control the wake of the front wheels, maximise mass flow into the underbody, and channel flow to cooling intakes. At the track, the front wing is primarily a tool to balance the characteristics of the car, in terms of understeer or oversteer, by moving the centre of pressure. The rear wing is used to set the overall downforce level of the car, then the front wing is altered to balance the distribution of downforce between the front and rear. Moving the centre of pressure forward gives an oversteer characteristic whereas a rearward movement gives understeer. The centre of pressure is by no means a fixed attribute, as the car completes a lap it will

continually change in heave, pitch, yaw and roll. Each of these motions affects the centre of pressure. Additionally, a driver will, in general, prefer a more rearward centre of pressure for high speed corners, to provide stability, and forward centre of pressure for low speed corners, to give a fast turning response. It is, therefore, important that a front wing produces downforce in a stable, consistent and predictable manner.

As the majority of downforce originates from the underbody it is undesirable for the front wing to extract a large amount of energy from the on-coming air. Maximising downforce production from the front wing is a redundant activity if the car cannot produce enough rear downforce to give an acceptable balance. Consequently, racing-car designers tend to load the outer portions of the wing so that any losses are confined to this area and away from the other key downstream components. The central region is then designed to direct flow that contains the highest possible energy to the underfloor, in turn enhancing the performance of the underbody diffuser. This outboard focussed philosophy makes wing and wheel interactions more important.

Interacting wake structures from bluff objects, such as the wheels, cause dramatic energy losses in the flow. Protecting sensitive features, such as the leading edge of the underfloor, from such wakes is important to improve performance through the use of more aggressive geometries. In isolation a rotating wheel is the source of both positive lift and a low-energy and highly unsteady wake. The front wing's components are designed to provide both outwash and vortex structures to control this wake and move it outboard, as well as altering the pressure field around it such that the wheel generates zero or negative lift instead.

The front wing is arguably the most difficult part of the car to represent in wind tunnel testing due to the varying turbulence level and total pressure conditions in which it must

operate on-track; yet in the wind tunnel it always receives high-energy, low-turbulence flow. Downstream features, on the other hand, are always operating in the wake of the front wing and front wheels, and thus are less susceptible to changes in oncoming flow condition. This is a particular problem for race-car aerodynamicists as in the wind tunnel the front wing will have laminar and transitional flows, whilst on track these are reduced by the higher Reynolds number, higher turbulence conditions, body accelerations, and body vibration.

The design of F1 front wings has become increasingly complex over recent years, and now includes multiple elements, multiple cascades, and highly three-dimensional curvature. A 2016-specification wing is shown in Figure 1.1. Whilst the inboard region of the main-plane is a regulation-defined symmetrical aerofoil section, the outboard region features multiple elements. The junction between these two regions forms a strong vortex, which is used further downstream to push the front-wheel wake outboard and away from the underbody. The wing elements curve downwards near to the outboard wing-tip and are shaped to maximise the diffuser effect by having a large expansion ratio. This curved region also houses the inevitable wing-tip vortex; the inclusion of a channel for the vortex prevents it bursting or stagnating on the ground when the wing is operating at a low ground clearance. This reduces the pitch sensitivity of the car as it helps provide downforce through consistent vortex-induced suction. If regulations allowed for it these elements would also form the endplate of the wing; the plate which is typically referred to as the endplate is in fact, purely to conform to regulations, and would often be omitted due the flaps being designed to act as endplates. Although the upper flap elements, which are mounted from the 'endplate', do aid in downforce generation, their primary function is to generate vortical structures that provide outwash to control the front wheel wake, particularly in yaw. The two functions of the front wing, therefore, are to generate down-

force using a multi-element flap configuration, and to generate flow structures to control the flow to downstream components.

1.3 Aerodynamic Development

Modern racing cars are developed through a combination of CFD simulations and wind tunnel testing. Currently in F1, despite the advances in computational methods, the majority of time and resource is spent on wind-tunnel testing. The advantage of wind-tunnel testing is the sheer number of test conditions and vehicle attitudes that can be assessed. Whilst CFD analysis suits for determining which new geometries should be tested in the wind tunnel, testing in the wind tunnel generally consists of generating aerodynamic-performance maps. This involves quasi-static testing of the car in pitch, heave, yaw and roll attitudes to understand how the performance of the car alters during the cornering manoeuvre. The goal is to ensure consistent downforce production and understand attitude changes due to centre of pressure movements during the manoeuvre. The difficulty with using CFD to produce the aerodynamic-performance maps is that a different mesh and new simulation is required for each attitude. Whilst this is possible, and there are methods to reduce both meshing and solving times, it is still a slow process in comparison to wind tunnel testing.

It is typical in wind-tunnel testing that the Reynolds and Mach numbers will be lower than those at which the part usually operates. The Reynolds number characterises critical on- and off-body flow features such as boundary-layer state (laminar or turbulent), flow separation and reattachment, and shear-layer stability. Sub-scale models will have a longer laminar boundary layer region, and once turbulence starts it will grow more quickly than on a full-scale car. Thus, recognising and understanding the consequences of testing at

reduced Reynolds numbers is important for correlating between track and tunnel. Key Reynolds and Mach number effects at wind-tunnel scale are:

- Energy of flow is reduced, increasing flow separation.
- Thicker boundary layers.
- Transition occurs later due to increased viscous effects.
- Increased wake sizes.
- A geometrically-scaled radiator at reduced Reynolds number will not produce the correct pressure drop for a given mass-flow rate .

1.4 Aims & Objectives

The low Reynolds number of racing-car wings, approximately $3.2 \times 10^5 \leq Re \leq 1.35 \times 10^6$, means that both laminar and turbulent boundary layer states are common. Whilst transition to turbulence on aerofoils and wings has been researched extensively in the aeronautical sector, there is very little conclusive work for wings operating in ground effect, where much lower pressures and stronger pressure gradients are common.

The flow-field of a modern racing car features a range of highly complex and sensitive flow physics. Due to the relatively low Reynolds and Mach numbers at which racing cars operate (in comparison to typical aeronautical applications), on-track and in the wind tunnel, it is common for both laminar, transitional, and turbulent boundary layers to occur. This is because the individual aerodynamic components, such as the front and rear wings, should be analysed using their chord lengths to calculate Reynolds number rather than the vehicle length. Although the majority of turbulence models for RANS-based CFD simulations assume fully-turbulent flow, the need for increased understanding of the effect of laminar flow has led to the development of transitional turbulence models. Despite

this, although transition aspects on ground-effect wings have been noted [88, 89], there is a lack of specific investigation into the subject.

The question posed by this thesis "can the flow characteristics of a full-scale racing-car wing be replicated at model scale, and represented adequately by simulation?". In addressing this question the research will extend, through the use of experimental and computational methods, the understanding of boundary-layer transition and Reynolds-number scaling for downforce-producing wings operating in close proximity to the ground. The following objectives were set to address the above question:

- Observe the transition mechanism for wings operating in ground effect and the influence of Reynolds number, ground clearance and incidence.
- Use the results to validate a numerical code such that other profiles can be observed.
- Ascertain whether two-dimensional effects are relevant to a three-dimensional single-element wing.
- Investigate how Reynolds number and transition affect wings of varying complexity, from single-element to highly-complex multi-element configurations.
- Evaluate the use of a transitional eddy-viscosity model and compare to a fully-turbulent model.
- Define the significance of laminar boundary-layer extent by forcing transition at various chordwise locations.
- Discuss the importance of boundary-layer transition and Reynolds-number scaling for wings in ground effect.

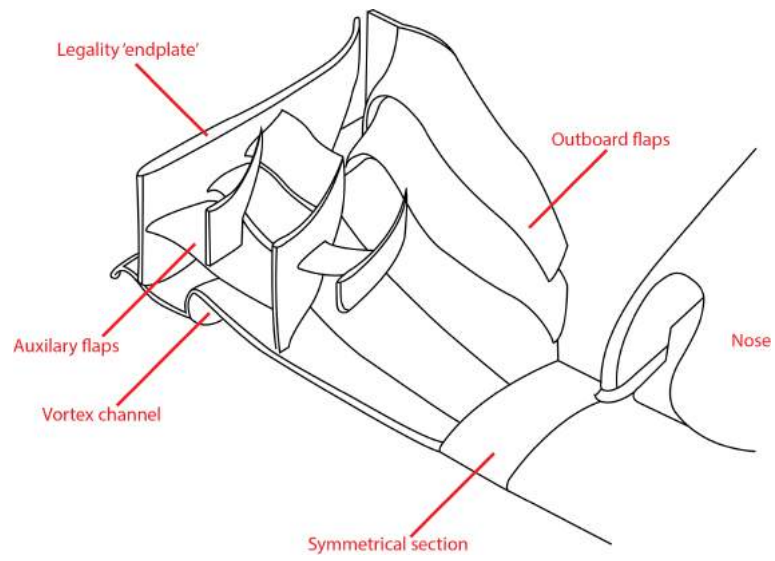


Figure 1.1: Scuderia Toro Rosso STR11 Formula One Car front wing

Chapter 2

Literature Review

2.1 Laminar-Turbulent Transition

The subject of turbulence is arguably the most fundamental aspect in the field of fluid dynamics. Moreover, for low-speed flows such as those encountered by ground vehicles, viscosity is an important variable as it leads to the formation of a boundary layer. This leads to the subject of this thesis, the influence of viscous flows on a wing operating in close proximity to the ground.

Viscosity is a physical characteristic that establishes the transport properties of the fluid. In particular, it defines the momentum transport perpendicular to the mean flow direction, which consequently affects the amplification of perturbations in the flow that lead to turbulence. Mathematically, the viscosity for a Newtonian fluid, such as air, can be described by a linear relationship with shear stress and strain rate (velocity gradient), as shown in Eqn. 2.1.

$$\tau = \mu \frac{du}{dy} \quad (2.1)$$

In this section the fundamental mechanisms of boundary-layer transition will be outlined, as will parameters which have an influence on the process, and finally, how the process can be modelled in RANS (Reynolds-averaged Navier Stokes) CFD (computational fluid dynamics). Emphasis will then be placed on studies that have experimentally observed transition, or conducted transition-sensitive RANS, on aerofoils.

2.1.1 The Reynolds Number

In 1883, Osborne Reynolds reported experiments in which he observed water flowing through a glass tube using a coloured pigment. In this work he observed the difference between laminar and turbulent flow states, and led to the concept of the Reynolds number.

Whilst the advances in CFD codes in recent years have made fully-computational development of racing cars possible, the testing of scale models in wind tunnels is still the prevailing practice, for the reasons discussed in Chapter 1. A key to winning in top-level motorsport is ensuring that components developed in the wind tunnel work in a similar manner on-track. This leads to one of the most fundamental concepts of aerodynamic testing, similarity. Whilst it is fairly obvious that geometric similarity is essential to any aerodynamic evaluation, the dynamic similarity can sometimes be left behind. The Reynolds number (Eqn. 2.2) is often cited as moderator of dynamic similarity as it represents the ratio between viscous and inertial forces in the flow. This is, however, a simplification of what is actually a far more complex subject.

$$Re = \frac{\rho V_{\infty} l_x}{\mu} \quad (2.2)$$

Consider boundary-layer growth on a flat plate at a defined freestream velocity (V_∞), and ambient temperature and pressure (ρ , μ). The Reynolds number at any given location is then defined solely by the distance from the leading edge (l_x); thus to satisfy dynamic similarity in a freestream velocity of half the magnitude this distance would have to be $2l_x$. What is missing is that the ratio of local inertial and viscous forces vary in the wall-normal direction. The Richardson Cascade [3] shows that the large eddies that characterise turbulence are unstable and continually break into ever smaller eddies until they become so small that viscosity dissipates any remaining kinetic energy into heat. The smallest eddies form the viscous sub-layer closest to the wall. In this range, the energy input from non-linear interactions (inertial forces) and dissipation of energy through viscosity are in equilibrium. At a distance infinitely far away from the wall, the fluid can be considered inviscid. The ratio of inertial to viscous forces must, therefore, vary in the wall normal direction, not with the distance from the leading edge.

The Reynolds number is a representation of how momentum is diffused in the wall-normal direction by viscosity or turbulence, relative to the freestream flow velocity, and thus influences the thickness of the boundary layer at that point. Coming back to the original example of a flat-plate boundary layer: as the speed is increased the Reynolds number at a distance l_x from the leading edge similarly increases, which leads to the diffusion of momentum becoming relatively slower and the boundary layer becoming increasingly thinner. This causes the velocity gradient to become greater. The notion that Reynolds number is a measure of the relative importance of inertial forces and viscous forces only makes sense when applied in the indirect sense that when Reynolds number increases the boundary layer has less effect on the rest of the flow-field.

The flow characteristics often associated with increasing Reynolds number are an earlier transition point, delayed separation, thinner boundary layers, and thus thinner wakes. Al-

though such characteristics are clearly true, as a number of studies have observed this, the fundamental argument that Reynolds number is the ratio between inertial and viscous forces is misleading to the true flow physics. Taking the thinner boundary-layer characteristic as an example; this can only be true when the Reynolds number is viewed as a global parameter for the object in question, such as the total chord of an aerofoil. This is because the local Reynolds number increases with distance from the leading edge, but given that a boundary layer increases in thickness as it moves away from the leading edge this is contradictory. It is only true to say that an aerofoil operating at a higher chordwise Reynolds number will have a thinner boundary layer at a given location than the same aerofoil operating at a lower Reynolds number. To avoid confusion, the Reynolds numbers alluded to in this work are always the global Reynolds number based on chord of the wing.

Despite sometimes being misinterpreted, the influence of the global (chordwise) Reynolds number at which a body operates is of great significance, as is its use in determining aerodynamic similarity. Important aerodynamic characteristics such as laminar-turbulent transition, separation, wake thickness and, as a result, the lift and drag coefficients, are all influenced by the global Reynolds number. Transition is arguably the most important of these characteristics, as the state of the boundary layer has a direct influence on the other characteristics.

2.1.2 Natural Transition

A schematic of the natural transition process is given in Figure 2.1. The transition from laminar to turbulent flow can be seen as a stability problem in which the laminar flow is acted on by small perturbations [4].

Stability theory is based on the premise that perturbations in a laminar boundary layer are either amplified or decay in time. The flow is considered stable if the disturbances die away and unstable if they grow. Small perturbations that lie at a certain wavelength and frequency range are amplified, whereas those with smaller or larger wavelengths are damped by viscosity. Therefore, boundary-layer transition will only occur if the flow is unstable and these disturbances are allowed to grow into turbulence; this is known as the critical local Reynolds number. Once the flow reaches this point, any initial perturbations in the flow are amplified as they move downstream and manifest themselves as Tollmein-Schlichting waves. The waves may eventually grow and lead to the domination of non-linearities and consequently transition to turbulence. The phenomenon was originally discovered by Ludwig Prandtl and further studied by his two students, Hermann Schlichting and Walter Tollmien, after whom the phenomenon is named.

Figure 2.2a shows typical neutral stability curves, plotting non-dimensional wavenumber (W) against Reynolds number (Re). This demonstrates how as the Reynolds number increases the flow becomes more unstable, and if the Reynolds number is sufficiently low, i.e. below the indifference Reynolds number (Re_{ind} , or limit of stability), all perturbation modes will be damped and the boundary layer always remain stable. It also shows that higher frequency perturbations are required to make the boundary layer unstable at lower Reynolds numbers, and that at high Reynolds numbers the boundary layer will be unstable even if only perturbations with a very low wavenumber are present. Figure 2.2b shows the neutral stability curves for a boundary layer in adverse, zero, and favourable pressure gradients. This highlights how a boundary layer in an adverse pressure gradient will be much more unstable than a boundary layer in a favourable pressure gradient.

Schubauer & Skramstrad [5] detected the existence of pure laminar oscillations in the boundary layer of a flat plate using a wind tunnel with unusually low freestream turbu-

lence. The oscillations were shown to be velocity variations accompanying a wave-type motion which can be predicted by a stability theory based on exponential growth of small disturbances i.e. Tollmein-Schlichting waves. Figure 2.3 contains oscillograms at various distances from the leading edge of the plate, showing these motions. Additionally, Nishioka, et al. [6] observed signals which demonstrated transition through Tollmein-Schlichting waves, whereby a signal spike denoted locally high shearing which became more frequent downstream until an irregular fully-turbulent region developed.

Coming back to the schematic of natural transition (Figure 2.1, secondary instabilities cause the Tollmein-Schlichting waves to grow into three-dimensional Λ -vortices. These vortices then break down into turbulent spots, which eventually merge and form a fully-turbulent boundary layer. Turbulent spots were shown by Emmons & Bryson [7, 8] and Schubauer & Klebanoff [9] to appear irregularly at arbitrary positions in the boundary layer and wander downstream.

2.1.3 Separation-Induced Transition

At high Reynolds numbers the laminar boundary layer rapidly becomes turbulent due to large inertial forces, and is therefore better able to overcome adverse pressure gradients [11]. At lower Reynolds number, however, studies have shown the occurrence of a volume of slowly recirculating air known as a laminar separation bubble.

A separation bubble is formed when the laminar boundary layer cannot overcome the local flow deceleration of an adverse pressure gradient, causing it to separate. Turbulence is then generated by the growth of Kelvin-Helmholtz instabilities in the separated shear layer, due to the inflectional velocity profile giving velocity shear, which causes three-dimensional motions [12, 13]. A Large Eddy Simulation by Yang & Voke [13] confirmed

the existence of Kelvin-Helmholtz instabilities as the initiator of turbulence, as did PIV measurements by Hain, et al. [14]. Interaction between turbulent motions generated in this process and the wall prompts reattachment of the separated shear layer to the aerofoil surface, leaving an enclosed region of dead air and recirculation; the bubble. Horton [15] suggested that the turbulent mixing was important for energizing the dividing streamline in order for reattachment.

The schematic produced by Horton [15] of an averaged laminar separation bubble flow is given in Figure 2.4, with the key features labelled. This view of the bubble is, however, an artefact of the time-averaged domain in which it exists. The chaotic nature of turbulent flow, which includes eddies of varying spatial and temporal scales, means that the instantaneous flow will actually consist of flow rolling into spanwise vortices as a result of the Kelvin-Helmholtz instabilities. A comparison between the time-averaged and instantaneous flows in a separation is shown by the LES results of Yang & Voke [13], which is given in Figure 2.5. This shows the typical Kelvin-Helmholtz flow pattern as the shear layer is rolled up into vortex sheets; it is, however, the nature of these vortices in forcing flow towards the wall that leads to the closed-bubble shape in the time-averaged result (2.5a). Hot-film measurements by Zhang et al. [16] on the suction surface of a wing showed that the reattached boundary layer is not necessarily fully turbulent, instead a transitional boundary layer can reattach and continue to develop towards fully turbulent downstream.

Tani [17] offered a review of the literature on laminar separation bubbles, which contained the effect of Reynolds number on the presence of the laminar separation bubble. It was shown that in order for the bubble to form there must be an adverse pressure gradient strong enough to cause the laminar separation. However, this of course does not necessarily mean that a bubble will form as a number of factors could force the flow to transition to

a turbulent state before the separation point and thus remain attached. Such an occurrence could be due to the Reynolds number being sufficiently high; whilst, in comparison, too low a Reynolds number can cause laminar separation without subsequent reattachment as the flow does not transition in the free shear layer. This leads to the conclusion that the bubble will only form in a certain Reynolds number range and depends on a number of factors, which will be discussed later in this section. Tani [17] suggested that such a Reynolds number range is 6×10^4 to 6×10^6 . Increasing the incidence of the aerofoil causes the bubble to move forward and contract, hence promoting transition, but if incidence is increased too much the bubble may burst and cause an abrupt loss in normal force.

There can be instances, when the adverse gradient becomes too large, that even the turbulent boundary layer cannot overcome the adverse pressure gradient and the bubble bursts, forming a long bubble that only reattaches near to the trailing edge. Increasing the adverse pressure gradient further will cause complete separation of the aerofoil.

When a laminar separation bubble occurs on an aerofoil it alters the pressure distribution, giving a region of constant pressure, due to it altering the effective shape of the aerofoil. A theoretical model of a laminar separation bubble was presented by Russell [49]; this aimed to allow the determination of critical points from on-the-surface pressure measurements. As shown in Figure 2.6, the separation of the laminar boundary layer causes the pressure to remain relatively constant, since the flow no longer follows the surface curvature, until transition in the separated shear layer occurs and the boundary layer reattaches. Hence, the laminar separation bubble contains two portions; a laminar portion, which is shown by the region of fairly constant pressure, and a turbulent portion shown by a dramatic pressure rise. By this model Russell advocated that the separation and reattachment points can be found by comparing the points where the pressure distribution respectively deviates

from and reunites with its inviscid counterpart. Although this shows how the pressure distribution is affected by the bubble, it is inherently inaccurate, most notably because the inviscid and viscous solutions would not lie on top of one another.

2.1.4 Bypass Transition

The transition from laminar to turbulent flow regimes through natural transition has been described, this is known as *orderly transition*. There exists another type known as *bypass transition*, this is for all other routes which deviate from the orderly, however it is usually synonymous with the boundary layer being exposed to vortical forcing from the freestream, or forced-transition mechanisms such as vibration, or boundary-layer trips. As the name suggests, the first few stages of natural transition are bypassed and turbulent spots are directly produced in the boundary layer. This makes linear stability theory irrelevant for bypass transition. As Tollmein-Schlichting waves have not been documented at freestream turbulence levels greater than 1%, it can be assumed that this is the boundary between natural and bypass transition mechanisms [18]. A study by Brandt et al. [19] used numerical simulation to show that when the freestream turbulence levels were greater than 1%, transition occurred quickly and bypassed the classic scenario of instability amplification.

Bypass transition is characterised by the appearance of streamwise elongated streaky structures of alternating high and low velocity. These streaks then proceed to break down into turbulent spots and finally a fully-turbulent boundary layer. These structures were first noticed by Klebanoff [20]. Arnal & Juillen [21] observed that for freestream turbulence levels of 0.5-1% the dominant disturbances within the boundary layer were not Tollmein-Schlichting waves; Kendall [22] denoted these disturbances as being Klebanoff modes. Further study on Klebanoff modes was completed by Wu & Choudhari [23].

A review paper by Zaki [24] used both experimental and numerical models to explain bypass transition. He showed that high-frequency perturbations are blocked by the boundary layer while low-frequency disturbances could penetrate it. Furthermore, two types of instability are possible, an outer instability that arises near to the boundary layer edge, and an inner one near to the wall.

2.1.5 Relaminarisation

Also known as reverse transition, relaminarisation is the transition of turbulent to laminar flow. This occurs due to high flow acceleration, such as in favourable pressure gradients. Such regions generally occur on an aerofoil close to the leading edge of a suction surface, and near to the trailing-edge of the pressure surface. Other possible causes of relaminarisation, as noted by Narasimha & Sreenivasan [25], are due to turbulence dissipation, stably stratified flows, thermal effects, and surface-mass transfer (air blowing or suction).

2.1.6 Influential Parameters

The range of transition mechanisms have been discussed, and all are clearly affected by the environmental conditions around them. The stability of the laminar boundary layer, due to the amplification or suppression of small disturbances, is directly influenced by a number of aspects.

Freestream Turbulence Level

Increasing freestream turbulence promotes transition due to the increase of perturbations in the freestream flow, which then grow in the boundary layer. Moreover, increasing freestream turbulence causes other effects, such as pressure gradient, Reynolds number etc., to be less influential [26].

Bennett [10] observed boundary layer transition on a flat plate with zero pressure gradient, using pressure tubes and hot-wire anemometry, at varying freestream turbulence levels. Results showed that the oscillations predicted by laminar stability theory, which is outlined in [4], are an important mechanism in transition. Higher levels of freestream turbulence were shown to hasten the transfer of energy from an amplified instability frequency to the rest of the spectrum; while lower turbulence levels saw fluctuations occur at a concentrated frequency and the transition region, as a consequence, covering a greater distance along the flat plate.

Pressure Gradient

Both favourable and adverse pressure gradients have an effect on transition. Watmuff [27] used flying hot-wire techniques to observe a separation bubble, which appears on a flat plate due to an adverse pressure gradient created by a controlled ceiling, at a Reynolds number of 4.68×10^5 . Watmuff found that the increase in disturbances in the separated shear layer, which is the mechanism for transition, is due to inviscid Kelvin-Helmholtz instabilities. Lian & Shyy [28] altered the incidence of an aerofoil to alter the magnitude of the adverse pressure gradient. As the incidence was increased both the transition and separation points moved upstream due to the larger adverse pressure gradient amplifying the unstable Tollmien-Schlichting waves that excite transition phenomena and cause the bubble to become shorter and thinner.

Abu-Ghannam & Shaw [26] detailed empirical relationships, which were a development from those of Hall & Gibbings [29], for the prediction of the start and end of the transitional region of a flat plate boundary layer under a range of freestream turbulence levels and pressure gradients that were validated against experimental results. Transition on the plate was detected using china-clay visualisation, surface pitot tube and hot-wire

anemometry in the boundary layer. On a flat plate with no pressure gradient the flow was found to transition at a lower Reynolds number as the freestream turbulence was increased, i.e. higher turbulence promoted transition. Under an adverse pressure gradient, the freestream turbulence was supported and transition occurred earlier whereas a favourable pressure gradient opposed and delayed transition. However, with transition described by the Reynolds number based on momentum thickness (rather than a streamwise distance), then at a constant freestream turbulence level the effect of an adverse pressure promoting transition is much greater than the influence of the favourable gradient retarding it. Additionally, it was shown that increasing freestream turbulence dampens the effect of pressure gradients on delaying or advancing the location of transition.

Reynolds Number

At lower Reynolds numbers, where viscous effects dominate, the damping action of viscosity aids in the suppression of perturbations. At higher Reynolds numbers this damping is not sufficient and disturbances are allowed to grow and intensify; thus promoting transition. Hot-wire anemometry measurements by Yoshioka et al. [30] showed that as Reynolds number increased the transition point moved upstream on an SD7003 aerofoil.

Mabey [31] offered a review into scaling effects that included flat plates and aerofoils. He stated that there are two origins of scaling effects: not achieving the full-scale Reynolds number and failing to achieve full-scale transition position. The effects of these are more pronounced on the adverse pressure region of the suction surface where the boundary layers are thicker. Mabey suggested that failure to fix transition at the appropriate position is much more serious than operating at an incorrect Reynolds number. Results showed free-transition scale effects can be very large and non-monotonic while fixed transition confines these effects to the regions close to the onset of separation.

An investigation into the response of a laminar separation bubble to Reynolds number changes and artificial disturbances was completed by Diwan & Ramesh [32] on a flat plate in a wind tunnel of freestream turbulence $< 0.03\%$ with a contoured wall giving an adverse pressure gradient. It was found that as the Reynolds number was decreased both the length and height of the bubble increased, with height increasing at a greater rate and thus reducing the aspect ratio of the bubble.

Surface Curvature

Two companion papers by Hoffman et al. [33] and Muck et al. [34] observed experimentally the effects of concave and convex curvature, respectively, on a turbulent boundary layer using conventional hot-wire anemometry techniques. The effects of each curvature were shown to be separate phenomena whereby concave curvature had a destabilising effect and convex curvature a stabilising effect. Whilst convex curvature tends to attenuate pre-existing turbulence and, in the case of mild curvature, not alter the statistical average eddy shape. On the other hand, concave curvature led to the production of quasi-inviscid, longitudinal Görtler vortices. Significant changes to the turbulent structure occurred due to both the curvature and the vortices. This significantly enhances mixing and increases skin friction compared to a flat plate boundary layer [35].

Surface Roughness

Premature transition may be promoted by any mechanism that artificially introduces perturbations into the flow with a view to increasing momentum thickness and, thus, promoting turbulence production. The action of forcing transition from a laminar to a turbulent boundary layer is common during wind tunnel testing to eliminate the later transition point caused by testing at reduced Reynolds numbers [31]. It is also, however, used in a number of more common applications. The most famous of these is golf balls; since the

mid-17th century golfers have known that a ball with roughness or indentations could fly further and in a more controlled manner. This is due to the indentations, or the dimples of the modern ball, generating disturbances and promoting transition to a turbulent boundary layer which remains attached longer than its laminar counterpart. The result of this is a thinner wake and a smaller pressure difference between the fore and aft regions of the ball. In addition to golf, sports such as cycling, speed skating and sailplanes (in the form of turbulators) also use methods to promote transition in order to gain turbulent boundary layers.

Klebanoff & Tidstrom [36] performed an experimental investigation into two-dimensional roughness elements on a flat-plate boundary layer. Transition was measured by taking oscillograms of the longitudinal component of velocity fluctuation at Reynolds numbers 3.54×10^4 to 4.33×10^4 using a hot-wire placed 1.5 mm from the surface. The authors concluded that the effect of the roughness elements could be considered to be a stability-governing phenomenon; meaning that they reduced the stability of the boundary layer and hence promoted instabilities that cause the production of turbulence. Depending on the magnitude of destabilisation, transition could occur much further downstream or directly after the roughness element. Transition appeared to be dependent both on the velocity profile that was generated and on whether the Reynolds number was sufficiently low that there was no amplification of perturbations.

Marzabadi & Soltani [37] looked at the effect of surface roughness on boundary-layer transition. Despite the application of the work being wind-turbine blade sections, the static test results are directly transferable to any sector using aerofoil geometries. At a chord-based Reynolds number of 4.2×10^5 eight hot-films positioned along the chord acquired shear stress on the wall, which could be used to derive the state of the boundary layer. Both an increase in incidence and surface roughness could cause the transition

point to move upstream towards the leading edge, and promoted turbulent trailing-edge separation in the static condition.

A numerical investigation by Gaster [38] showed that the mean flow velocity over a periodic array of ridges remained unchanged as it was shifted up and down by the ridge. It was shown that the amplification of Tollmien-Schlichting waves occurred, which promoted transition to turbulence. The movement of the mean velocity profile shows that the surface roughness is introducing wall-normal velocity components into the flow and thus destabilising the flow, such that the fluctuating velocity components that characterise turbulent flows are produced.

Surface Temperature

The act of heating or cooling a surface has an effect on a boundary layer's stability. Heat transfer from the wall (heating) is destabilising and thus hastens the onset of transition. Heat transfer from the boundary layer to the wall (cooling), has the opposite effect and acts to stabilise the boundary layer. Linke [39] observed this effect by measuring the friction drag on a hot plate; when the plate was heated the friction drag increased, thus showing that a turbulent boundary layer covers a larger portion of the plate as a turbulent boundary layer gives higher friction drag.

The effect of surface temperature is a direct result of the dependence of viscosity on the temperature. Liepmann & Fila [40] used a heated plate to show that increased wall temperature hastened transition. They concluded that the effect of surface temperature is similar to the effect of a pressure gradient, in which a heated wall destabilises the flow over it and a cooled wall stabilises it.

2.1.7 A Transition-Sensitive Eddy Viscosity Closure Model

Reynolds-averaged Navier-Stokes (RANS) is the most commonly employed CFD method because of its fast solving time and robust solution. It is considered the workhorse of industry. The formulation of RANS, however, means that it can never correctly model turbulent flows. Once the Navier-Stokes equations are decomposed into mean and fluctuating components and averaged we are left with more unknown quantities than equations. This is known as the closure problem, and thus the RANS equations cannot be solved without further assumptions, which is why turbulence models are required. This problem is addressed using the assumption that turbulence can be considered as an additional viscosity: the eddy viscosity. The momentum transfer caused by turbulent eddies is modelled by the eddy viscosity; thus the Reynolds stresses are proportional to the mean rate of deformation. Further information can be found in Versteeg [41].

The classical turbulence models, such as Spalart-Allmaras, $k - \varepsilon$, and $k - \omega$, are all considered fully-turbulent i.e. no laminar effects are included. Clearly, for low Reynolds number flows where laminar boundary layers are important this is an invalid assumption. Although methods such as Large Eddy Simulation (LES) and Direct Numerical Simulation (DNS) already correctly solve turbulence (though LES still models the smallest turbulence scales) such that laminar and transition effects can be predicted, the required mesh density and solving time mean that they can be inappropriate for industrial flows. This has led to the formulation of transition-sensitive closure models.

An initial attempt to incorporate laminar conditions was the 'Low Reynolds Number' approach, where extra damping functions were included for the near-wall region. These models predict transition through diffusion of freestream turbulence into the boundary layer. As the work of Savill [42, 43, 44] showed, however, even with carefully chosen damping functions the models still struggle to predict transition. Another method was

to use empirical correlations in order to calculate an intermittency factor and determine the onset of transition. The $\gamma - Re_\theta$ model of Menter et al. [45] has shown good performance and as such has been included into the commercial solver STAR-CCM+. It is not, however, available in the standard release of FLUENT. The main disadvantage of this approach, however, is its reliance on correlations. As transition can be affected by such a large range of factors, which were discussed in the previous section, creating a correlation-based model which is applicable to all cases seems difficult, if not unrealistic.

In order to reduce the reliance of a solution on empirical correlations, it was suggested that modelling of an additional physical phenomenon for the laminar flow was necessary. Walters & Cokljat [46] devised a purely physics-based eddy viscosity model, known as $k - k_L - \omega$, based on the $k - \omega$ framework and the concept proposed by Walters & Leylek [47]. The model employs three transport equations for turbulent kinetic energy (k), laminar kinetic energy (k_L), and the scale-determining variable, also interpreted as the specific dissipation rate (ω). The additional transport equation, compared to the typical two-equation $k - \omega$ models, is included to model pre-transitional phenomena such as Klebanoff modes and Tollmein-Schlichting waves. An additional feature of the $k - k_L - \omega$ model is that it includes anisotropic dissipation constants.

The model is based on the assumption that velocity fluctuations prior to transition can be divided into two parts: small vortices that contribute to turbulence production, and large longitudinal vortices near to the wall contributing to the production of non-turbulent fluctuations. The near-wall turbulent kinetic energy is similarly divided into two parts: small-scale vortices that contribute to the production of turbulent energy, and large-scale vortices that contribute to the production of laminar kinetic energy.

The first two transport equations account for the production of turbulent (Eqn. 2.3) and

laminar (Eqn. 2.4) kinetic energy respectively through both the amplification of low-frequency disturbances in the boundary layer and the selectivity of the boundary layer to freestream eddies. This enables the model to be capable of predicting both natural and bypass transition mechanisms. Additionally, as a laminar boundary layer is modelled, this can separate and lead to separation-induced transition. The model characterises the transition process as the transfer of laminar kinetic energy into turbulent kinetic energy due to increasing pressure-strain terms. The initiation of this process is based on the restriction of turbulence damping functions and growth of these pressure-strain terms in the Reynolds stress terms. Laminar kinetic energy represents the growth of disturbances in the pre-transitional boundary layer. After transition, this damping is restricted only to the viscous sub-layer, such that it remains effectively laminar.

$$\frac{D(k)}{Dt} = \frac{\delta}{\delta x_j} \left[\left(v + \frac{\alpha_T}{\sigma_k} \right) \frac{\delta k}{\delta x_j} \right] + P_{k_T} + R_{BP} + R_{NAT} - \omega k - D_T \quad (2.3)$$

$$\frac{D(k_L)}{Dt} = \frac{\delta}{\delta x_j} \left[v \frac{\delta k_L}{\delta x_j} \right] + P_{k_L} - R_{BP} - R_{NAT} - D_L \quad (2.4)$$

$$\begin{aligned} \frac{D\omega}{Dt} = & C_{\omega 1} \frac{\omega}{k} P_{k_T} + \left(\frac{C_{\omega R}}{f_W} - 1 \right) \frac{\omega}{k} (R_{BP} + R_{NAT}) - C_{\omega 2} \omega^2 \\ & + C_{\omega 3} f_{\omega} \alpha_T \frac{f_W^2 \sqrt{k}}{Y^3} + \frac{\delta}{\delta x_j} \left[\left(v + \frac{\alpha_T}{\sigma_{\omega}} \right) \frac{\delta \omega}{\delta x_j} \right] \end{aligned} \quad (2.5)$$

The third transport equation (Eqn. 2.5) represents the scaling variable (ω). The first, third and fifth terms of the equation represent the production, destruction and gradient transport terms, and are analogous to the similar terms of the k_L and k_T equations. They are also

similar to the terms that appear in other models based on the $k - \omega$ framework.

The advantages of transition-sensitive models are not limited to boundary-layer formation, however, as laminar flow is also important for vortex structures. Although vortices are typically associated with the turbulent wake of a racing car, the high swirl velocity has a stabilising effect that tends to overcome the destabilising axial velocity deficit and leads to the laminarisation of the core. Therefore, whilst a vortex core may begin turbulent due to the rollup of a turbulent boundary layer, the turbulence decays rapidly due to the swirl component. Experimental measurements of Reynolds stress with a hot-wire probe by Devenport et al. [48] showed the first conclusive evidence of a laminar vortex core. A typical deficiency of classical RANS methods is the over-estimation of total pressure loss in the vortex core, and over-estimation of vortex dissipation or burst as a result of the dissipation terms, which are included to model losses due to the fluctuating velocity components of turbulent flow, leading to vorticity being underestimated. It is, therefore, reasonable to assume that a model that includes laminar flow will produce more realistic vortices.

An important aspect of turbulence modelling is how it models the near-wall flow. There are two approaches: the first is to resolve the entire boundary layer, and the second is to use wall functions. The two approaches can be described by considering the regions of a turbulent boundary layer in terms of the dimensionless wall distance, Y^+ (Equation 2.6, Figure 2.7). Y^+ is the distance from the wall non-dimensionalised by the quantities that dominate the local flow physics: viscosity and shear stress.

$$Y^+ = \frac{U_* y}{\nu} \quad (2.6)$$

$$U_* = \sqrt{\frac{\tau}{\rho}} \quad (2.7)$$

The wall function approach uses a coarser grid, in which only a single cell houses the laminar (viscous) sublayer and buffer layers. Instead, the boundary layer is modelled using the law of the wall, which states that the average velocity of a turbulent boundary layer is proportional to the logarithm of distance from the wall (log law). Resolving the boundary layer, on the other hand, involves placing sufficient grid points in the boundary layer such that the velocity profile, down to the viscous sublayer $Y^+ \leq 5$, is accurately predicted. As the viscous sublayer exhibits laminar characteristics, its presence is important for flows involving adverse pressure gradients and separation. The wall function approach is more appropriate for geometry-induced separated flows, such as around bluff bodies, where the boundary layer is not important. This is because the wall function approach will generally under-estimate separation as it predicts a higher velocity in the near-wall region, as inflected velocity profiles are not accounted for. In order to resolve the boundary layer the first cell should be small enough to capture the viscous sublayer, hence $1 \leq Y^+ \leq 5$ is typically employed. However, target values of $Y^+ \leq 1$ are also common as they provide better resolution of the boundary layer but also, in the $k - k_L - \omega$ case, are capable of capturing the laminar sub-layer.

2.2 Transition on Aerofoils

The importance of boundary-layer transition on the performance of aerofoils has been observed extensively in the aeronautical sector. The key differences between an aerofoil section for aeronautical use and a wing operating in ground effect for motorsport applications is that the latter generates significantly larger pressure gradients, is generally of a

lower aspect ratio, and has highly complex, three-dimensional flow features. Moreover, the difference between full-scale and model-scale Reynolds numbers for motorsport applications, which is in the order of 10^5 , is much smaller than in aeronautical applications where the difference can be in the order of 10^6 . As such, the flow characteristics of a model-scale racing car model are more representative of its full-scale counterpart than an aeronautical application such as a large aircraft. Understanding transition for an aerofoil section out-of-ground-effect is, however, the sensible starting point for this work.

An aerofoil generates force by imparting curvature upon the streamlines to generate a pressure difference between the upper and lower surface. The pressure gradient along the wall in the chordwise direction, as flow is accelerated to generate a lower pressure and then decelerated to recover back to freestream pressure, has a considerable influence on the boundary-layer state. In regions of accelerating flow (favourable pressure gradient) the boundary layer will generally remain laminar, whereas even a weak decelerating flow could trigger transition [4].

Typically on the suction surface of an aerofoil, the flow is accelerated in the fore-region and decelerated in the aft-region. The acceleration has a stabilising effect such that the laminar boundary layer is maintained, but the deceleration then destabilises the boundary layer and promotes turbulence production. Natural transition is, however, not necessarily a fast process as disturbances need time to grow and, thus, the sudden deceleration can cause the laminar boundary layer to separate before transition can occur. This leads to the formation of a separation bubble.

Altering the angle of incidence alters the pressure gradients on the aerofoil surface, and thus has an effect on the transition mechanism. Using the criteria outlined by Russell [49], the locations of separation, transition and reattachment for a separation bubble on an

out-of-ground-effect GA(W)-1 aerofoil at varying incidence and chord-based Reynolds number of 4×10^4 were determined by Hu & Yang [52] using averaged and instantaneous PIV. As incidence was increased separation, transition and reattachment points all moved upstream towards the leading edge in such a manner that the overall bubble length remained constant i.e. independent of incidence. However, the laminar portion of the bubble increased in size, and the turbulent portion contracted, as incidence was increased. It was found that the bubble only existed for $8^\circ < \alpha < 11.5^\circ$, above this range the aerofoil was seen to have stalled and below this range the adverse pressure gradient was not strong enough to cause the laminar boundary layer to separate. Similar conclusions were drawn by Genc et al. [54] when observing the effect of incidence on a NACA2415 aerofoil for a chord-based Reynolds number range of 5×10^4 to 3×10^5 .

In addition to incidence, factors such as Reynolds number and ground proximity also affect the pressure distribution, and thus transition characteristics. Ghadimi et al. [55] found, using a method that combines the vortex panel method and a boundary layer method, that both the transition and separation points, and boundary layer thickness, were affected by changes to either incidence, Reynolds number or ground clearance. Increasing Reynolds number, incidence and ground proximity all brought the transition point upstream towards the leading edge. The separation point was advanced (moving towards the leading edge) by increasing incidence and ground proximity, due to the larger adverse pressure gradient the air flow had to overcome, but was delayed by increasing Reynolds number, as the flow had more energy to overcome the adverse pressure gradient. While these were observed for an uplift-producing aerofoil in ground effect, the trends should be comparable for a downforce-producing aerofoil given that in both cases an increased incidence or reduced ground clearance (lower h/c) both increased the suction peak.

In the studies discussed above, the location of transition has been cited as a discrete point.

Flow does not, however, instantaneously transform from smooth laminar flow to chaotic turbulent flow, but instead turbulence grows over a period of both space and time. Hence, it is unrealistic to state a transition *point*, as clearly defining the point at which transition starts and ends is somewhat arbitrary. Hot-film anemometry by Zhang et al. [16] showed that the boundary layer is not necessarily fully turbulent upon reattachment to the aerofoil surface, and instead may still be in a transitional state. Power Spectral Density (PSD) results showed that instabilities began to form before separation, transition phenomena then rapidly grew in the free shear layer, which caused the boundary layer to reattach. Initial high amplitude peaks were observed at a frequency in the same order as Tollmien-Schlichting waves for that corresponding momentum thickness (based on a correlation by Warren & Hassan [56]). The favourable pressure gradient meant that boundary layer momentum thickness remained almost constant as it moved downstream meaning that the Tollmien-Schlichting frequency also remained constant rather than increasing as would be expected. Despite it being erroneous to term transition as being at a discrete point, as a starting point, Russell's model (Figure 2.6) is a good generalised picture of the influence of a separation bubble.

PIV measurements of a laminar separation bubble forming on an SD7003 aerofoil at very low Reynolds numbers ($2 \times 10^4 < Re < 6 \times 10^4$) were conducted by both Burgmann [50] and Hain et al. [14]. The results clearly showed that it is Kelvin-Helmholtz instabilities that lead to vortex formation in the separated shear layer of a laminar separation bubble [14]. When this vortex grows too strong and large in size it reaches a critical state that causes an abrupt burst and consequently a large ejection of fluid into the mean flow. The three-dimensional vortices are half-moon, or horseshoe, in shape and cause a strong vertical motion of flow away from the wall [50].

Overall, it has been conclusively shown that transition occurs on aerofoils through a lam-

inair separation bubble, providing that a bypass transition has not occurred a priori. Although the fundamental flow physics of a separation bubble are understood, the fact that the majority of experiments cannot provide synchronised spatial and temporal measurements means that there is not complete understanding of the finer flow structures present in the transition and reattachment phases.

2.2.1 Use of the $k - k_L - \omega$ Closure Model

Motives for the development of a purely physics-based transition-sensitive closure model for use in RANS computations were outlined above in Section 2.1.7. The $k - k_L - \omega$ model was produced to satisfy those needs. In this section its use in predicting transition on aerofoils is examined.

When Walters & Cokljat [46] originally outlined their closure model, which was eventually implemented in FLUENT, they presented a number of test cases. These included a number of aerofoil cases; the most relevant of those to this work was that of an Aerospaiale A-airfoil configuration operating at $Re = 2 \times 10^6$, $M = 0.15$, and $\alpha = 13.3^\circ$. The skin-friction distribution on the suction surface predicted by CFD was compared to that measured in experiments (Figure 2.8). The authors cite that the model also very closely matched the transition location of $x/c = 0.12$ found in experiments through flow visualisation. Downstream of $x/c = 0.2$ where the CFD and experiments can be directly compared in terms of skin friction, it can be observed that the model underestimates skin friction initially, but gives better correlation downstream.

The $k - k_L - \omega$ model was tested in FLUENT, along with other models, by Genc [57] to calculate flow over a NACA64A006 thin aerofoil at $Re = 5.8 \times 10^6$. Of the tested models, $k - k_L - \omega$ performed the best. At lower incidence the model recreates the pres-

sure distribution very closely, however at $\alpha = 9^\circ$ there is substantial disagreement as the model greatly over-predicts suction. Comparison of the downforce vs. incidence curve would lead to the conclusion that the model is very accurate across the entire range $0^\circ < \alpha < 10^\circ$. It is, however, coincidental as at $\alpha = 9^\circ$ it is shown that the model substantially over-predicts close to the leading edge, and substantially under-predicts near to the trailing-edge such that they equal out to give a good, but coincidental, prediction of downforce.

Numerical analysis of six two-dimensional aerofoil profiles with the transition-sensitive $k - k_L - \omega$ and fully-turbulent $k - \omega$ SST models at a range of Reynolds numbers from 6×10^4 to 2×10^6 was conducted by Chitta et al. [58]. It was observed that the fully-turbulent nature of the $k - \omega$ SST model meant that no transition mechanism was predicted as the entire aerofoil was covered by a fully-turbulent boundary layer, unlike the $k - k_L - \omega$ model which was capable of accurately capturing the separation bubble.

Comparisons between two transition-sensitive models, $k - k_L - \omega$ and a transition-sensitized $k - \omega$ SST, were made by Dhakal et al. [59] for a PAK-B aerofoil cascade at $2.5 \times 10^4 < Re < 3 \times 10^5$. They found that $k - k_L - \omega$ was able to produce reasonable results for low freestream turbulence intensity cases, whilst the transition-sensitized $k - \omega$ SST model was better in the high freestream turbulence intensity cases. It was noted that both models predicted a separation point that was almost independent of Reynolds number.

The XFOIL and MSES codes were used in conjunction with the $k - k_L - \omega$, and implemented in the solver OpenFOAM, by Eisele & Pechlivanoglou [60] to evaluate single- and multi-element wing configurations. As XFOIL does not allow for multi-element configurations it was only used for the single-element tests. XFOIL and MSES predicted similar lift and drag; compared to the experiments drag was significantly underestimated,

but lift and stall characteristics were accurately captured. All of the RANS models tested, $k - k_L - \omega$, $k - \omega$ SST and Spalart-Allmaras, showed similar trends. Although pre- and post stall behaviour for both lift and drag was in good agreement with the experiments, the maximum lift coefficient was under-predicted.

In the studies discussed above it has been shown that the $k - k_L - \omega$ model is capable of giving accurate predictions of separation-induced transition on an aerofoil operating out of ground effect. These were all two-dimensional cases, but turbulence is a fundamentally anisotropic, three-dimensional mechanism.

2.3 Wings in Ground Effect

In this section, the fundamental characteristics of wings operating in ground effect to produce downforce are discussed. Although there are a number of other studies, not discussed here, that have utilised a wing in ground effect, their conclusions are not directly related to the work in this thesis or they are based on specific applications.

Ever since the late 1960's racing cars have made use of inverted wings to increase downforce and improve cornering grip. Initially, these were mounted high above the cockpit, but designers quickly realised that having a wing at the front and another at the rear gave the best configuration in terms of absolute downforce and managing the centre of pressure. Structural failures and accidents led to rule changes that prohibited high-mounted aerofoils and, in turn, led to wings being closely integrated into car structures. This produced front wings that operated in close proximity to the ground.

It was not until the early 1990s, however, that published studies into the characteristics of inverted wings in ground effect were produced in earnest. A review into Grand Prix racing-car aerodynamics by Dominy [61] explained that a wing in ground effect would generate a larger suction peak than its freestream counterpart due to the constraining of flow. The area confined between the wing and ground results in an increase in the fluid velocity, a requirement to satisfy continuity. He also postulated that the adverse pressure gradient produced by such a large suction would cause the boundary layer on the wing to separate; this can be seen as being similar to increasing the incidence of an aerofoil.

Whilst Dominy's paper was important for defining the expected flow physics of a wing in ground effect, he offered only a pressure distribution for an unspecified aerofoil with a slotted flap and Gurney flap, in and out of ground effect. An experimental study by Knowles et al. [62] two years later provided the first evidence of how ground effect al-

tered the pressure distribution of a GA(W)-1 wing. Ground clearances between $h/c = 1.0$ and $h/c = 0.12$ were tested and showed that as the wing moved closer to the moving ground the wing's lift curve slope increased and the stall angle reduced. They attributed the increase in lift to a Venturi effect caused between the ground and the suction surface of the wing, thus agreeing with Dominy's original conjecture. The experimental work was supported by 2D panel method computations; although the same trends were observed in both experimental and computational results, the panel method significantly over-predicted suction values by comparison with the (3D) experiments.

A series of studies by Ranzenbach & Barlow [63, 64, 65, 66] used both experimental and computational approaches for both single- and multi-element wings. Although a moving ground was employed for the computations, the experiments were conducted in a fixed-ground facility. The lack of a moving ground results in a boundary layer growing on it; although this also happens underneath the wing as a result of higher-than-freestream velocity, in a fixed-ground facility a boundary layer will be present before the wing and thicker when underneath the wing. Marshall [67] showed that a wing was considerably affected by the thickness of the boundary layer on the ground underneath it up to $h/c = 1.0$. LDA measurements of the wake showed that at lower ground clearances the wake is entrained into the boundary layer, which leads to a greater velocity deficit in the wake and premature breakdown of the wake. This shows that although it is inaccurate to use a fixed ground, the trends that such experiments show can still give valid information. In their studies Ranzenbach & Barlow used multiple aerofoil profiles at constant incidence and varying ground clearance. Similarly to the observations of Knowles et al. [62], Ranzenbach & Barlow observed that the downforce increased as the ground clearance was reduced. A critical ground clearance was found, however, beyond which any further reductions in ground clearance results in a decrease in downforce. Ranzenbach & Barlow

attributed this to the merging of the boundary layers from the wing's suction surface and the ground, causing a reduction of the flow velocity. This critical ground clearance was found to be geometry dependent and, moreover, increased (occurred when the wing was further away from the ground) when a more cambered wing was used. The authors termed this the "force reversal" phenomenon.

Four configurations of three-dimensional, double-element wings were tested, both in and out of ground effect, with a fixed-ground facility at varying Reynolds number, incidence and flap position by Jasinski & Selig [68] with force measurements and a wake study with a 7-hole pressure probe. It was shown that as Reynolds number increased downforce would also increase. Drag reduced due to profile drag reducing at a greater rate than the increase in induced drag. As would be expected, an increase in flap deflection or flap planform area resulted in higher downforce levels. The effect of endplates was also observed, with larger endplates making the wing more efficient.

The most widely-cited studies of wings in ground effect were those conducted at the University of Southampton, where a number of authors contributed to the field. The first paper by Zerihaan & Zhang [69] reported force and pressure measurements of a Tyrrell-026 (which is similar to the GA(W)-1, but more fore-loaded) single-element wing at varying incidence and ground clearances. As with previous studies they found an increase in downforce and drag with reduced ground clearance, until again a critical ground clearance was reached, below which downforce decreased and drag continued to increase. These regions were termed the force-enhancement region and force-reduction region, respectively, and this appears to have become the standard terminology. The surface pressure measurements showed that trailing-edge separation occurred in the force-reduction region, with the separation point moving upstream as the wing was lowered. This led to the authors' conclusion that merging boundary layers, as Ranzenbach & Barlow [63] had

proposed, was incorrect and that the actual cause was separation due to the large pressure gradient. This confirms, therefore, the original explanation given by Dominy [61] eight years before. Somewhat similarly to Ranzenbach & Barlow, Zerihaan & Zhang found that increasing incidence to generate more downforce caused the force-reduction region to occur at higher ground clearances. Two Reynolds numbers were trialled and it was shown that the higher Reynolds number case led to maximum downforce being produced at a lower ground clearance. It should be noted that the uncertainty in the force results was calculated solely as a function of the standard deviation in repeated tests. Errors from other sources were not quantified.

During their experiments Zerihaan & Zhang [69] made use of both free transition and fixed transition, concluding that fixing transition caused the boundary layer to be thicker at the start of the adverse pressure gradient and thus make separation more likely; this causes circulation to be lost and downforce to be reduced as a result. However, given that only a single transition location was used, at 10% chord, it cannot be stated definitively that in all cases in which fixed transition is used performance will be diminished without further testing of multiple transition locations and Reynolds numbers. Force balance results for the fixed-transition cases were stated by the authors to be less repeatable than the free-transition tests.

In a follow-up study by Zhang & Zerihaan, the mean and instantaneous wake behind the single-element wing was found using LDA and PIV techniques [70]. The instantaneous flow was shown to consist of discrete, alternate vortex shedding in which the vortices become larger as the wing moves closer to the ground. In the mean flow the velocity deficit in the wake and boundary layer thickness increases with decreasing ground clearance due to small-scale turbulence. At ground clearances so low that separation occurs on the wing's suction surface the amplification of instability waves was found to lead to non-

linear rollup of shear layers i.e. large vortices. Zhang & Zerihaan subsequently focussed on the wing-tip vortices produced by the wing, using LDA, PIV, surface pressures, force measurements and surface flow visualisation [71]. Results showed that the wing's downforce was affected both by the tip vortices and separation at the trailing edge. At higher ground clearances a concentrated tip vortex was produced off the inner-lower edge of the endplate on the wing through separation, contributing to the downforce enhancement process. However, as the wing moved closer to the ground vortex breakdown occurred, reducing the force-enhancement mechanism.

With both the on- and off-surface characteristics of a single-element wing clearly defined, Zhang & Zerihaan [72, 73] completed two further studies on a multi-element configuration with a flap of equal span to the main plane. It was found that the main plane produces the majority of downforce and dominates wake development. The results found were mostly in line with the characteristics presented by Smith [74], whereby the overall mechanism of the flap is to enhance the performance of the main plane through a number of effects. By increasing the flap incidence for an unchanged main-plane incidence, the force-enhancement and force-reduction regions were entered at a higher ground clearance. The authors attributed this to increased circulation forming a stronger vortex, and earlier separation due to a larger adverse pressure gradient. The overall effect of the flap on the pressure distribution was to increase suction, due to a diffuser-type effect increasing mass flow under the wing, and to increase pressure, due to the slat effect, on the main plane, which along with the downforce of the flap itself resulted in much greater total downforce than a single-element wing.

The multi-element wing produced significantly more downforce such that stronger wing-tip vortices were observed and were shown to have a significant effect on the downforce vs. ground clearance curve (Figure 2.9). The state of the wing-tip vortex allowed the

force-enhancement region to be broken down into three sections:

- **Region 1: force enhancement and concentrated vortices.** Both the constraining of flow under the wing and suction from the tip vortex contribute to downforce. Downforce continues to increase as ground clearance reduces.
- **Region 2: force enhancement with vortex breakdown.** Due to the breakdown of the tip vortex there is a small dip in downforce initially, however downforce continues to increase after this as ground clearance reduces due to the large suction generated inboard.
- **Region 3: force-reduction region.** The downforce-reduction region can thus be attributed solely to boundary layer separation as flow can no longer overcome the adverse pressure gradient. Downforce decreases and drag increases continually as the ground clearances further reduces.

The tip vortices also have the effect of constraining the flow due to an inboard cross-flow as a result of the pressure gradient spanwise across the main plane. The flap on the other hand features outboard cross-flow as a result of the main tip vortex direction.

Tip-vortices from multi-element wings were also examined by Moseley [75], Garrood [76] and Pegrum [77] at Imperial College, London. Moseley [75] took off-surface measurements behind a double-element wing and noticed that, similarly to Zhang & Zerihan [72], that vortex breakdown occurred at lower ground clearances. Vectors of instantaneous velocity which he obtained from PIV led him to the conclusion that the vortex underwent a spiral-type breakdown. Garrood [76] used laser-smoke flow visualisation to observe the difference in tip-vortex formation when a moving ground was used, as opposed to a fixed ground. For the fixed-ground case a distinct core could be identified where the smoke particles had been centrifuged out. No core was observed when the moving ground was then enabled. He found that breakdown occurred at $h/c = 0.29$ with the moving ground,

however the phenomenon was not present above a fixed ground. Pegrum continued this investigation by observing the vortex system behind a McLaren front wing with and without a moving ground. This vortex system is shown in Figure 2.10. Pegrum [77] did not observe the same phenomenon, and instead found that the vortex centres were largely unaffected by the ground being stationary or moving. The strength of the vortices, however, was found to be lower for the fixed ground case; this was attributed to the presence of the boundary layer that grew on the fixed ground. In addition to investigations of the influence of the moving ground, Pegrum revealed the vortex system behind a highly-complex wing (McLaren Formula One wing). The strength of each vortex was individually dependent on ground clearance i.e. some would strengthen whilst others would weaken as ground clearance was reduced.

Galoul & Barber [78] completed an experimental investigation, using LDA, into the effect of endplates. Two vortices were present from the wing tip: one from the lower edge and one from the upper edge. The lower vortex was the stronger of the two and as a result it altered the trajectory of the upper endplate vortex. The two vortices were found to merge between one and two chord lengths downstream. Although this result will have a strong geometry dependence, especially that of the endplates, the notion that the stronger, and thus lower pressure, vortex core manipulates the weaker vortex is important.

The influence of geometry was examined by Vogt & Barber [79] by comparing the Tyrrell-026 wing used by Zerihaan & Zhang [69], to a NACA 4421 profile through RANS computations with a Reynolds-Stress Model (RSM). In terms of geometry the Tyrrell wing is similar to the NACA 4412, as it was based upon it. Development by Tyrrell, however, led to it being more cambered near to the trailing edge, having an apex closer to the leading edge and a leading edge positioned higher. Results showed that the Tyrrell wing generated more downforce, as it made better use of the diffuser effect and thus generated more

mass flow through the channel between the ground and aerofoil. These results are not unexpected given that, overall, the Tyrrell wing is more cambered, thus imparting more curvature to streamlines and, as a result, a greater pressure gradient across them. The higher leading edge of the Tyrrell wing gave a higher stagnation streamline (at the same ground clearance) and thus allowed more air into the under-section; to satisfy continuity this must cause a lower pressure to be generated in the channel between the ground and aerofoil. A consequence of the greater downforce level was a larger drag penalty, although it was still more efficient across the tested range. Despite the Tyrrell wing producing more downforce and a larger adverse pressure gradient, it was the NACA 4412 which stalled first due to ground proximity. This appeared to be due to the Tyrrell wing developing more of a plateau rather than an abrupt stall (as seen by the NACA section). This could be attributed to the apex of the Tyrrell being located further upstream, however the pressure distribution shows a much steeper pressure recovery because of the suction levels. It may also be due to the Tyrrell wing having two suction peaks in which the second favourable pressure gradient provides enough momentum to help it overcome the second suction peak and remain attached. These two suction peaks are due to the geometry of the Tyrrell wing, whereby it has two regions of high curvature thanks to the higher leading edge and sharper apex. Overall, this study showed how relatively small geometric changes can drastically alter the performance of an aerofoil. The authors followed this up with a second study in which they concluded that a more cambered wing would produce more downforce [80].

Reynolds-number scaling effects were investigated by Correia et al. [81] through wind-tunnel testing and numerical modelling with a 2D code. It was found that the laminar separation bubble that formed on the suction surface altered the effective shape of the wing to aid downforce production. As the Reynolds number was increased, the bubble size

was altered, downforce increased and drag reduced significantly. The authors postulated that the bubble re-energised the boundary layer during the reattachment process such that it was more capable of overcoming the adverse pressure gradient. This notion was proposed due to observation of the bubble being eliminated by tripping the boundary layer at $x/c = 0.25$, which resulted in a large reduction in downforce due to trailing-edge separation.

Observation of a wing situated behind another car i.e. in a large turbulent wake, by Correia et al. [82], showed that a fully-turbulent boundary layer across the entirety of the wing was produced. This eliminated the laminar separation bubble and caused trailing-edge separation to occur. In comparison to the freestream condition, when operating in the wake downforce and drag were reduced and the tip-vortex strength depleted. These results were attributed to lower dynamic pressure, increased turbulence intensity and local flow incidence changes that the wake caused.

In an effort to understand the influence of compressibility Keogh et al. [83] conducted a computational study in which the Mach number was increased whilst maintaining a constant Reynolds number, and both Mach and Reynolds numbers were increased together. The Mach number was shown to be as influential as Reynolds number, highlighting that both parameters would have an effect during sub-scale testing. The use of fully-turbulent closure models, however, means that key aspects of low Reynolds number flows were not modelled; the full influence of viscous effects, therefore, were omitted from the solution.

Keogh et al. [84] and Roberts et al. [85] almost simultaneously published work on the influence of cornering for a wing. When a racing car navigates a corner two yaw effects occur: constant yaw angle ($^{\circ}$) due to side slip, and the dynamic yaw rate, or rate of change of yaw angle ($^{\circ}/\text{sec}$), as the vehicle rotates about its centre of gravity. A result of this is that

in the rotating reference frame the airflow has both centripetal acceleration and Coriolis terms; thus there is a velocity gradient but with constant static pressure. This makes simulating this condition in a standard automotive wind tunnel almost impossible. Keogh et al. investigated both of these conditions in a fully-computational approach, showing that considering both the yaw angle and yaw rate effects individually, and when combined, gave different results in terms of both surface pressure and vortex trajectory. There is still a question over the reality of such a condition, however, as it can be argued that the yaw rate is only sufficiently high for very small periods of time; when the vehicle undergoes fast change of direction events. It may, therefore, have very little significant aerodynamic influence except in an instantaneous moment of time. Work by Molina & Zhang [86] showed that a wing undergoing a sinusoidal heaving motion could be characterised by a quasi-static approach when the frequency is low. Further work would be required to ascertain whether the same is true for the yawing motion, and whether the effect is large enough such that wind tunnel testing of a car at various yaw angles is not sufficient.

A study into yaw angles of up to 5° on a multi-element wing was conducted by Roberts et al. [85]. This comprised of both experimental and computational tests with a 50% model representative of a racing-car front wing and nose. The transition-sensitive closure model, $k - k_L - \omega$, was used and provided an accurate prediction of experimentally-derived force coefficients and surface-flow structures. It was shown that the surface pressures were altered such that loading became asymmetric, which in turn altered the formation of vortices by either strengthening or inhibiting them depending on their rotational sense. This resulted in a highly asymmetrical wake, which was amplified in the far-wake region as the vortices moved parallel to the wind direction. In addition to flow structure changes, which appeared to be amplified as either the yaw angle was increased or ground clearance reduced, an overall reduction in both downforce and drag coefficients of up to 4%

occurred.

2.3.1 Computational Methods

A number of studies have utilised CFD to study a wing in ground effect, some of which have been discussed previously. In this section, the methodology of these studies is focussed on, with particular attention to how the methods have progressed over the years.

The first computational studies of inverted wings in ground effect were panel-method simulations performed by Katz [87] and Knowles et al. [62]. The mirror-image technique was used to represent the ground. This method involves splitting the aerofoil into a series of panels, each of which has linearly-varying vorticity along its length. The vorticity between each panel is assumed continuous, although the gradient of vorticity can be discontinuous. Flow tangency at each panel is then found by solving for vorticity. This method does not include viscous effects. Some consequences of this limitation were Katz not observing force-reduction at very low ground clearances, and the poor agreement with experimental results observed by Knowles et al. at low ground clearance and/or high incidence angles.

Zerihan & Zhang [88] compared the results of their experimental study [69] to RANS computations using a two-dimensional structured grid. Both the Spalart-Allmaras and $k - \varepsilon$ models were used. They stated that they found Spalart-Allmaras to be more robust, though both models required an unsteady solver to be used to achieve convergence of a steady result. The static pressure distribution was predicted with good accuracy by both models. At higher ground clearances both models predicted very similar distributions; at lower ground clearances, however, Spalart-Allmaras tended to over-predict suction. The deficiency of the computations was found in the wake, where a thicker wake with greater

velocity deficit than that found in experiments was predicted. Although it does not state as much in the conference paper, in his PhD Thesis [89] Zerihan stated that in order to give good correlation a boundary-layer trip at located at $x/c = 0.1$ was required.

The flow around a GA(W)-1 profile was simulated by Lawson et al. [90] using a RANS solver with Spalart-Allmaras turbulence model. The results were compared to PIV and surface pressure measurements and showed good agreement at all ground clearances for zero incidence. At $\alpha = 15^\circ$ the correlation was much poorer, where a thinner boundary layer and later separation were observed in the CFD; this was attributed by the authors to poor modelling of wind tunnel conditions in terms of freestream turbulence level, boundary-layer suction ahead of the rolling road and road-surface roughness.

Multiple RANS turbulence models were trialled by Mahon & Zhang for two-dimensional, single-element [91] and multi-element [92] wings. Comparison of a large range of turbulence models showed $k - \omega$ SST to be the most accurate at modelling surface pressures while *Realizable* $k - \varepsilon$ was most accurate in the wake. At low ride heights only $k - \omega$ SST was able to predict the onset of stall. At very low ride heights a jet flow (large mean strain rate) was present between the ground and the lower boundary of the wing's wake. This is an area that turbulence models have historically struggled with; yet, the *Realizable* $k - \varepsilon$ model was shown to give much improved results in these instances as it calculates eddy viscosity locally, thus allowing the model to remain realizable in regions of large mean strain rates. It was found that a finite trailing edge, and corresponding recirculation region, caused the wake to be thicker; thus it was hypothesized that using a sharp trailing edge could reduce the wake size.

The effect of using either an incompressible or compressible flow solver was evaluated with three different turbulence models by Doig et al. [93]. The same Tyrrell-026 aerofoil

that was used by Zerihaan & Zhang [69] was discretised into a fully-structured grid with $1 \leq y^+ \leq 2$. In the aeronautical sector it is considered that the effects of compressibility appear at Mach numbers above $M = 0.3$. However, the authors found that for a wing in ground effect the force results from each solver began to diverge, due to changes in local density altering surface pressures, by several percent at as low as $M = 0.15$. As ground clearance decreased this critical Mach number at which results diverged from one another also reduced. Consequently, a freestream Mach number of $M = 0.15$ was advised to be the cut-off above which a compressible solver was required.

The first work to use a method other than RANS was Heyder-Bruckner & Zhang [94], who simulated the experimental work of van den Berg & Zhang [95] by conducting a Detached Eddy Simulation (DES) study. In addition to verifying van den Berg's results it was shown that DES and RANS gave similar results at higher ground clearances where flow could be considered almost steady, but at lower ground clearances, where vortex breakdown and separation readily occurred, DES performed much more accurately. DES also modelled endplate vortices more accurately compared to PIV, however this did not always have an effect on the resulting static pressure distribution.

A number of studies have used a RANS solver with one of the classical fully-turbulent closure models to observe the characteristics of wings in ground effect. These include Soso & Wilson [96], Moryossef & Levy [97], Mokhtar [98], Vogt & Barber [79, 80], Kuya et al. [99], Molina & Zhang [86], and Molina et al. [100]. Despite the low Reynolds number conditions of each of these tests, none has included laminar phenomena. The fact that Zerihaan [89] and Doig et al. [93] both used force-transition experimental cases when comparing to their fully-turbulent models shows that laminar effects were clearly manifesting. It is, therefore, important to understand whether the transition-sensitive models that are now available are more suited to the application of wings in ground effect at

realistic Reynolds numbers.

2.4 Summary

The review of literature has focussed on three key aspects that are relevant to this thesis. Firstly, the fundamentals of boundary-layer transition were examined, in which the different mechanisms of transition and parameters that influenced transition were explained.

The second section cited studies from the aeronautical sector in which boundary-layer transition was investigated using experimental methods. These showed that separation and reattachment was the primary mechanism of boundary-layer transition. Computational studies that used the transition-sensitive closure model, $k - k_L - \omega$, showed that accurate predictions of separation-induced transition had been achieved for aerofoils operating out-of-ground effect. The cited works were, however, all two-dimensional studies and thus omitted the anisotropic, three-dimensional nature of turbulence.

The final section examined previous work on wings in ground effect using both experimental and computational approaches. A laminar separation bubble was also noted by Zeriha & Zhang [69] to be the transition mechanism for an inverted wing in ground effect operating at $Re_c = 4.67 \times 10^5$, although no further investigation of the bubble was conducted. It has been comprehensively shown that as the distance between an inverted wing and the ground decreases the wing will generate more downforce as flow is constrained between it and the ground, hence increasing suction levels. This phenomenon holds true until the ground clearance reduces to a critical value, beyond which the flow can no longer overcome the adverse pressure gradient associated with the increased suction levels, and leads to downforce reduction as the wing continues to move closer to the ground.

Despite numerous studies on inverted wings operating in ground effect having been conducted at relatively low Reynolds numbers, the aspects of laminar boundary layers and boundary-layer transition were often overlooked. Even the most recent computational studies [83, 84] have made use of fully turbulent closure models despite operating at a Reynolds number at which transitional phenomena are still prevalent and, in the case of the latter study, observing the effect of Reynolds number scaling. Despite the low Reynolds number conditions of each of these studies, none has included natural transition phenomena as a laminar region was only included up to $x/c = 0.1$ such that comparison could be made to forced-transition experiments. Moreover, computations by Zerihan [88] and Doig et al. [93] both used forced-transition experimental results when comparing with their CFD simulations, showing that laminar effects were clearly manifesting in the experimental results.

Overall, it has been shown that despite laminar and transitional flow phenomena clearly being evident in model-scale wind tunnel testing of wings in ground effect, there is little understanding of their importance or influence in comparison to the aeronautical sector. Moreover, whilst the use of transition-sensitive RANS models has been successfully applied to aerofoils out of ground effect, the majority of ground-effect studies have been conducted only with classical fully-turbulent models.

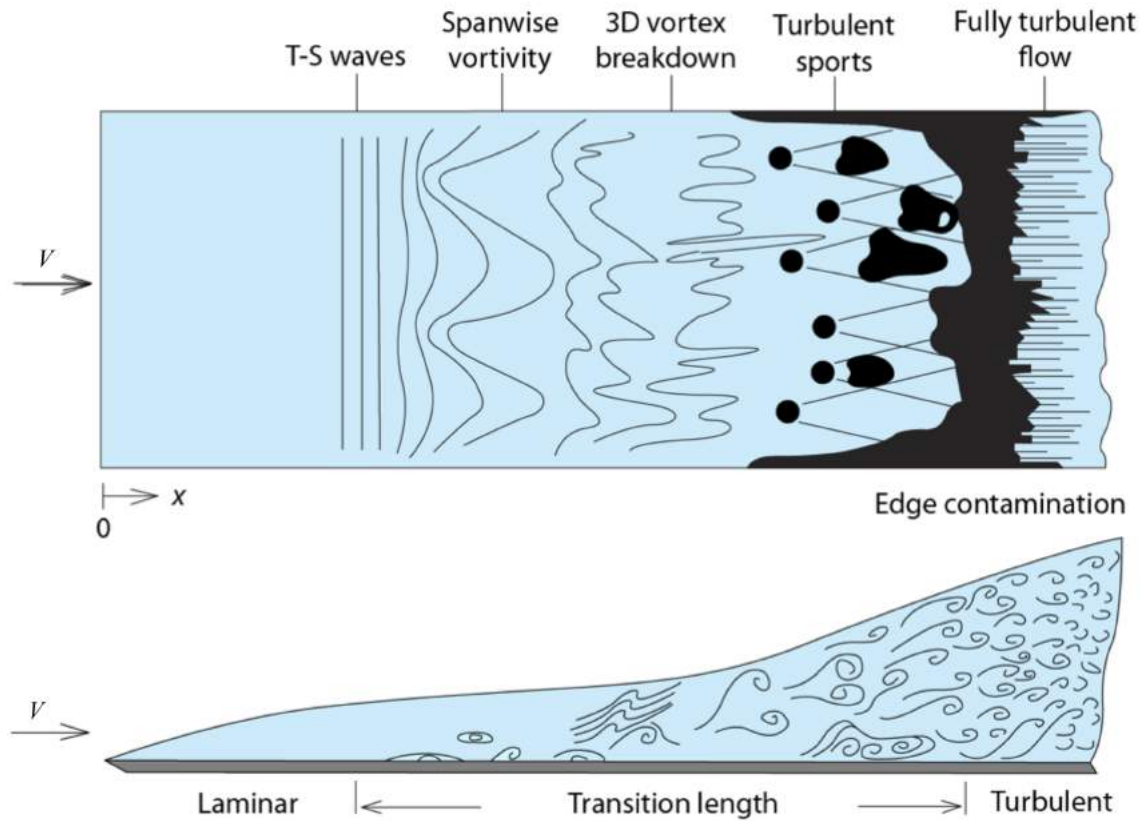


Figure 2.1: Schematic of natural transition over a flat plate [101]

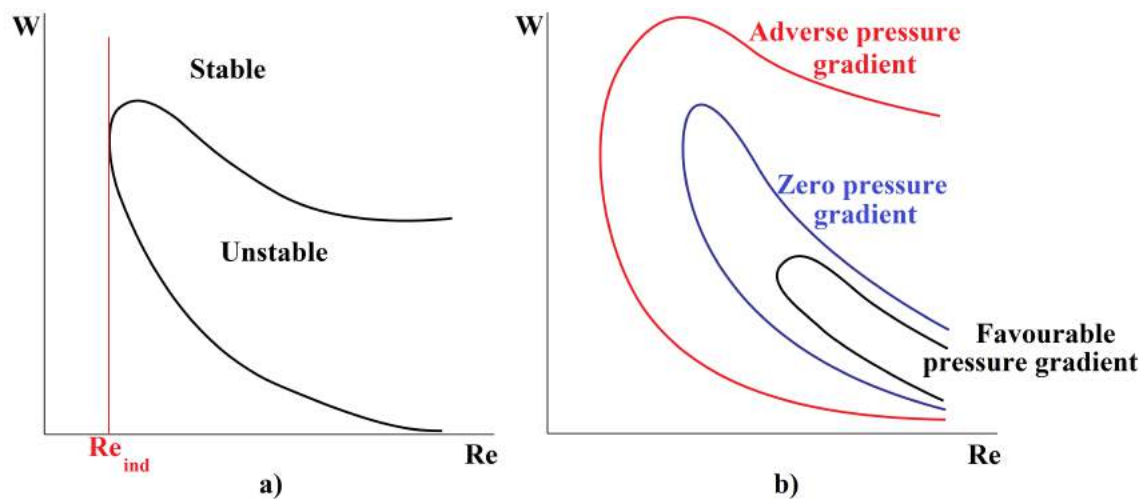


Figure 2.2: Curves of neutral stability for two-dimensional incompressible perturbations in a) plane boundary layer, and b) boundary layer under various pressure gradients

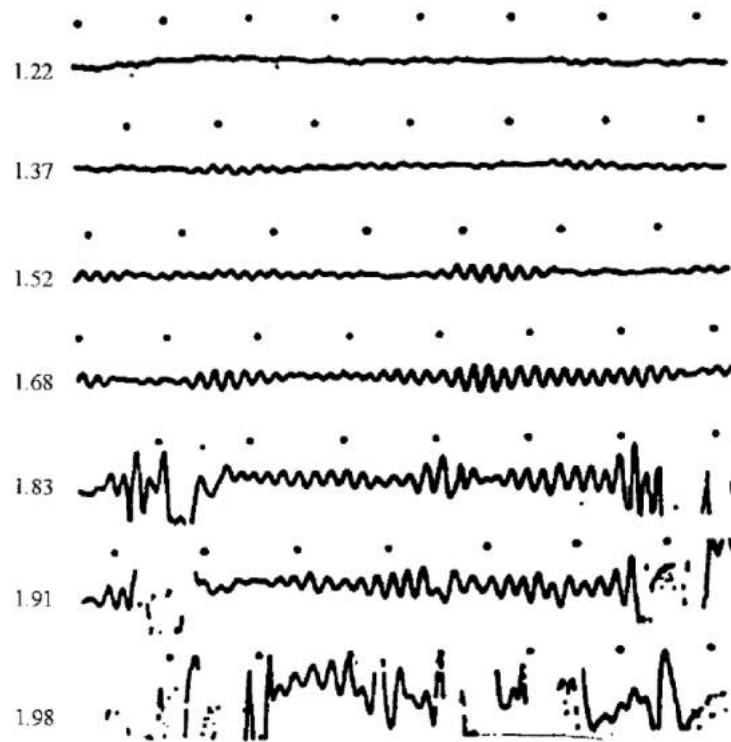


Figure 2.3: Oscillograms of hot-wire anemometer signals showing fluctuations in the boundary layer of a flat plate at set distances from the leading edge (in metres) for a fixed freestream velocity [5]

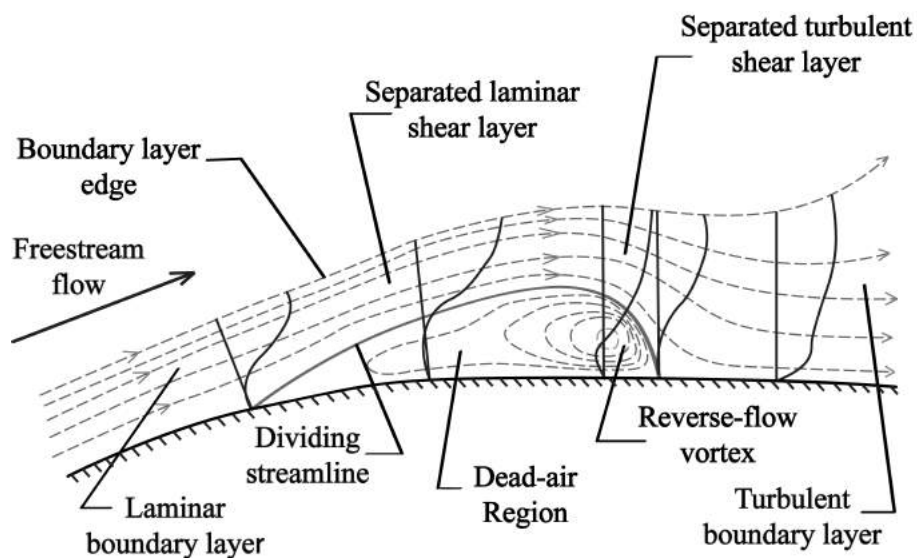


Figure 2.4: Schematic of a laminar separation bubble [15]

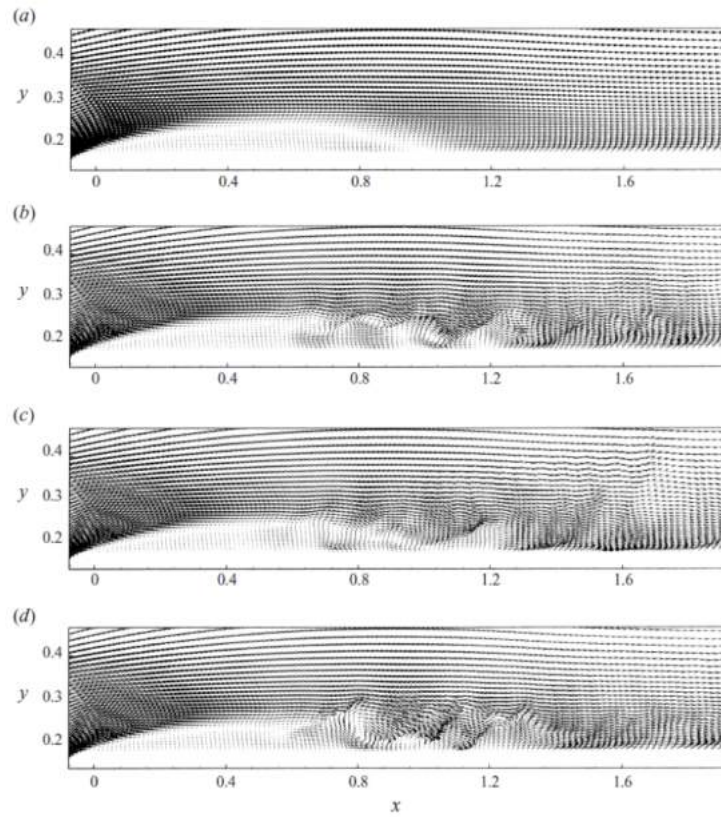


Figure 2.5: Velocity vectors of a laminar separation bubble a) averaged result b-d) instantaneous results [13]

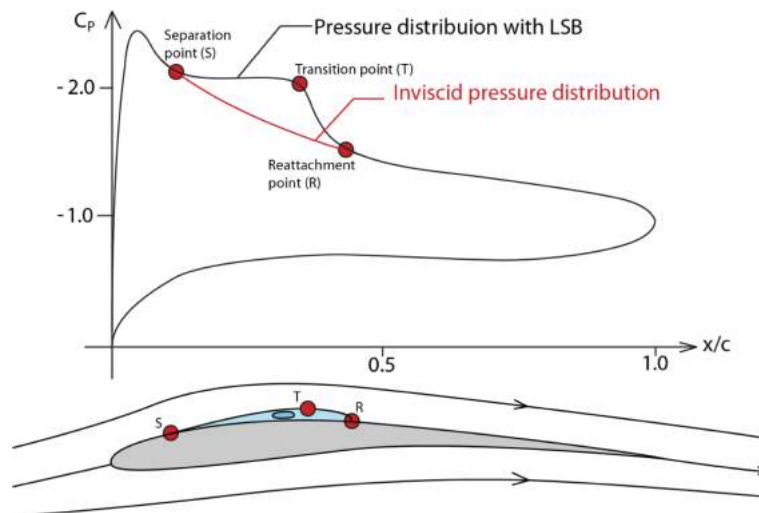


Figure 2.6: Static Pressure distribution about an aerofoil with a laminar separation bubble [49]

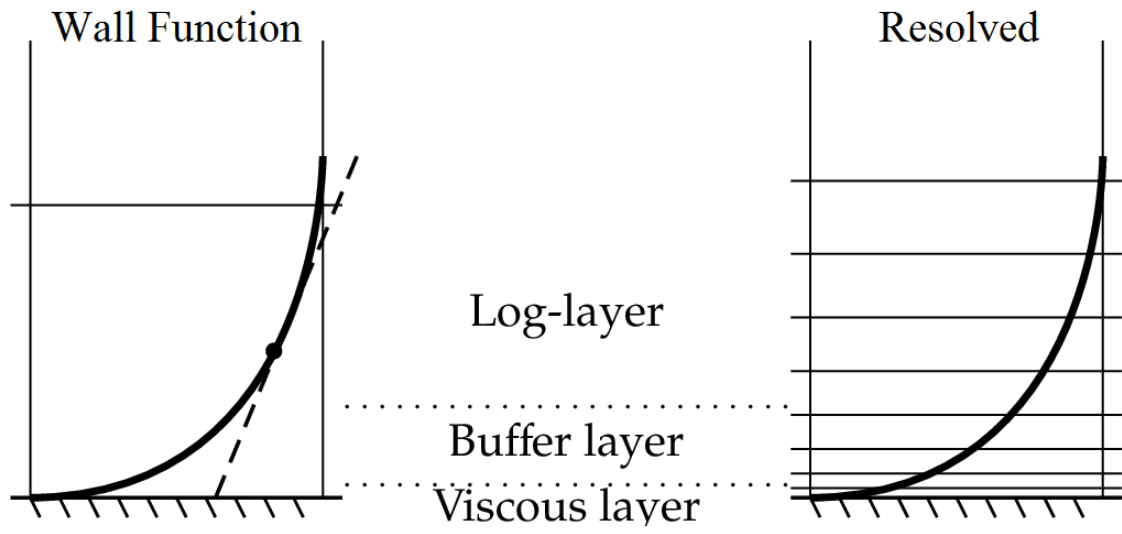
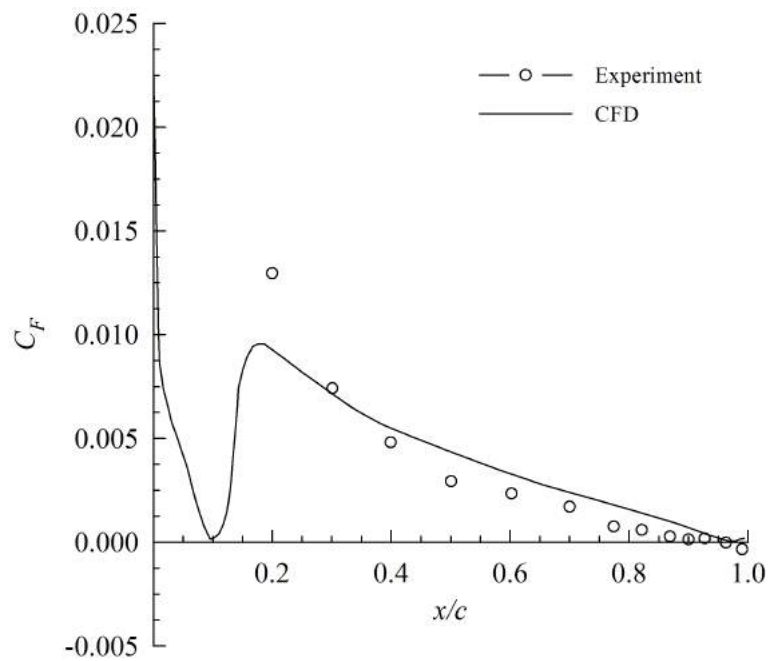


Figure 2.7: Regions of a turbulent boundary layer

Figure 2.8: Comparison of predicted and measured skin friction for $k - k_L - \omega$ in the test case of Walters & Cokljat [46]

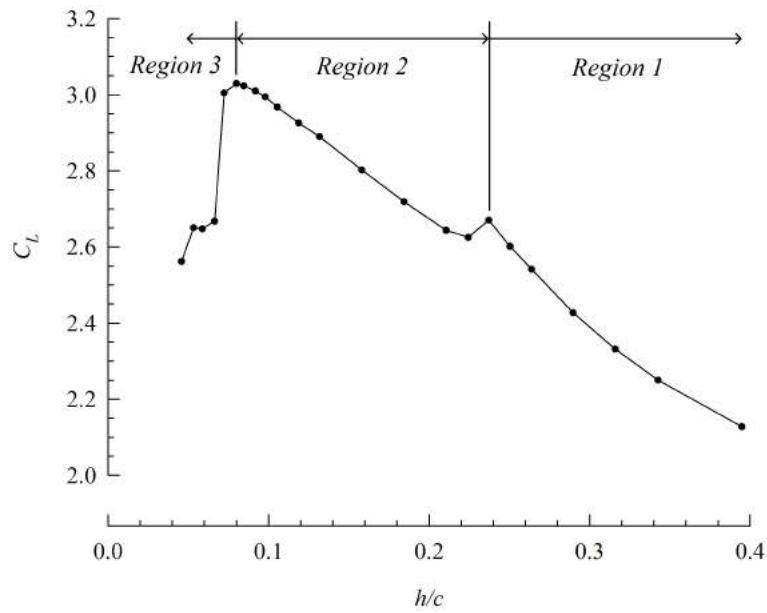


Figure 2.9: Downforce curve for a double-element wing [72]

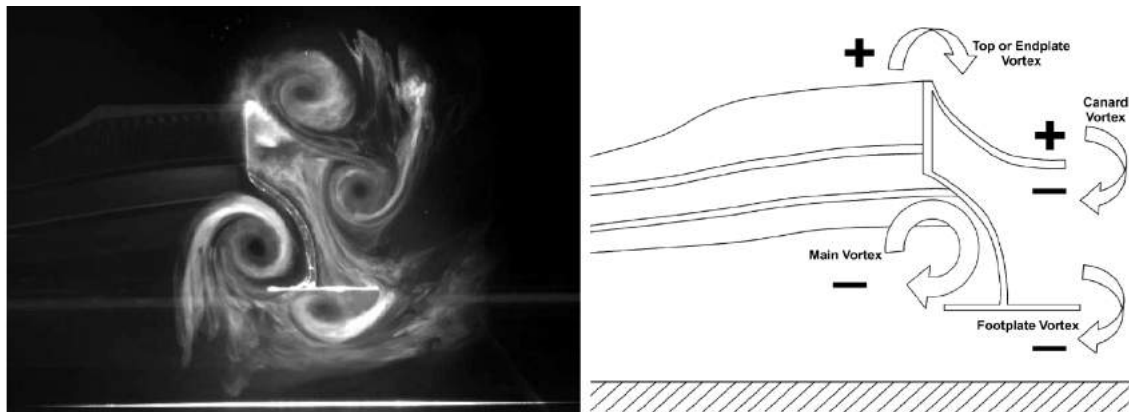


Figure 2.10: Vortex system behind a McLaren F1 front wing a) laser-smoke visualisation (looking upstream from behind wing) and b) schematic of vortex system showing vortex formation due to pressure gradients [77]

Chapter 3

Description of Experiments

3.1 Test Facility

Wind tunnel tests were conducted in the DS Houghton Wind Tunnel (Figure 3.1) at the Defence Academy of the UK in Shrivenham, Oxfordshire. The wind tunnel was formerly used by Reynard Racing Cars Ltd. for the development of single-seater racing cars. It is a 2.7 m x 1.7 m closed-return, three-quarter-open test section equipped with a continuous-belt rolling road, which replicates the road underneath the car. The rolling road is automatically synchronised with the freestream velocity by the wind tunnel control system and suction applied from underneath the belt to prevent it being raised by the aerodynamic forces generated by models, or belt expansion. Boundary-layer suction is applied through perforated plates ahead of the road. The distribution and optimisation of this suction, coupled with a knife-edge transition to the belt, gives a flow velocity that is effectively 100% of the freestream value down to 1.58 mm from the ground at the leading edge of the model. Before air enters the test section it flows through two sets of turning vanes, 4 turbulence-reduction screens, and the nozzle. This results in steady, aligned flow

with a freestream turbulence of 0.3% [102]. Models are supported by an overhead strut that incorporates an automated motion control system to adjust the model's ground clearance to the road during the experiment to within 0.01 mm. Further information on the tunnel is given by Knowles & Finnis [103].

A wind tunnel is a confined volume and, thus, does not fully represent the real-world conditions of infinitely spaced boundaries, but rather presents boundary conditions somewhat similarly to a CFD domain. The frontal area of the wind-tunnel model in relation to the cross-sectional area of the wind-tunnel test section is known as the blockage ratio. It is desirable to have a minimum blockage ratio, to minimise the impact of the wind tunnel's boundaries. The principal effect of blockage is to cause an acceleration of flow between the model and the wind-tunnel boundary. For a closed-test section tunnel this boundary is a solid wall, whilst for an open-test section tunnel this boundary is static (non-moving) air. Blockage is commonly reported to have less influence in an open-test section tunnel because there is space for the air to expand into, and thus the constraining of flow between the model and wind tunnel's boundary not as significant as it would be for the solid walls of a closed-section tunnel. The blockage can be accounted for, however, through use of blockage corrections. For closed-section wind tunnels these correction schemes are well defined, with a number of studies verifying the method proposed by Maskell [104].

For open-test section tunnels the correction schemes are heavily dependent on the wind tunnel geometry, including jet dimensions, collector area and distance to walls; and thus corrections are typically facility specific [105]. A two-measurement method that accounts for blockage and static pressure gradient effects was described by Mercker & Cooper [106] and shown to appropriately adjust drag coefficients towards an on-road condition for automotive applications, thus allowing results from different wind tunnels to be compared on a more equitable basis. Lounsberry & Walter [107] developed the concept further to

allow for easier implementation and showed improved correlation between model-scale and full-scale wind tunnel results. Such an approach is useful for large blockage models, such as in the automotive sector. If the blockage ratio is sufficiently small, however, blockage effects can be considered negligible due to the air being capable of flowing freely around the model. In the present work the maximum blockage is 2.3%, which is sufficiently small that it has negligible influence on the experimental measurements.

The wind tunnel operation and data logging is controlled by a Pi Mistral system. This allows for test "templates" to be generated that automatically vary all operating parameters, including model ground clearance, wind and road speed, and boundary layer and belt suction. All logged data are then exported and plotted in real time using Pi Aero software.

3.2 Test Conditions

Experiments were conducted at a variety of freestream velocities between 12ms^{-1} and 35ms^{-1} . Unless stated otherwise, the tunnel was operated in constant Reynolds number mode. In this mode, the wind tunnel control system regulates the wind speed so that the Reynolds number, based on standard sea-level conditions and the selected wind speed, remains constant. The tunnel uses a system of chillers to maintain the wind and road temperatures during the test; this is especially important for tests of 25ms^{-1} and above, as below this speed the temperature rise is not enough to have a significant effect on the Reynolds-number variation across a run. Prior to tests the tunnel was warmed up so that the temperature would reach 25°C , then this was maintained throughout the tests using the chillers. Unfortunately the chillers developed a fault, which could not be fixed, part way through the test programme and were not available for use in all of the two-dimensional wing experiments. Although the influence of the temperature on the test Reynolds number

is negligible due to the control system regulating the wind speed in reaction to it, the heating of the wing surface can influence the boundary layer, as discussed in the literature review. To minimise this effect, the test was not started until the wind temperature was approximately 18°C , which resulted in a maximum temperature of approximately 26°C even for the highest wind speed tests. For tests where constant Reynolds number mode could not be employed, the variation of Reynolds number during the test is quantified and stated.

3.3 Test Models

In this work, four different wing configurations were used. The first was a two-dimensional (2D) wing, which in practice is a high aspect ratio wing that allows tests to be conducted on what is essentially just an aerofoil section.

The second wing was a simplified Formula-One type wing, the geometry was designed by Correia [108] and thus is referred to as 'CSW' (Correia Simplified Wing), and can be set up as either a single-element (CSW-SE) or a multi-element (CSW-ME) wing. The use of single- and multi-element configurations allows the incremental incorporation of three-dimensional effects, building on the results of the two-dimensional experiments.

The final wing was a 50% scale 2009 Formula-One (F1) wing provided by a current constructor; this is a highly complex geometry that includes multiple features designed to create secondary flows. This wing was used to highlight the differences between research based on two-dimensional results or a simplified geometry and a highly complex F1 wing.

This section explains the details of wind tunnel testing, including the facility, models, and experimental techniques that have been used. It should be noted that details are explained

in the order in which results are presented, rather than chronological order. The results are ordered such that the simplest case, the two-dimensional wing, is presented first and the most complex case, the three-dimensional wing configurations, presented last. The chronological testing order, however, was the three-dimensional wing geometries first, followed by the two-dimensional wing last. This is simply because the three-dimensional models were available first, as they were manufactured for a previous study by Correia [108].

3.3.1 Two-Dimensional Wing

A computer-generated image of the two-dimensional (2D) wing experimental setup inside the DS Houghton wind tunnel is given in Figure 3.2. An untapered, untwisted, rectangular-planform GA(W)-1 aerofoil section wing of aspect ratio 5.14 and chord 350 mm was used. The wing consisted of three hollow sections of span 570 mm that could be arranged in any spanwise order. Each section was comprised of two pieces, an upper and a lower, as shown in Figure 3.3. The upper section was essentially inserted into the lower section so that the join between the two pieces was confined to the upper (pressure) surface of the wing, as this study focusses on the lower (suction) surface. This design also improved the rigidity of the section. The pieces were secured together by 4 counter-sunk bolts from the upper surface. Each of the three sections was designed for a specific measurement technique: one section was fitted with static pressure tapings; one section had hot-film gauges; and one section was left smooth for surface-flow visualisation and LDA measurements. The hollow nature of the pieces meant that any necessary instrumentation could be placed inside the wing (Figure 3.4). Additionally, cables could be passed between the sections through an access hole (A in Figure 3.3). The sections were fitted together by two brass pins, a larger one (B in Figure 3.3) located near to the leading edge

and a smaller one (C in Figure 3.3) close to the trailing edge. Additionally, two long bolts (D in Figure 3.3) were passed between two adjacent sections, again to ensure a tight fit and provide further rigidity.

As shown in Figure 3.2, the wing was mounted in the test section by two endplates situated either side of the rolling road. The endplates, whose geometry can be seen in Figure 3.5, were fixed to a space frame comprised of three triangular supports. This space frame was fixed in place by bolting it to runners in the T-piece that runs parallel to the road. The endplate allowed the wing to be orientated at multiple ground clearances and angles of incidence. The larger holes (A in Figure 3.5) in the endplate is for the brass pins (B in Figure 3.3) that extrudes from the wing section and designates the point of rotation for incidence changes. The incidence is then altered by bolting into one of the five available holes (B in Figure 3.5), each hole is equivalent to a 1° change in incidence. This method means that the ground clearance, measured from the lowest point on the wing to the road, alters as the incidence alters. For consistency the ground clearance will always be referred to by the zero incidence case, or if you will, to one decimal place (as the change is very small). The actual measured change in ground clearance is provided in Table 3.1. The long cut-out (C in Figure 3.5) between the sets of holes is there for cables to be connected to instrumentation inside the wing.

The influence of blockage ratio was discussed previously, and it was stated that if the blockage ratio was sufficiently small in an open-test section wind tunnel then the effect of blockage can be considered negligible. The blockage ratio for the 2D wing was 2.3%, and thus blockage was not an influencing factor in the experiments.

Table 3.1: Ground clearance increase due to change of incidence

Incidence (α)	Difference (mm)	Normalised Difference (h/c)
0	-	-
3	4.055	0.012
5	5.896	0.017
8	7.558	0.022
10	8.094	0.023
13	8.231	0.024
15	8.993	0.026

Positioning and Setup

The wing was positioned on the rear set of ground clearance holes (labelled as Set 2 in Figure 3.5) for all experiments in this work. This is because the pressure gradient across the test section meant that it could not be guaranteed that the wing would be operating in comparable conditions if it were mounted 437.5 mm up- or downstream in the test section. The leading edge of the wing was positioned 750 mm from the start of the road; this corresponds to the region of the test section with the smallest streamwise variation in static pressure.

As the 2D wing was supported directly by the staging either side of the rolling road it was important to discover whether any vibrations, whether from the fan or the road, were being transmitted to the model; or even if the wind itself were causing the wing to vibrate. A 6-axis accelerometer rated up to 300 Hz was fixed inside the central wing section on the lower section at approximately $x/c = 0.5$. At a wind speed of 15ms^{-1} the fan has a frequency of 75 Hz, neither this frequency or any harmonics were observed by the accelerometer. A low-power frequency of approximately 500 Hz was observed, however given that the accelerometer is only rated to 300 Hz, this is outside the accelerometer range. Furthermore, 500 Hz is too high to have an influence on the low-speed flow. Hence, it was deemed that the wing was not being influence by the fan. Tests conducted

with the fan and road disabled showed that the accelerometer was sensitive enough to pick up vibrations caused by simply walking on the road, thus showing that were the fan or road vibrations influencing the wing then the accelerometer would have been capable of detecting them.

The orientation of the wing relative to the test-section centre line and ground plane was determined from a number of measurements. A number of spacers were used during the setup to alter the orientation of the model to improve its position; the measurements given here are those at which the model was orientated during all tests. Measurements of the distance from the trailing edge to the rolling road were taken at various spanwise locations using a height gauge to ensure that the trailing-edge height remained sufficiently constant across the span. The height varied by 0.24 ± 0.04 mm between the two ends of the wing; this is equal to a roll angle of $0.008^\circ \pm 0.0013^\circ$. In the measurement region, however, which is the centre 100 mm of the wing; the height from the ground to trailing edge varies by just 0.014 mm. The height of the leading- and trailing-edges of the endplates were also measured to ensure that they were not causing the wing to be set at roll. This showed 0.14 ± 0.04 mm between the leading edges of each endplate, and 0.04 ± 0.04 mm between their trailing edges. The yaw angle was determined by measuring from the start of the rolling road to the leading edge of the model. The model was positioned so that both sides were exactly the same distance away from the start of the road to within the accuracy of the ruler used; so the stated yaw angle is $0^\circ \pm 0.033^\circ$. The accuracy of the incidence is based on two parameters, the manufacturing tolerance of the four holes that designate the wing's location, and the measured incidence of the endplates using the height gauge. The wing's incidence was found to be $0.054^\circ \pm 0.022^\circ$ (nose down). Lastly, the ground clearance was determined by finding the distance from the top of the larger brass pin to the road. Subtracting the distance from there to the lowest point of the wing

(at zero incidence), based on the CAD design, gives the ground clearance. Including manufacturing tolerances, this gave a ground clearance accuracy of -0.059 ± 0.05 mm. Overall, these measurements demonstrate that the model has been accurately positioned in the tunnel.

In practice, no experimental flow will be truly two-dimensional, as this would require no tolerance in the geometry and its orientation, and an infinite aspect ratio. Although the model has been set up to a high accuracy, as described above, this does not guarantee two-dimensional flow. Therefore, tests were conducted to investigate this. The first was a simple wool tuft test. Tufts were placed all over the wing and endplate to observe flow direction and any areas of high separation. A camera was mounted above the test section and the wool tufts on the upper surface of the wing photographed. The original photograph has been transformed to an 8-bit grayscale image, and a histogram equalisation function used, to provide an image (Figure 3.6) where the wool tufts can be clearly seen. This shows there is some unsteadiness on the wing close to the endplates; this caused the tufts to vibrate in the flow. In the central region, however, where all the measurements were taken, the tufts all appear straight and steady. In the second test, a traverse-mounted Pitot-static tube was used to measure the pressure immediately upstream of the wing (Figure 3.7). The long length of the probe meant that it vibrated easily in the flow, and as a result had to be rested on the upper surface of the wing. Additionally, as the traverse was bolted into the staging next to the road and then stood on legs across the road, the rolling road was disabled for these tests. The results of this test are given in Figure 3.8. This shows that the dynamic, total and static pressures vary by less than the accuracy of the pressure transducer (4.98 Pa) across the central section of the wing. The fact that the dynamic pressure decreases in the positive spanwise direction and static decreases in the opposite direction, thus giving a constant total pressure, leads to the conclusion that the

wing is in a slight roll, which as shown by the orientation measurements is true. Overall, the results of both the wool tufts and pressure measurements show that the flow is of an acceptably two-dimensional nature in the measurement region of the wing.

Test Cases

The two-dimensional wing model can be orientated at various ground clearances and incidence angles. It is possible for the wing to be set at 176 individual orientations, but in practice, far fewer than these were used. As explained in the previous section, only the rear set of ground clearance holes on the endplate were used, giving a ground clearance range of $0.1 \leq h/c \leq 0.9$ in intervals of $h/c = 0.2$. The main tested ground clearance, and that which is considered the datum ground clearance, was $h/c = 0.3$ as this is the most representative ground clearance of a racing car front wing. At each ground clearance the wing could be set at incidences in the range $0^\circ \leq \alpha \leq 15^\circ$ in steps of 1° ; only the incidences given in Table 3.1, however, were tested. The test Reynolds number, based on the 350 mm chord of the wing and the ambient temperature and pressure recorded by the control system during testing, are given in Table 3.2.

Many of the decisions taken when designing the model were taken to improve structural rigidity due to the high loads that the wing would be placed under, specifically when at incidence and close to the ground. The wing's deflection was measured using an theodolite

Table 3.2: Test Reynolds numbers (chord-based) for 2D wing experiments

Rated Test Speed ms^{-1}	Re
12	2.88×10^5
15	3.59×10^5
20	4.79×10^5
25	5.99×10^5
30	7.19×10^5

by observing where the trailing edge was during a wind-on run, then adding weights to the wing when the wind was turned off until the trailing edge deflected to that same point again; the deflection was then measured using a height gauge. At $\alpha = 0$, $h/c = 0.3$ and a wind speed of 30ms^{-1} the wing's deflection was negligible, but at $\alpha = 3$, $h/c = 0.3$, and a wind speed of 25ms^{-1} the trailing edge deflected by 0.16 mm. For this case, the static pressure showed suction of 1000 Pa, and thus this was deemed the maximum that would be exerted upon the wing. Based on this, some of the ground clearances and incidences were limited in the speed at which they could be tested, specifically those of higher incidence and closer to the ground. The main limitations were that the highest Reynolds number condition of 8.32×10^5 was only tested at zero incidence at ground clearances of $h/c > 0.3$, and a maximum Reynolds number of 4.75×10^5 could be used for $h/c = 0.1$.

As indicated previously, four measurement techniques (which are explained in detail in later sections) were utilised on the 2D wing: surface pressure measurements, evaluation of shear stress with hot-film sensors, surface-flow visualisation, and LDA velocity profiles. Each of these techniques was used for the wing at varying Reynolds number, ground clearance, and incidence

During force-transition tests the boundary layer on the suction surface of the wing was tripped with a roughness-type trip that was manufactured in-house from double-sided tape and Grit 60 sand ($265\mu\text{m}$ diameter). This grit size satisfied the roughness-based Reynolds number criterion that von Doenhoff & Horton [109] showed to be required for transition to occur. The total thickness of the trip (tape + grit) was 0.415 mm. Further detail on the mechanism by which roughness-type strips induce transition can be found in Klebanoff & Tidstrom [36]. To place the trip in the correct location, a template that could be placed on the wing was produced. The distance around the wing leading edge, from the join between the upper and lower surfaces to the desired chordwise location was found using the CAD

geometry from which the wing had been manufactured. Assuming that the perimeter of the wing in CAD and reality are equal i.e. omitting manufacturing tolerance, the trips were placed within ± 0.5 mm of the stated location.

3.3.2 Three-Dimensional Wing Configurations

Simplified Wings

The front wings used on racing cars range from the simple to the highly complex. The simplified wing configurations (CSW) utilised in this work represent not only the interim stages between the two-dimensional wing and highly complex F1 wing, but are analogous to lower Formulae racing car wings. The wing is used in two configurations, a single element (Figure 3.9), referred to as CSW-SE, and a multi-element (Figure 3.10) configuration referred to as CSW-ME.

The main-plane element is an untwisted, untapered, rectangular planform, constant GA(W)-1 aerofoil section of span 750 mm and chord 119.7 mm. This gives an aspect ratio of 6.27. It is angled at -0.5° incidence (nose up); this was a design decision taken by Correia [108] when designing the wing. Due to manufacturing reasons the main-plane element has to be built with a finite trailing edge thickness of 1.9 mm. This represents 1.6% of the main-plane element chord length.

Two rectangular endplates of length 274 mm, height 99.8 mm and thickness 10 mm were screwed onto the wing tips. As shown in Figure 3.9, the lowest point of the endplate does not meet the lowest point of the wing; the endplates lowest edge is 6 mm below the bottom of the wing. It should, therefore, be noted that all references to ground clearance are taken from the lowest point of the wing, not the endplate.

The multi-element configuration has the addition of two identical flaps which are untwisted, untapered and rectangular-planform constant Reynard Motorsport "Kylie" flap aerofoils. Both have a span of 250 mm, chord of 160.2 mm and are secured in place to the endplate by two screws and to the main plane by a bracket. The flaps can be positioned at three different flap angles, although for this research they remained fixed at 23.9° throughout. Due to the geometry of the flaps they were made from two separate parts. The front section, which features all of the curvature and thus can be considered the aerofoil section, was made from aluminium in similar fashion to the main-plane element. The rear section was made from a steel plate which was then bonded and screwed to the front piece. Similarly to the main-plane element, the flaps were required to have a finite trailing-edge thickness, which is 2.1 mm and is equal to 0.8% of the flap chord length.

The CSW-SE wing is comparable to that used by an entry-level monoposto racing car. The Tatuus F4-T014 (Figure 3.11), for example, features a constant cross-section and rectangular planform single-element wing fitted with simple rectangular endplates. The aerofoil section employed in this wing is very similar to the GA(W)-1. Other than the nose section and size of the endplates, the CSW-SE is very similar to that of the Tatuus. In the same way that as a racing driver progresses up the motorsport ladder, encountering cars of progressively higher downforce, the CSW-ME is similar to that of cars used in F1 feeder series. A Dallara T12 car, which is shown in Figure 3.12, is not too dissimilar to the CSW-ME wing configuration in that both use a constant cross-section main-plane with two semi-span flap elements. The Dallara wing, however, features an additional endplate on the inner end of the flap, and a small footplate on the outside of the endplate. Overall, the CSW wing configurations provide not only an interim stage in the progression from simple to complex, but are also directly relevant to the motorsport industry.

F1 Wing

This wing (Figure 3.13) was originally designed for the 2009 Formula One World Championship Regulations, it was kindly provided the constructor for use in academic projects at Cranfield University. It is an example of the highly complex geometries used in high-level motorsport, featuring a main plane that uses multiple aerofoil sections, multiple flap cascades that are both twisted and tapered, and an endplate with three-dimensional curvature.

The main-plane element involves a significant amount of twist as it moves through at least three different aerofoil sections. The first is a symmetrical aerofoil shape, which is mandated by the regulations for the central 500 mm of the wing, or in the case of the 50% scale model, up to 125 mm from the centreline. This aerofoil is orientated at 3.8° nose-up incidence and is shown in Figure 3.14a. The wing then twists quickly and significantly towards the profile shown in Figure 3.14b, which is a typical downforce-producing aerofoil featuring camber. This section actually appears very similar to the Tyrrell-26 wing that was studied by Zerihan & Zhang [69]. The aerofoil used at the wing tip (Figure 3.14c) is a typical high-lift aerofoil in that it is a thin and highly-cambered. Despite the twisting between these aerofoil shapes the wing maintains a constant chord of 119.7 mm across its 750 mm span.

There are two multi-element cascades implemented into the design, the first consists of flap elements for the main plane and will be referred to as the upper and lower flaps. The second is a lone cascade that makes use of two elements, and will be referred to as the cascade wing. This is shown visually in Figure 3.13. The lower flap element is fixed in place whilst the upper flaps are adjustable; however, for this research they were maintained at 30.7° . The geometry of each flap varies significantly with each using different aerofoil sections, levels of twist, chord lengths, and span. Both the flaps were

secured to the inner face of the endplate and by a bracket to the element below it.

The endplate design is significantly more complex than the CSW in that rather than just being a plate it includes both a footplate and backplate. However, again these are not simple plates as both involve three-dimensional curvature.

The wing was delivered as a fully-assembled part directly from the F1 constructor, no manufacturing took place in house. The main plane and lower flap elements were made from machined aluminium whilst the upper flap elements were made of carbon fibre composite. The endplates were made of aluminium for the most part, only the front portion of it was not as this was made from a continuous piece of rapid-prototyping resin, which also included the cascade wing.

As the endplates were not detachable the 4 pin method used for the orientation of the CSW could not be used here. Instead measurements were taken at the front and rear of the endplates upper edge, as this is designed to be positioned parallel to the freestream flow direction. Unlike the CSW, the bottom edge of the endplate was coincident with the lowest point on the main plane, hence ground clearance reference was to this.

Installation in the Wind Tunnel

The main-plane of all the wing configurations tested was suspended by two vertical pylons from a force balance located at the bottom of the wind tunnel's overhead strut. The force balance was then enclosed by a streamlined body; as well as isolating the balance from the freestream air this body simulates the nose section of a monoposto racing car, hence this body will be referred to as "the nose". The overhead strut location can be seen in Figures 3.9 & 3.10 as the aerofoil shape on top of "the nose". Although the nose will inherently have an effect on the wing aerodynamically, as it is not in contact with the

Table 3.3: Maximum blockage ratio for test models (including strut)

CSW-SE wing	4.93%
CSW-ME wing	5.50%
F1 wing	5.84%

force balance the loads that it generates are not directly measured. Between the nose and wing the model replicates the entire front assembly of a racing car. Characteristics of the CSW-ME at varying ground clearances, yaw and roll angles through force measurement and LDA wake surveys can be found in Correia [108].

Figure 3.10 shows that the bracket that holds the flap elements in place is embedded into the main-plane element. Obviously this would compromise the characteristics of the CSW-SE and thus modelling clay covered with aluminium tape was used to cover the holes.

The wing was positioned through the use of 4 pins, two placed into either end of the wing one at the front and one at the back. Between these pins the roll and pitch of the model could be analysed. Shims were placed between the vertical strut and the mounting plate on the force balance to change the orientation of the wing as the height of the pins from the ground was measured.

As previously discussed, if the blockage ratio in an open-test section wind tunnel is sufficiently small, then the effect of blockage can be considered negligible. Table 3.3 shows the blockage ratio for all 3D wings is small enough that blockage effects can be considered negligible.

Test Cases

The model was only designed for force measurement and surface-flow visualisation. Wake surveys were not taken as no LDA equipment was available at the time. The ground

clearance was varied in the range $0.125 \leq h/c \leq 1.667$ for all wing configurations. As explained earlier, the ground clearance is defined as the distance between the lowest point of the wing and the rolling road. The ground clearance could be altered during tests by the automatic strut via a "template" assigned in Pi Mistral. The test began with the wing at 200 mm ground clearance ($h/c = 1.67$) and was then moved towards the ground in progressively smaller increments to a minimum height of 15 mm ($h/c = 0.125$). As is the norm for ground-effect studies, the ground clearance (h) is normalised by the wing chord length (c) giving the non-dimensional ground clearance (h/c). To allow easy comparison between different wing configurations, the main-plane element chord length is used throughout, irrespective of whether a flap is fitted or not.

To give some perspective on typical ground clearances for racing car front wings, an F1 car's front wing will be 75 mm above the ground when the car is aligned parallel to the track. With aerodynamic downforce and pitch changes (static and dynamic) this decreases significantly. In fact, teams have to use chassis methods such as anti-dive suspension geometry, heave springs and bump rubbers to stop the wing touching the ground. Similarly, F3 cars have a static ground clearance of approximately 70 mm. As a 50% model is being tested, the ground clearance must also be scaled. As a result, a ground clearance of 75 mm for a full-scale car is equal to 32.5 mm for a 50% scale model. The typical non-dimensional ground clearance operating range for a racing front wing is $0.2 \leq h/c \leq 0.3125$.

Although the three-dimensional wings are quite different in terms of geometry, all have the same main-plane chord length of 119.7 mm. Therefore, to make comparison between the wings operating at the same freestream velocity obvious, the Reynolds number is based on the main-plane element chord rather than the total chord including the flap elements. Tests were conducted with the wind tunnel with a constant Reynolds number

Table 3.4: Test Reynolds numbers (chord-based) for 3D wing experiments

Rated Test Speed (ms^{-1})	Re
20	1.63×10^5
22.5	1.83×10^5
25	2.04×10^5
27.5	2.24×10^5
30	2.44×10^5
32.5*	2.65×10^5
35*	2.85×10^5

correction applied, this means that the control system automatically varies the wind and road velocity to keep the Reynolds number constant. The actual number is calculated based on standard atmospheric conditions, this gives the chord-based Reynolds numbers given in Table 3.4. The values marked with an asterisk (*) are applicable to the CSW-SE only. The loads generated by the other two wings became too high at these speeds for the balance and fixings. The uncertainty in the Reynolds number, based on the fluctuation in its value due to changes in ambient viscosity and density measured by the control system during tests, was calculated to be ± 418 .

Forced-transition tests were conducted using two trip configurations: a roughness-type trip and a smooth serrated trip. As explained in the previous chapter, any object that artificially introduces perturbations into the boundary layer with a view to generating mixing and increasing momentum thickness can be considered a trip.

The serrated trip was manufactured and supplied by an F1 constructor. The dimensions of the serrated trip can be seen in Figure 3.15; it features a serrated leading edge and is of thickness 0.25 mm. The roughness-type trip was discussed previously. Both trips were 12 mm in streamwise length, equivalent to 10% of the main-plane chord length, and were placed on the aerofoil with an accuracy of $\pm 0.0042c$. The trips were placed between $x/c = 0.1$ and $x/c = 0.5$ at $x/c = 0.1$ intervals, giving five locations for each trip.

The trips were placed on the wing's suction surface by removing the endplates and then measuring to the chosen chordwise location using a ruler. The line was then drawn using a marker pen between two marks at either end to show where the leading edge of the trip had to be placed. The trip was then carefully placed on the wing. The error in trip placement, based on the accuracy of the ruler, is $\pm 0.5mm$.

Tables 3.5 and 3.6 show all the test cases that were conducted.

Double Element CSW

Rough Trip

Forces	20	22.5	25	27.5	30
v					
Clean	3002	3005	3001	3006	3004
0.1	3013	3014	3015	3016	3012
0.2	3024	3025	3026	3023	3027
0.3	3029	3030	3031	3032	3028
0.4	3039	3040	3038	3041	3042
0.5	3051	3052	3053	3049	3050

Flow Viz

v	mm	15	20	25	27	30
Clean	x	x	x	x	x	x
0.1	x	x	x	x	x	x
0.3	x	x	x	x	x	x
0.5	x	x	x	x	x	x

F1 Trip

v	20	22.5	25	27.5	30
Clean	3002	3005	3001	3006	3004
0.1	3009	3010	3008	3007	3011
0.2	3018	3019	3021	3022	3017
0.3	3035	3036	3037	3033	3034
0.4	3044	3045	3047	3048	3043
0.5	3057	3058	3054	3055	3056

Flow Viz

v	h/c	15	20	25	27	30
Clean	x	x	x	x	x	x
0.1	x	x	x	x	x	x
0.3	x	x	x	x	x	x
0.5	x	x	x	x	x	x

Single Element CSW

Rough Trip

Forces	20	22.5	25	27.5	30	32.5	35
v							
Clean	2007	2009	2003	2012	2011	2013	2004
0.1	2087	2089	2090	2091	2085	2088	2086
0.2	2081	2083	2078	2077	2084	2082	2079
0.3	2074	2071	2075	2076	2070	2073	2072
0.4	2065	2068	2063	2062	2069	2067	2064
0.5	2060	2058	2056	2061	2055	2059	2057

Flow Viz

v	mm	15	20	25	27	30
Clean	x	x	x	x	x	x
0.1	x	x	x	x	x	x
0.3	x	x	x	x	x	x
0.5	x	x	x	x	x	x

Serrated Trip

Forces	20	22.5	25	27.5	30	32.5	35
v							
Clean	2007	2009	2003	2012	2011	2013	2004
0.1	2016	2018	2020	2019	2021	2014	2017
0.2	2026	2027	2023	2028	2022	2029	2025
0.3	2033	2031	2034	2035	2036	2030	2032
0.4	2039	2045	2040	2038	2037	2046	2044
0.5	2048	2051	2049	2053	2054	2047	2050

Flow Viz

v	h/c	15	20	25	27	30
Clean	x	x	x	x	x	x
0.1	x	x	x	x	x	x
0.3	x	x	x	x	x	x
0.5	x	x	x	x	x	x

Table 3.5: CSW wind-tunnel test cases

F1 Wing		Rough Trip										F1 Trip									
Forces		v										v									
		20	22.5	25	27.5	30											20	22.5	25	27.5	30
Clean		4001	4002	4003	4004	4005											4001	4002	4003	4004	4005
0.1		4014	4015	4016	4017	4013											4009	4010	4008	4011	4012
0.2		4025	4026	4027	4028	4029											4020	4021	4023	4019	4024
0.3		4030	4031	4032	4033	4034											4034	4036	4037	4038	4039
0.4		4040	4041	4042	4043	4044											4045	4046	4047	4048	4051
0.5		4053	4054	4055	4056	4052											4057	4058	4059	4060	4061

Flow Viz		v										v									
		h/c	15	20	27.5	30											h/c	15	20	27.5	30
Clean		x	x	x	x	x											x	x	x	x	x
0.1		x	x	x	x	x											x	x	x	x	x
0.3		x	x	x	x	x											x	x	x	x	x
0.5		x	x	x	x	x											x	x	x	x	x

Table 3.6: F1 wing wind-tunnel test cases

3.4 Experimental Methods & Uncertainties

3.4.1 Force Acquisition

The measurement of aerodynamic forces was conducted for the three-dimensional wing models using a six-component Aerotech force balance. As explained earlier, the three-dimensional wings were suspended from the force balance, which in turn was mounted on the bottom of the overhead strut Figure 3.17. Force data were acquired at a frequency of 10 Hz for 20 seconds. Nil-force measurements, with wind off, were taken before and after each run to account for any offsets in the measurements during the experiment. This was done to ensure any change in ambient conditions, such as temperature, during the run had not significantly altered the force-balance reading. Two methods were considered: the first to use the average of the pre-test and post-test nil-force measurements, and the second to use an offset that moved linearly from the pre-test measurement to the post-test measurement. The latter proved to be the preferred method.

Repeatability tests were conducted on each wing configuration. These consisted of ten runs of ground clearance sweeps at the same freestream velocity taken one after another. For all ground clearances the largest difference of any measurement to the mean, in terms of absolute (and percentage) lift coefficient and drag coefficient, were 0.026 (0.69%) and 0.011 (1.33%) respectively. Furthermore, it was found that extending the settling time between the wind reaching test velocity and the first data point being taken, as well as taking measurements from the highest ground clearance first, improved repeatability.

The overall uncertainty in lift and drag coefficients at a 95% confidence level is ± 0.014 and ± 0.0023 respectively for all configurations used in experiments. A parametric study of each variable, including roll angle, pitch angle, yaw angle, dynamic pressure and ground clearance, at multiple settings, with the variation between two settings assumed

to be linear, was conducted. The total uncertainty was then calculated using the accuracy for each variable and results of this parametric study combined with the error of the force balance. The value quoted above is the maximum uncertainty calculated for all configurations.

The downforce and drag coefficients for all wings were calculated using the same area (0.09m), the main-plane planform area, as this was constant between all the wings.

3.4.2 Hot-Film Anemometry

In order to evaluate shear stresses on the suction surface of the 2D wing, Dantec Glue-On hot-film sensors were set into the wing as shown in Figure 3.18. The sensors were inset into the suction surface of the wing such that they were flush with the surface at equal streamwise spacing from $x/c = 0.45$ to $x/c = 0.85$ at intervals of $x/c = 0.05$.

The hot-film sensors give a measurement of the wall shear stress through use of an electrically-heated element that relies upon the similarity between the wall-adjacent velocity profile and the temperature profile of the thermal boundary layer that is generated by the element. The principle behind hot-film sensors is the same as hot-wire anemometry, except that the wire is set into a film so that it can be placed onto a surface. The wire is connected to one arm of a wheatstone bridge circuit and heated by an electrical current. An amplifier is used to control the current so that the resistance, and thus temperature, remains constant. This temperature is dependent on the cooling imposed by the fluid flow passing over the wire. The output voltage represents heat transfer (cooling effect of the flow), and thus is a direct measure of the skin friction coefficient.

The output from all 9 gauges was recorded simultaneously at a frequency of 2 kHz, with a 1 kHz anti-aliasing filter, for a total of 98 seconds using 9 individual amplifiers mul-

tiplexed to a single output through an in-house constant-temperature anemometer (CTA) design commissioned specifically for this project. As the data acquisition software only allowed for a maximum of 65,253 samples in one recording, three consecutive readings were taken. This was also done for the wind-off measurements, which were taken before and after each individual case.

The output of the hot-film sensors was uncalibrated, and thus only semi-quantitative information could be acquired. Zhang et al. [16] evaluated the hot-film data as quasi-wall-shear stress (τ_Q , Eqn. 3.1). The mean output of τ_Q of each gauge gives an indication of the wing's shear-stress coefficient (C_F) distribution. It should be noted that E is the mean output voltage and E_0 is the average of the wind-off voltage from before and after the test.

$$\tau_Q = \left(\frac{E^2 - E_0^2}{E_0^2} \right)^3 \quad (3.1)$$

$$CV = \frac{\sigma}{\tau_Q} \quad (3.2)$$

The coefficient of variance (CV, Eqn. 3.2) is used to indicate boundary-layer state. In the laminar state CV is low because the flow is steady, whereas CV is higher in the turbulent boundary layer because of the velocity fluctuations that characterise turbulent flow. A sharp rise in CV can be used to determine the point of boundary-layer transition. It should be noted, however, that a high value of CV can also be the result of a low mean as well as of a high standard deviation. Zhang et al. [16] used normalised RMS to characterise the boundary-layer state. However, it appears that their RMS is of an AC coupled value and hence their normalised RMS is equivalent to CV.

The time-history of the hot-film output voltage was transformed into a PSD. This was

completed by first splitting the time-history into individual segments containing 1024 samples; each of these was zero padded to 8192 points. The PSD for each segment was computed and then averaged. The average of all wind-off segments was then subtracted from the average of all the wind-on segments to give the final PSD for each gauge.

Given that the coefficient of variance of the hot-film data is used as a indication of turbulence, the standard error for each gauge changes for each case. Gauges positioned in the turbulent boundary layer will inherently have a larger uncertainty as a result of the larger variance. An example shear stress distribution, with error bars indicating the uncertainty for each measurement location, is given in Figure 3.16. In this distribution, the first 4 locations are situated in the laminar boundary-layer region, whilst the final three are in the turbulent region. This demonstrates how the uncertainty is much smaller in the laminar boundary layer, however the uncertainty is still sufficiently small in the turbulent region to make readings meaningful.

3.4.3 Laser Doppler Anemometry

LDA (Laser Doppler Anemometry) is a non-intrusive measurement technique that uses a laser that generates a fixed-frequency coherent beam to illuminate a particle moving in the flow. The particle creates a differential Doppler shift in the frequency of the reflected light that is proportional to its velocity.

A single laser is used to measure an individual component of velocity. In this work a two-channel LDA system was used, a green laser of wavelength 514.5 nm was used for the streamwise component, and a blue laser of wavelength 488.0 nm was used for the vertical component. For each channel, two beams of equal intensity, one of which was phase-shifted by 40 MHz, were generated and focussed and crossed at an angle κ at the focal

length of the lens (2290 mm) to give a measurement volume. The interception of these beam pairs creates interference fringes due to superposition, so when a particle moves through the measurement volume, crossing the fringes, light is scattered. The scattered light contains a Doppler shift, which is proportional to the velocity of the seeding particle along the direction of the beam pair axis. The scattered light is captured by receiving optics at the back of the probe and converted to an electrical signal by photomultiplier tubes. Scattered light from particles travelling outside the measurement volume do not get detected. The system can be expanded to include a second component (two-component) by adding an additional laser, where the beam pair axis is perpendicular to the first pair axis. Expanding to a three-dimensional system requires a third laser and a second probe.

The diameter of the measurement volume (V_D) is calculated using Eqn. 3.3, where f is the focal length of the lens, λ the wavelength of the laser, D_B the diameter of the beam, and κ the beam angle. The diameter for the green and blue beams was calculated to be 0.204 mm and 0.259 mm respectively. The length of the measurement volume (V_L), given by Eqn. 3.4, was calculated as 7.24 mm and 9.19 mm for the green and blue lasers respectively.

$$V_D = \left(\frac{4f\lambda}{\pi D_B} \right)^3 \cdot \left(\frac{1}{\cos \kappa} \right) \quad (3.3)$$

$$V_L = \left(\frac{4f\lambda}{\pi D_B} \right)^3 \cdot \left(\frac{1}{\sin \kappa} \right) \quad (3.4)$$

The probe was orientated on the traverse so that it was perpendicular to the wind-tunnel centreline with the green lasers in the horizontal plane (measuring x-direction) and blue lasers in the vertical plane (measuring y-velocity). A photograph of the LDA system in

use in during experiments can be seen in Figure 3.19. A 2.3 m lens was used to put the measurement volume at the wind tunnel centreline, the x and y locations of the measurement volume were then altered by the traverse being programmed to move between set points. For each measurement line approximately 40 - 50 points were taken. The spacing of these points varied between 4 mm in low velocity gradients, and 1 mm in higher velocity gradients. The grid was determined by first making a coarse run, with 2 or 4 mm spacing and then refined for the next run. The minimum distance from the ground that the measurement volume could be placed, before being impeded by the staging, was 50 mm.

Wake surveys were taken behind the wing. The endplates prevented the beam being placed any closer to the wing's trailing edge than $x/c = 1.4$. Three measurement lines were taken downstream of the trailing edge at $x/c = 1.5, 2.0$ and 3.0 ; a diagram of these planes is given in Figure 3.20.

A TSI Six-Jet Atomizer was used to produce seeding for the measurements. Seeding was introduced into the tunnel by a nozzle placed in the collector. The nozzle was attached to a beam so that the seeding was produced near to the centre of the collector. The seeding then travelled around the closed-return system to reach the test section. The advantage of this method is that seeding is equally dispersed throughout the air stream by the time it reaches the test section, the disadvantage being that the turbulence-reduction screens had to be cleaned after every few runs. At 15 ms^{-1} and 20 ms^{-1} this method was able to produce data rates of up to 1.2 kHz in the streamwise channel (green), and 1 kHz in the vertical channel (blue). A typical data rate, however, would be approximately 800 Hz and 1 kHz in the streamwise and spanwise channels respectively. As such, for the 15 ms^{-1} and 20 ms^{-1} cases measurements were taken in coincident mode, meaning that a measurement was only taken when it was recorded by both beam pairs at the same time. A total of 5000 coincidental samples were recorded at acquisition data rates of approximately 1.1

kHz for each beam (green and blue); however, the total sampling time for each point was 20 - 40 secs because of the coincidental limitation. Analysis of the unsteady measurements for data points showed that the average time between samples was approximately 0.007s; equivalent to a data rate of 143 Hz. Autocorrelation of the unsteady measurements showed a time integral $T_{int} = 0.0008s$, thus in order for independent samples to be taken samples should be recorded at 0.0016s intervals. Hence, in 20 secs over 12,000 independent samples could be taken.

At 25 ms^{-1} the seeding was not sufficient for acceptable data rates to be achieved. Only 500 Hz in the streamwise and 120 Hz in vertical channel could be achieved. To give 5000 coincident points this results in times of up to 3 min per data point. This was not deemed usable as during testing the chillers were out of order, and thus the wind could not be cooled. Whilst this was not a problem for wind velocities of 20 ms^{-1} and below, the temperature would increase at too high a rate for 25 ms^{-1} . This temperature rise causes the Reynolds number to change, and thus taking 2 hours to do each profile was deemed inappropriate because of the Reynolds number variation. For the 25 ms^{-1} runs, it was therefore decided to record only the streamwise channel. With a data rate of approximately 500 Hz and 10,000 samples per point being taken, each profile took approximately 20 min to complete. A total of 10,000 samples per point were taken to give a margin of error ($\sqrt{\frac{1}{N}}$) of 1%.

A single measurement line was also taken underneath the wing, by shining the laser through the hole in the endplate usually used for allowing cables out of the wing. This gap was wide enough that the beam pair for the streamwise velocity component could be shone through, however the vertical component was not measured as it greatly reduced the length of the measurement line as it was blocked by the wing at the upper end of the measurement line, and the endplate at the lower end of the measurement line. Thus, for

the under-wing measurements on the x-component of velocity is given. This line was taken at $x/c = 0.375$ and started approx. 2 mm below the wing surface. Similarly to the 25ms^{-1} case, where only the stream-wise component was measured, 10,000 samples per point were taken.

For surface measurements the wind tunnel was operated in constant Reynolds number mode, so that the wind velocity would change accordingly with temperature change across the run. As LDA measurements are velocity based, however, the velocity change across the run due to the tunnel temperature change would skew the results; unless each point could be normalised by the freestream velocity at the exact time they were taken. Unfortunately with the current setup that was not achievable. The tunnel was, therefore, operated in constant velocity mode whilst LDA measurements were taken. Before each run, however, constant Reynolds number mode was enabled as the tunnel was run up, so that the required speed to match the correct Reynolds number could be seen, then constant velocity enabled at that speed. This was conducted for each individual measurement line. Thus for a given Reynolds number, three separate wind speeds may have been used, but as the measurements are non-dimensionalised by the appropriate wind speed, they are comparable. Although this method means that there is some variation in Reynolds number, and this variation is stated for each measurement, it is the most accurate way of conducting the experiment with the current setup.

The Reynolds-number variation during each test was calculated using the diagnostics of temperature and pressure during the experiment. The maximum Reynolds number was observed to be approximately 1.31% higher than the target Reynolds number for the highest wind speed cases; however this fell to approximately 0.45% for the lowest wind speed cases. Although it is desirable to keep the Reynolds-number variation during the test to a minimum, it was not possible due to the chillers being broken during the experiment,

Table 3.7: Average Reynolds number and temperature change recorded by diagnostics for LDA wake profiles

Test Re	Average Re change during test	Average temperature change during test
$Re = 3.59 \times 10^5$	1566 (0.45%)	0.6 °C
$Re = 4.79 \times 10^5$	5992 (1.25%)	1.9 °C
$Re = 5.99 \times 10^5$	7824 (1.31%)	2.8 °C

and thus not possible to control the air temperature. It should also be noted, however, that there is a 20% difference between each target Reynolds number; hence even a 1.3% Reynolds-number variation is not enough to significantly impact the results. Moreover, the tunnel was allowed to cool and the velocity reset to the correct magnitude for each measurement line.

The sampling uncertainty of each data point is dependent on the turbulence stresses, whereby the higher the stress the greater the uncertainty. The maximum sampling uncertainty, at a 95% confidence level, in the measured velocity for any data point was found to be $\pm 0.027 \text{ ms}^{-1}$; this data point was that which occurred at the centre of the wake, where the highest turbulent stress occurred. Outside the wake the sampling uncertainty reduced to $\pm 0.0039 \text{ ms}^{-1}$.

3.4.4 Static Pressure Measurements

Surface pressure measurements were taken on the suction surface of the 2D wing through thirty-one tappings of equal spacing from $x/c = 0.1$ to $x/c = 0.85$ at varying ground clearances, wing incidences, and Reynolds numbers in free- and forced-transition cases. To reduce the interaction between the tappings they were also spaced in the spanwise direction. This gave two diagonal lines of tappings of span 50 mm. A diagram of the tapping locations is given in Figure 3.21. The tubes from the tappings were connected to a scanivalve. Each tube was then individually tested to ensure that there were no leaks in

the system. This was accomplished by pressurising the tube, with the tapping sealed over, with a Druck pressure calibrator and confirming that the pressure was maintained.

Freestream dynamic and static pressures were measured by a Pitot-static tube mounted above the test section. Prior to placing the model in the tunnel a Pitot-static tube was also placed at the location where the model would be situated so that the dynamic pressure at the model location could be found from the Pitot-static tube during the test. A scaling factor of 0.98 was applied to the measured dynamic pressure as a result of this test. The scanivalve (wing-surface static pressure) and freestream dynamic pressure were connected to a $0 - 10''H_2O$ (gauge) transducer whilst the static pressure was connected to a $\pm 1''H_2O$ transducer.

Prior to each run zero readings were taken and the transducer nulled. The static pressure was then logged at 2 kHz with a 1 kHz filter for 20 secs for each location consecutively. This ensured that enough independent samples were made such that a 95% confidence level was met. This was found by finding the integral time scale from the autocorrelation of an initial run, which showed that samples must be taken 0.0054 sec apart in order to be independent and that 1600 samples were required to give results at a 95% confidence level; hence a total sampling time of at least 8.6 secs was required.

Repeatability tests were conducted by running the same cases consecutively. The first four runs were taken at the end of the day, and the fifth as a repeat in the morning. The static pressure at each location was within 1% of the measurement of the first tests. Morning repeats were conducted each day of testing, again results stayed within 1%.

The static pressure measured at each location was normalised by the freestream static and dynamic pressure using Eqn. 3.5. As the wind tunnel was operated in constant Reynolds number mode, the dynamic pressure changes as the air heats up. By taking readings of

dynamic pressure simultaneously with each tapping this change is accounted for in the normalisation process.

$$C_P = \frac{P - P_\infty}{q} \quad (3.5)$$

3.4.5 Surface-Flow Visualisation

Using an oil or paint to visualise surface flow features is a useful technique regularly employed by aerodynamicists. In essence the paint allows the user to view surface shear stresses and as such allows characteristics such as separation, reattachment, flow regime, vortical flows, and general flow direction to be viewed. This is clearly a very helpful technique, however some care must be taken when analysing the surface flow patterns with reference to off-surface flow structures as the flow-visualisation gives no indication of what is occurring even a small distance from the surface. Therefore, some features that appear significant in the flow-visualisation can actually be energetically quite small, and vice versa.

The paint consisted of a mixture of paraffin, oleic acid and ultra-violet florescent pigment to produce a low-viscosity liquid. The oleic acid is an important inclusion as it is an emulsifying agent (anti-coagulant) and as such ensures dispersion of the powder throughout the paraffin to prevent clumping. The paint is applied to the surface in question using an aerosol spray bottle immediately before running the tunnel. The mixture is designed to give a low-viscosity fluid that can be moved easily by the air to create a pattern, but not susceptible to the acceleration/deceleration of air during start up and shutdown of the wind. The model is run with the wind on for 10 to 45 minutes in order for the paraffin to evaporate and leave dried powder trails on the surface. The drying time varies as it is dependent on the freestream velocity, as higher wind velocities also increase the tunnel temperature aiding in evaporation of the paraffin. Once dry the model was removed to a

dark room and then photographs taken under ultra-violet light.

Post-processing of the photographs included transformation to an 8-bit grey-scale palette, followed by a histogram-equalisation process. This technique enhances contrast, which allows the streaklines left on the surface to be more easily deciphered.

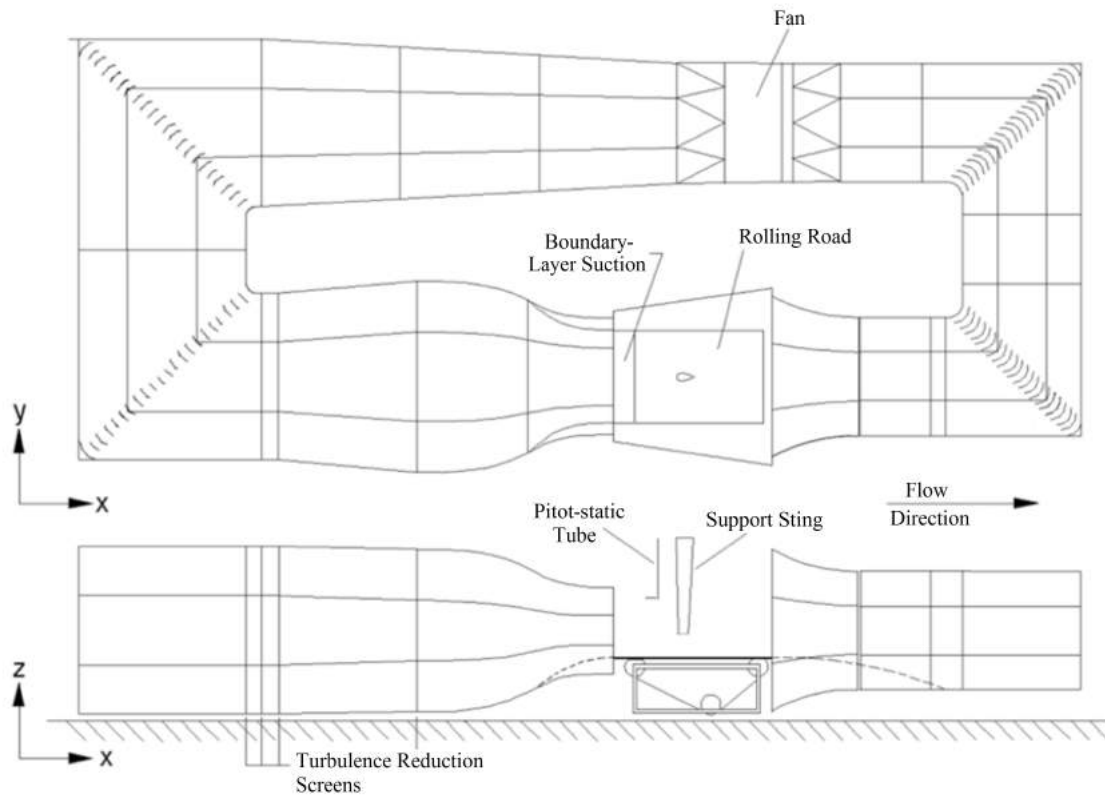


Figure 3.1: Schematic of DS Houghton Wind Tunnel

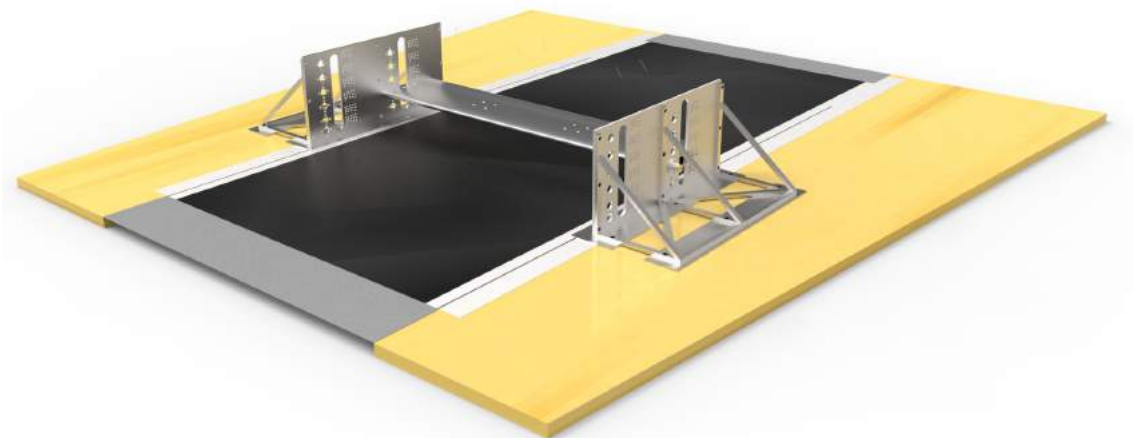


Figure 3.2: CAD illustration of 2D wing in wind tunnel test section

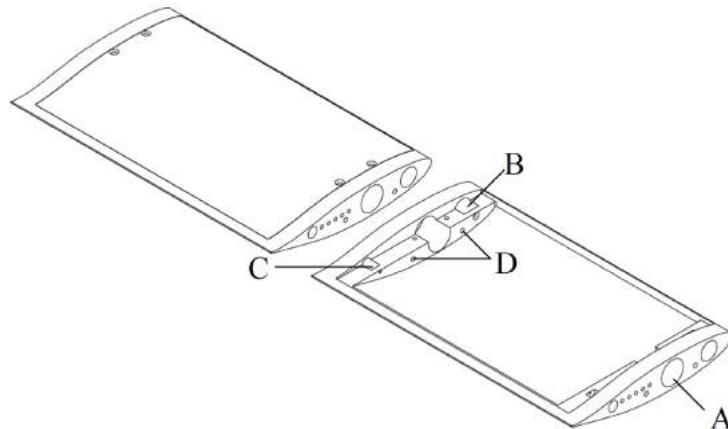


Figure 3.3: Diagram of 2D wing section, showing upper and lower sections fitted together (left), and lower section (right) in isolation. Letters indicate cable access hole (A), large-brass pin slot (B), small-brass pin slot (C), and fixing bolt holes (D)

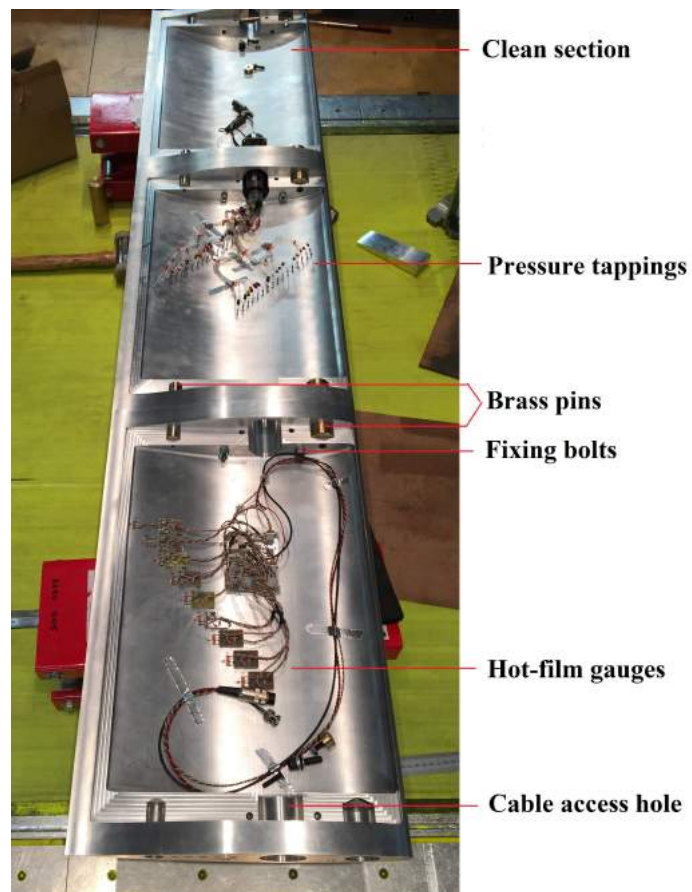


Figure 3.4: Photograph showing the instrumentation inside the 2D wing

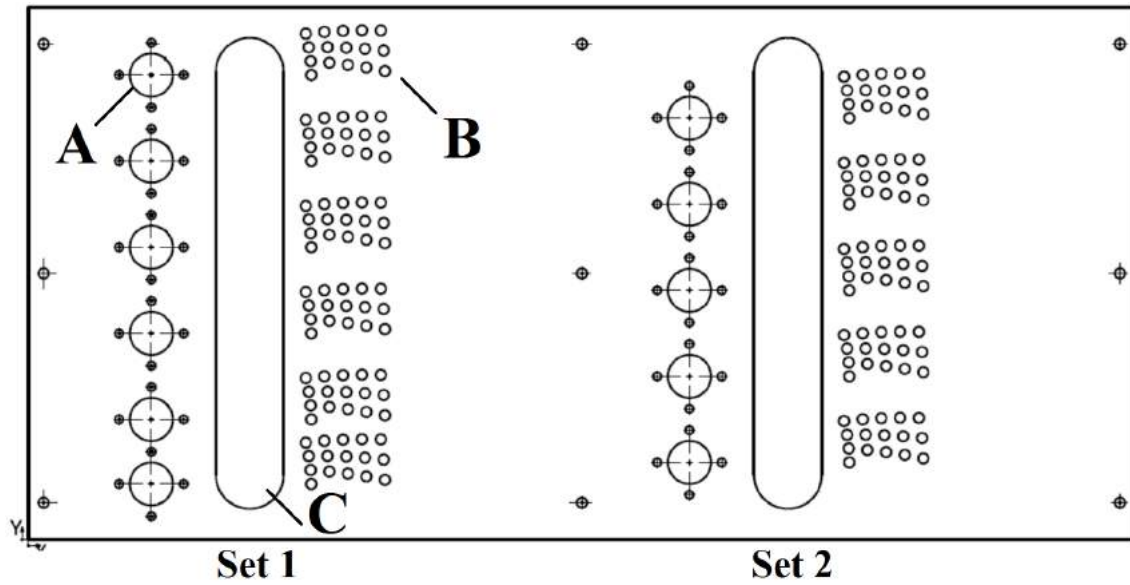


Figure 3.5: Schematic of endplate geometry (2D wing)

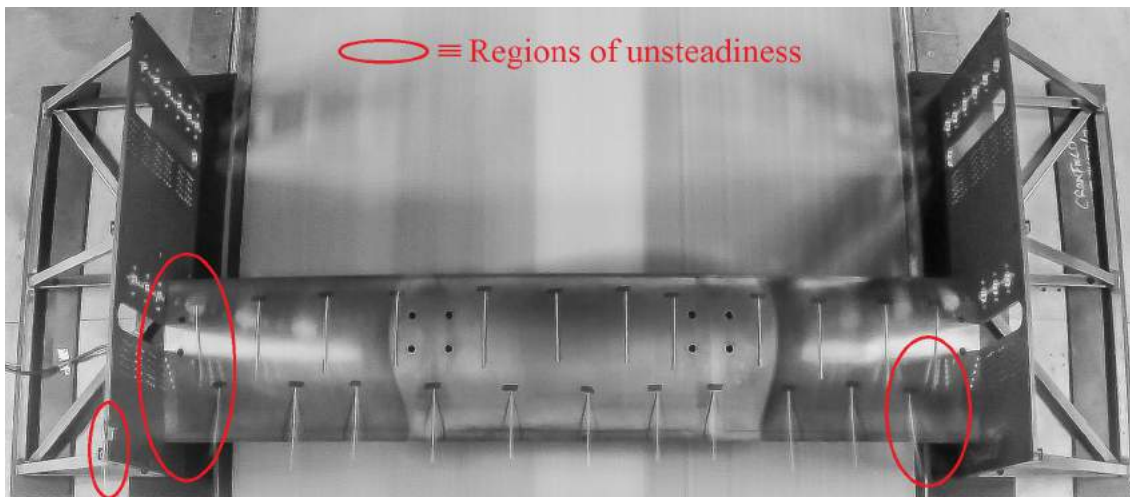


Figure 3.6: Top view (looking vertically downwards) of wing fitted with wool tufts with tunnel running at 25ms^{-1} (2D wing, flow moving top to bottom)

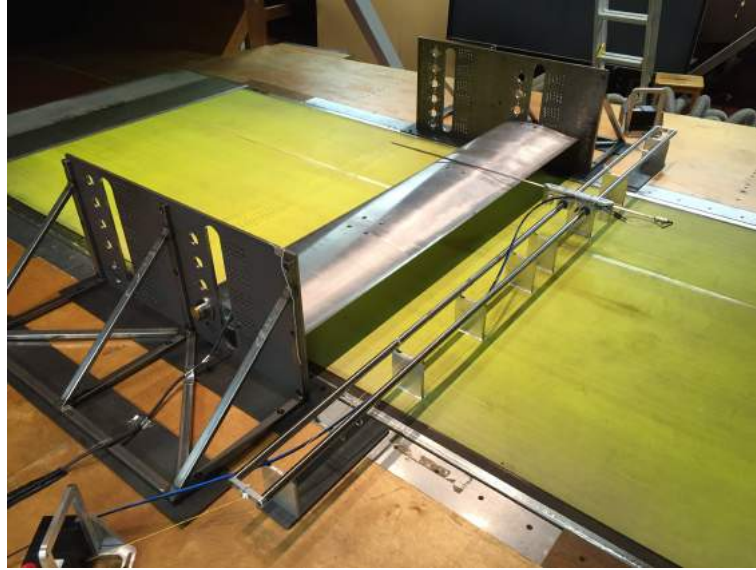


Figure 3.7: Photograph of setup for spanwise pressure tests (2D wing, flow from left to right)

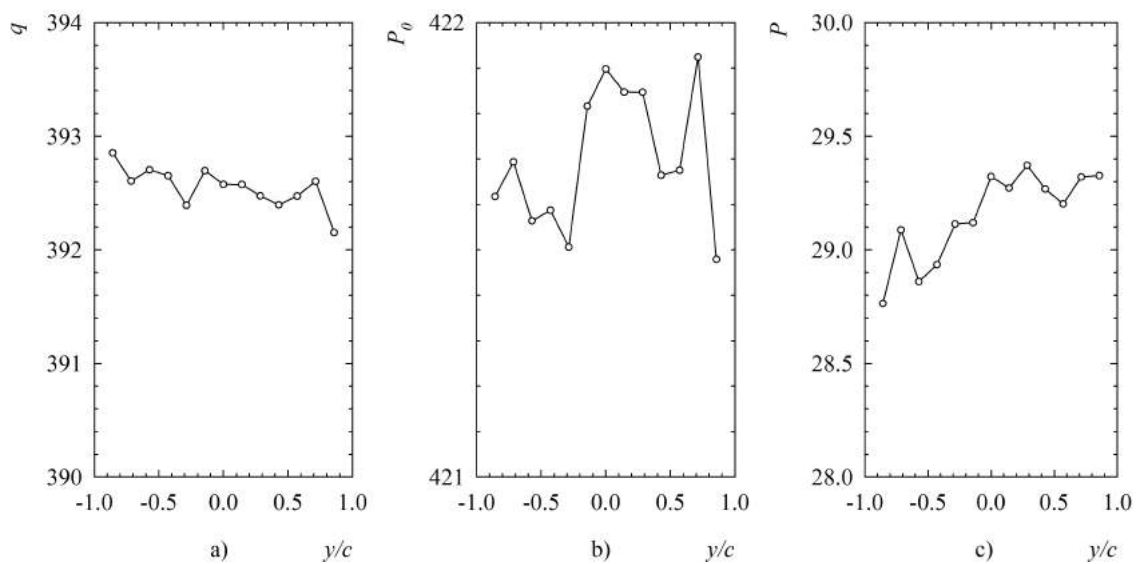


Figure 3.8: a) dynamic pressure, b) total pressure and c) static pressure at various spanwise locations measured with Pitot tube ($y/c = 0$ is wind tunnel centreline, 2D wing)

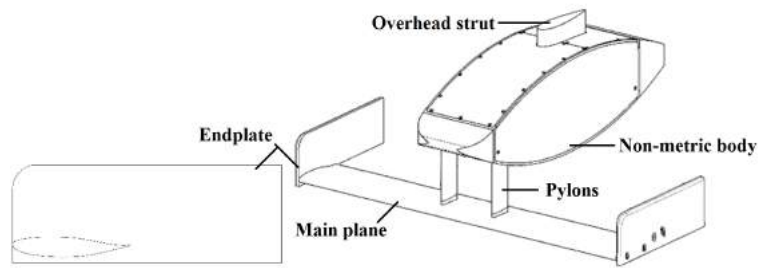


Figure 3.9: Diagram of single-element wing (CSW-SE)

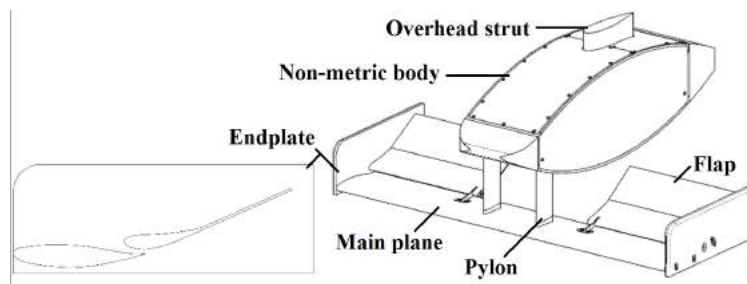


Figure 3.10: Diagram of multi-element wing (CSW-ME)



Figure 3.11: Photograph of Tatuus F4-T014 car as used in 2015 ADAC Formel 4 series [118]



Figure 3.12: Photograph of Dallara T12 car as used in 2015 Formula V8 3.5 series. Copyright Formula V8 3.5 [119]

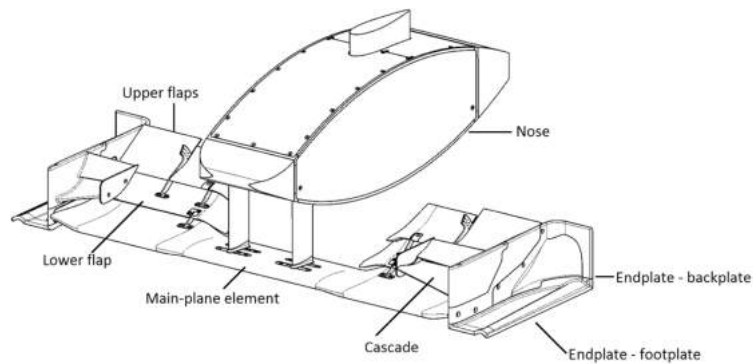


Figure 3.13: Diagram of F1 Wing

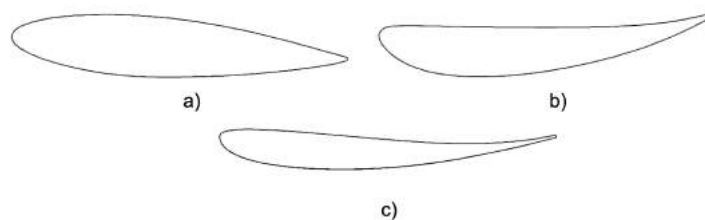


Figure 3.14: Aerofoil sections used at the a) central region, b) mid-semi span (taken at 225 mm from centreline) and c) wing tip of the F1 wing

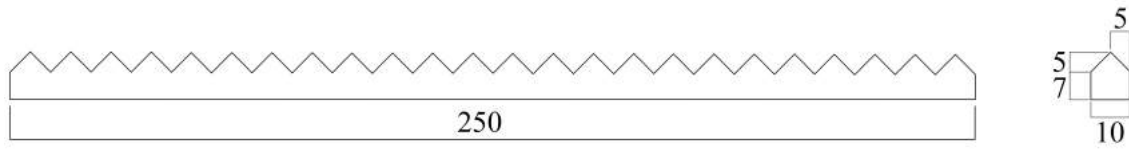


Figure 3.15: Dimensions of serrated trip

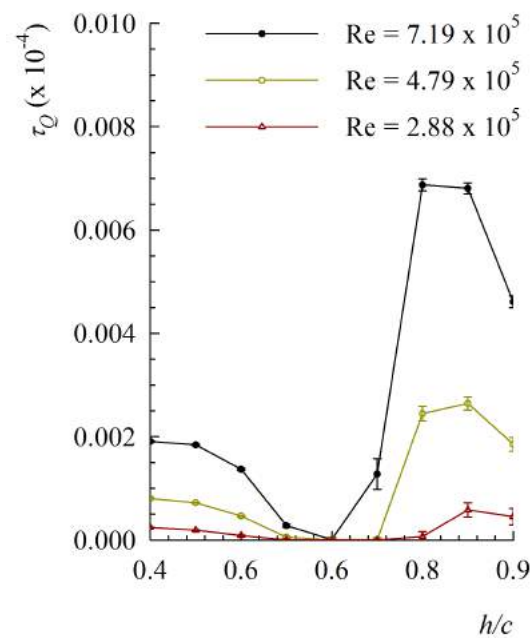


Figure 3.16: Quasi-shear stress distribution on the suction surface of 2D wing with error bars showing uncertainty at each measurement location for varying Reynolds number

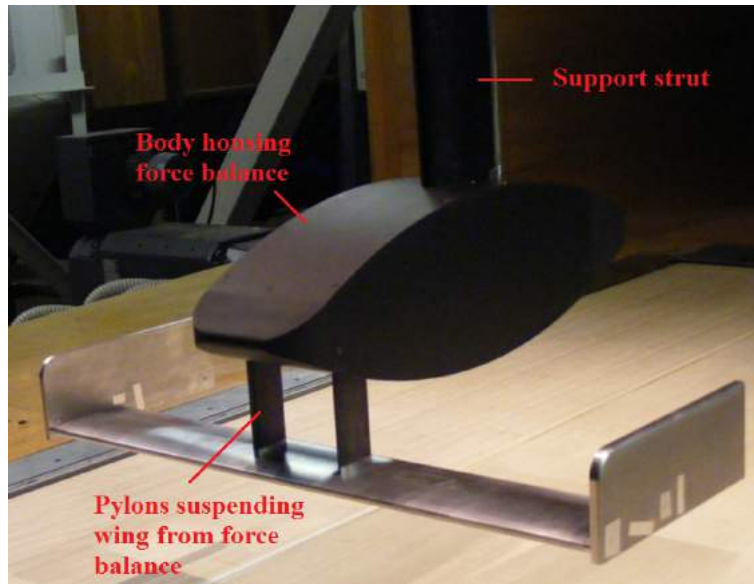


Figure 3.17: Photograph of three-dimensional wing (single-element configuration) installation

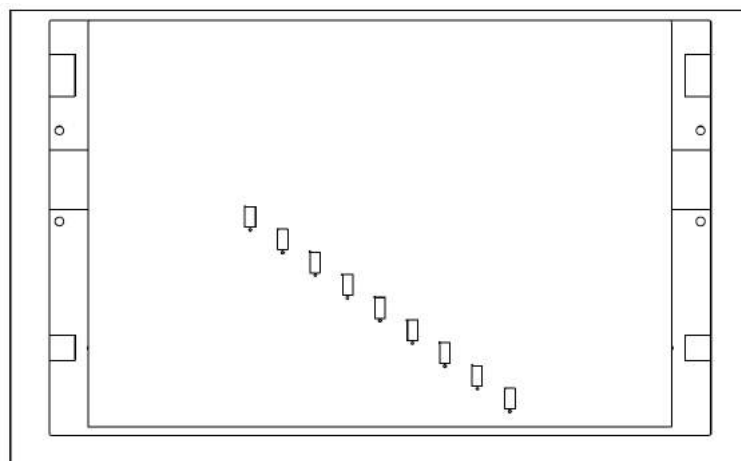


Figure 3.18: Diagram of hot-film sensor locations



Figure 3.19: Photograph of the LDA system in use

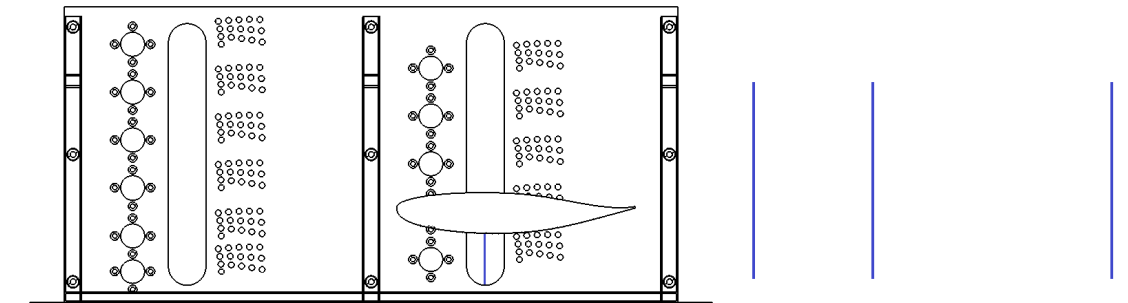


Figure 3.20: LDA measurement planes

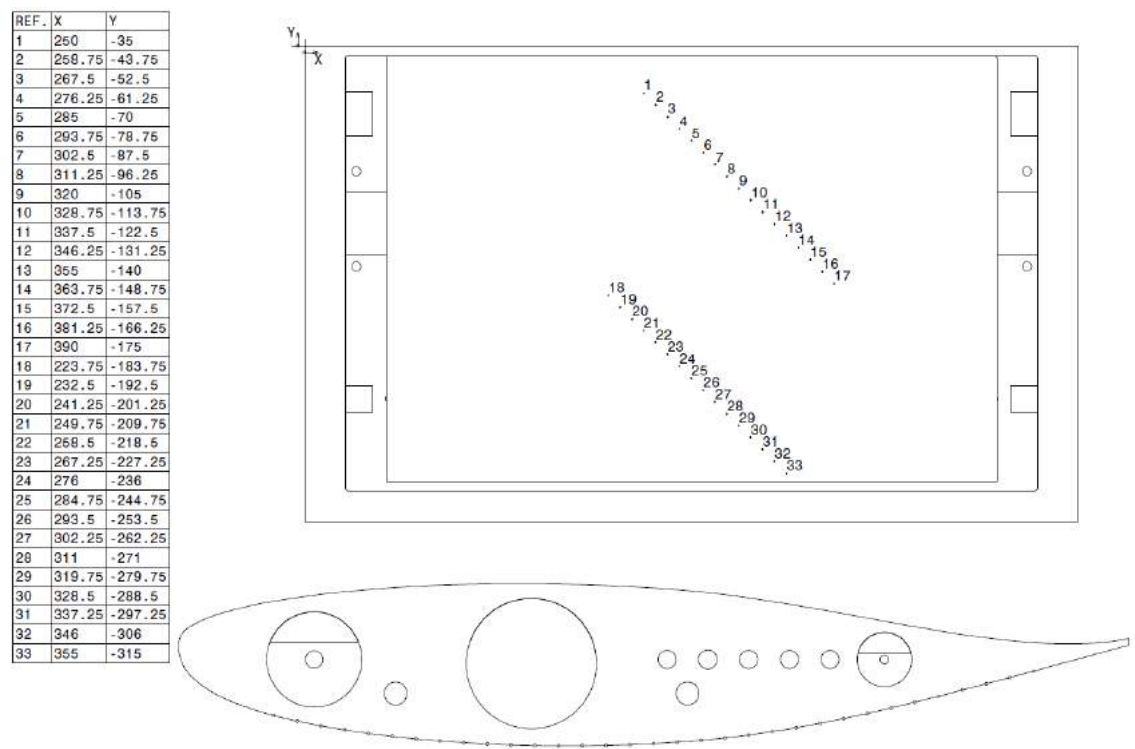


Figure 3.21: Static-pressure tap locations

Chapter 4

Description of Computations

4.1 Two-Dimensional Euler Simulation

4.1.1 Overview of MSES

MSES (Multi-Element Streamline Euler Solver) is a code developed by Prof. Mark Drela [53, 110]. It is based on the steady-state conservative Euler equations, as the name suggests, with a perfect-gas assumption, removing cross-stream pressure variation and considering only adiabatic freestream flow. Although the Euler equations are an inviscid formulation, it is assumed that the viscous effects are restricted solely to the boundary layer, and thus, can be modelled separately. So, whilst the inviscid flow outside the boundary layer is modelled by the Euler equations, the boundary layer is calculated using two equations based on the Prandtl boundary-layer formulations. The viscous part of the flow is assumed to displace the inviscid flow away from the wall, by making the edge of the boundary layer (δ^*) the boundary condition for the inviscid flow, creating a new effective shape around which the Euler equations are solved. The resulting system of non-linear

equations is discretised onto a grid, which is formed from the aerofoil coordinates, and solved using the Newton Method. The advantage of producing an intrinsic grid from the aerofoil coordinates is that the continuity and energy equations can be replaced by a condition of constant mass flux and constant stagnation enthalpy along each streamtube. This simplifies the process by reducing the number of unknowns from four to two. Further details of the grid generation are given in the next section.

Laminar-to-turbulent transition is predicted by a simplified variant of the e^N method [111], known as the envelope method, which determines the amplitude of the greatest Tollmien-Schlichting wave frequency at each node on the aerofoil surface. The premise is that when the spatial amplification of disturbances reaches a critical point, which is defined by the user, transition will occur. This value, known as the critical amplification factor (N), relates to the disturbance level in which the aerofoil operates. A lower value of N will be achieved closer to the leading edge, thus transition will occur at a more upstream chordwise position, whereas a higher value has the opposite effect. Thus, a lower value is analogous to the aerofoil operating in higher freestream turbulence. MSES also allows for forced-transition cases to be simulated by specifying the point where the turbulent boundary layer begins, unless the e^N criterion has already been fulfilled further upstream.

In a similar manner to other panel method simulations [62, 87], the ground plane was implemented using the mirror-image technique. This is an established concept for simulating the presence of a moving ground in a wind tunnel that involves mirroring the geometry such that the ground plane is represented by the dividing streamline, which is equally acted upon by both geometries. The weakness of this method is that the streamline does not remain at constant freestream velocity, as the moving ground would. It does not, therefore, account for the residual boundary layer that forms due to the velocity gradient normal to the floor.

4.1.2 Grid Generation

The aerofoil coordinates were used to produce the two aerofoil sections required for the simulation. It should be noted that all dimensions in MSES are normalised with aerofoil chord length. The grid was produced in a domain that extended $2c$ upstream from the leading edge, $5c$ downstream of the trailing edge, and $2.5c$ above and below the centre-line of the relevant section. This domain is shown in Figure 4.1a.

A total of 250 nodes were placed on each aerofoil section, as shown in Figure 4.1b. The grid was then produced by displacing the horizontal streamline shown in Figure 4.1a above and below each aerofoil, then splitting it by the surface nodes and 45 normal lines upstream of the leading edge and 50 lines downstream of the trailing edge. The grid is then smoothed, giving the final grid (Figure 4.1c & 4.1d). MSES is limited on the number of streamlines (max. 70) and normal lines (max. 480) that can be used to produce the grid. In total, approximately 70 streamlines and 350 normal lines, depending on configuration, were used to create the grid.

4.1.3 Solving

For each simulation the Mach number and chord-based Reynolds number were prescribed. The Mach number is equivalent to the wind velocity used in the experiments to produce the target Reynolds number.

The critical amplification factor used for all simulations was e^7 . Smith & Gamberoni [112] correlated the transition point found in experiments with the real part of the Orr-Sommerfeld eigenvalue and determined that when disturbances had grown approximately 8000 times (e^9) transition would occur. Van Ingen [111] suggested lower values of e^7 or e^8 should be used. Comparison of surface-flow visualisation and numerical results for a

free-transition GA(W)-1 wing in ground effect by Correia et al. [81] found a criterion of e^7 to be most representative of test conditions in the DS Houghton wind tunnel. As a result, a criterion of e^7 was used in this study, other than in cases where varying freestream turbulence was simulated.

For forced-transition simulations, the location of transition was prescribed at the same chordwise locations that the leading edge of the boundary-layer trip was placed in the forced-transition experiments. In keeping with the experiments, transition was forced only on the suction surface. Additionally, in order to satisfy the mirror-image technique, it was prescribed on both aerofoil sections.

For the solving process an inviscid solution was first derived; viscous effects were then enabled and the solution iterated until a final converged solution was reached. This method both reduced computation time and produced a more stable simulation, particularly at lower ground clearances for which divergence had been a problem when attempting to solve with viscous effects directly from the initial conditions. This is a consequence of the code being unable to model flow downstream of trailing-edge separation. The solution was considered converged once the scaled residuals decreased below 10^{-3} .

4.2 Three-Dimensional RANS

Reynolds-Averaged Navier-Stokes (RANS) simulations were primarily employed in the present work because despite advances in CFD methods, such as Large Eddy Simulation (LES) and Detached Eddy Simulation (DES), the RANS approach is still the work horse of the motorsport industry. The RANS approach involves the solving of only the average flow, whilst DES and LES methods are time dependent, which leads to dramatically shorter

solving time for RANS. Moreover, as LES and DES solve for eddies in the flow, they require a mesh much finer than RANS.

LES fully solves the time-dependent Navier-Stokes equations for all but the smallest eddies in the flow, by way of a low-pass filter that removes small-scale information (time and length scales) from the numerical solution. As the Reynolds number increases, however, the smallest scale that appears in the flow becomes increasingly smaller, and thus the mesh must be much finer. Thus, for motorsport-application Reynolds numbers, the mesh requirement is often too large to make it a viable method. DES is a hybrid LES-RANS method that uses LES in areas of large length-scale eddies, but uses RANS for small-scale eddy regions such as the boundary layer; therefore, a much coarser mesh can be used than is needed for LES. Most motorsport teams still employ RANS because of the faster solving time.

RANS has been employed in this study because it is the most applicable to the given problem. DES would offer no substantial improvement as the boundary-layer flow is modelled by RANS, hence for a wing without large flow separation DES is not required. LES would solve the boundary-layer flow and is capable of modelling laminar-turbulent transition. The Reynolds number of the experiment, however, is sufficiently high that a very fine grid would have to be generated for LES, which with the computational resources available would have led to very long solving times. Moreover, due to the complex geometry a structured mesh would have been difficult to generate, thus increasing solving times even further.

4.2.1 Facilities & Software

All RANS CFD undertaken in this work was conducted using Ansys FLUENT 15.0 and ICEM CFD 14.5. FLUENT is a finite-volume code that is widely used both in industry and in academia, and considered by many to be the industry-standard CFD package. ICEM is a meshing software, described below.

Simulations were conducted on Cranfield University's high-performance computing network, Astral. This is comprised of a dedicated cluster of nodes with a total of 1280 cores available and a peak measured performance of 18.8 TFlops. In this research 128 CPU cores and 512 GB of shared memory were used in each simulation.

4.2.2 Geometry Preparation

The meshing software used for this work, ICEM CFD, is a surface-based modelling program. The CAD files that were used for the manufacturing of the experimental models were also used as the basis for creating the CFD geometry. A number of simplifications were made to the geometry to prepare it for meshing: the elimination of bolt holes, joints between multiple sections and brackets for the flap elements so that simple, clean surfaces were left. The overhead strut, to which the wind tunnel model was mounted, was not incorporated into the geometry as it was deemed unnecessary due to its small effect on the wing. Only half of the wing was modelled in order to reduce the mesh size, as it was assumed that flow is symmetric about the wing's centreline

4.2.3 Grid Generation

Due to the complex nature of the geometries being modelled, a fully unstructured approach to meshing was taken. The wing was placed in a domain that extended $7l_m$ ($l_m =$

Total Model Length) downstream of the rearmost point of the model and $3l_m$ upstream of the wing's leading edge. The domain extended $3l_m$ laterally from the symmetry plane and $3l_m$ in total height; giving a blockage of 1.85%. The boundaries were far enough from the model that the static pressure remained at freestream magnitude ($C_p = 0$) i.e. they were not influenced by the wing.

Slices of the computational grid for the single-element wing and multi-element wing are shown in Figures 4.2 and 4.3. The total number of elements differed depending on the model configuration, however the same approach was used for each case. The total number of elements ranged between approximately 26×10^6 for the single-element wing and approximately 45×10^6 for the multi-element and F1 wings.

Triangular elements were produced on all wing surfaces to a maximum size of 0.75 mm using an Octree formulation. A minimum size, the limit to which the software could refine in areas of high curvature, was set to 0.1 mm. Prism layers are grown from the surface elements in unstructured meshes in order to place a large number of cells in the boundary layer, specifically in the wall-normal direction. For the grids used in the present work a total of 21 prism layers were extruded from the surface elements. The total prism-layer height was enough to capture the entire boundary layer profile, up to at least 99% thickness, at all points along the wing. The prism layers were grown at a rate of 120% from an initial height equivalent to $Y^+ < 1$ on all wing surfaces. Prism layers were also placed on the ground underneath the wing so that the residual boundary layer, which forms as a result of the higher-than-freestream velocities that are produced under the wing, is modelled correctly. A total of 10 prism layers with $Y^+ < 1$ were used; again this ensured that the entire boundary-layer thickness was contained inside the prism layers. The volume was filled with tetrahedral elements created by a Delaunay triangulation; close to the wing these were refined to a maximum size of 0.75 mm. A second phase

of refinement to 5 mm was then placed in the near-field, and finally the elements were allowed to grow towards the far-field and domain boundaries with a maximum size of 100 mm

The final solution was determined to be mesh independent by systematically altering individual mesh parameters, such as Y^+ , volume refinement and surface element size. The maximum Y^+ value at any point on the wing was 0.7 for the final mesh. A finer mesh with a maximum Y^+ of 0.2 was also run; the force coefficient difference was found to be less than 10^{-4} between the two meshes, and negligible difference in the pressure distribution was observed. For both cases the total prism layer was enough to contain the entire boundary layer. Reduction of the maximum surface element size to 0.5 mm also displayed negligible differences. This is because the refinement criteria allowed a minimum size of 0.1 mm in areas of high curvature for both meshes. An initial mesh involved lower refinement of the volume elements around the wing, but this caused artificial dissipation of vortex cores; hence the two refinement boxes of 0.75 mm and 5 mm described above were required. Although the off-surface features were improved through this refinement, there was no difference in the aerodynamic forces. Despite this, the better spatial resolution for off-surface features was considered important and so the multiple refinement boxes were kept. Overall, as a result of this parametric study, the solution was considered to be mesh independent.

4.2.4 Boundary Conditions

A velocity-inlet condition was applied to the upstream boundary, and a pressure outlet prescribed to the downstream. The ambient conditions of the fluid were set to standard atmospheric pressure of 101325 Pa and temperature 288.15 K; the velocity-inlet condition was then set according to the Reynolds number being simulated. Turbulence quantities

were prescribed at the inlet, unless otherwise stated, as $k = 1.215m^2s^{-2}$, $k_L = 10^{-6}m^2s^{-2}$ (where applicable), and $\omega = 2.012s^{-1}$. These values were based on a parametric study of inlet conditions to find which offered the best prediction of experimental results; a description and the results of this study can be found in Appendix A.3. This study was conducted because when the turbulence intensity of the DS Houghton wind tunnel (0.3%) was used, no turbulent reattachment occurred i.e. a fully-laminar separation was observed. It was found that using a higher turbulence intensity (3%) to increase the turbulent kinetic energy (k) improved the solution.

Symmetry planes were used for both sides and the roof of the domain. Although only the plane at the wing centreline is a symmetry plane, a symmetry condition was also applied to the far-side and roof. This is because the symmetry condition enforces zero normal gradient to all quantities. Given that the domain is large enough to give constant static pressure across these boundaries, the symmetry condition is satisfactory as freestream values will be held by the boundary condition. The floor was defined as a no-slip wall translating in the same direction and with the same magnitude as flow from the inlet. Finally, all wing surfaces were defined as no-slip walls.

4.2.5 Numerical Method

As previously stated, the finite-volume code FLUENT was used for all simulations in this work. A three-dimensional, steady-state, pressure-based, segregated solver was employed. The pressure and velocity fields were coupled using the SIMPLE algorithm with pressure terms discretised with a second-order scheme. Due to the unstructured nature of the mesh, spatial gradients were evaluated using the Green-Gauss Node-Based method. Finally, the convective terms of all equations were discretised using a second-order up-wind scheme. This is in line with the recommendations of Lanfrit [115].

The flow-field was initialised with zero velocity, and turbulence quantities and static pressure equal to those prescribed at the inlet. A total of 12,000 iterations was completed during each simulation, taking approximately 20 hours (wall clock) to compute. Convergence was deemed to be achieved as the scaled absolute residuals of downforce, drag and velocity ($V_{x,y,z}$) had decreased below 10^{-5} at the end of the 12,000 iterations.

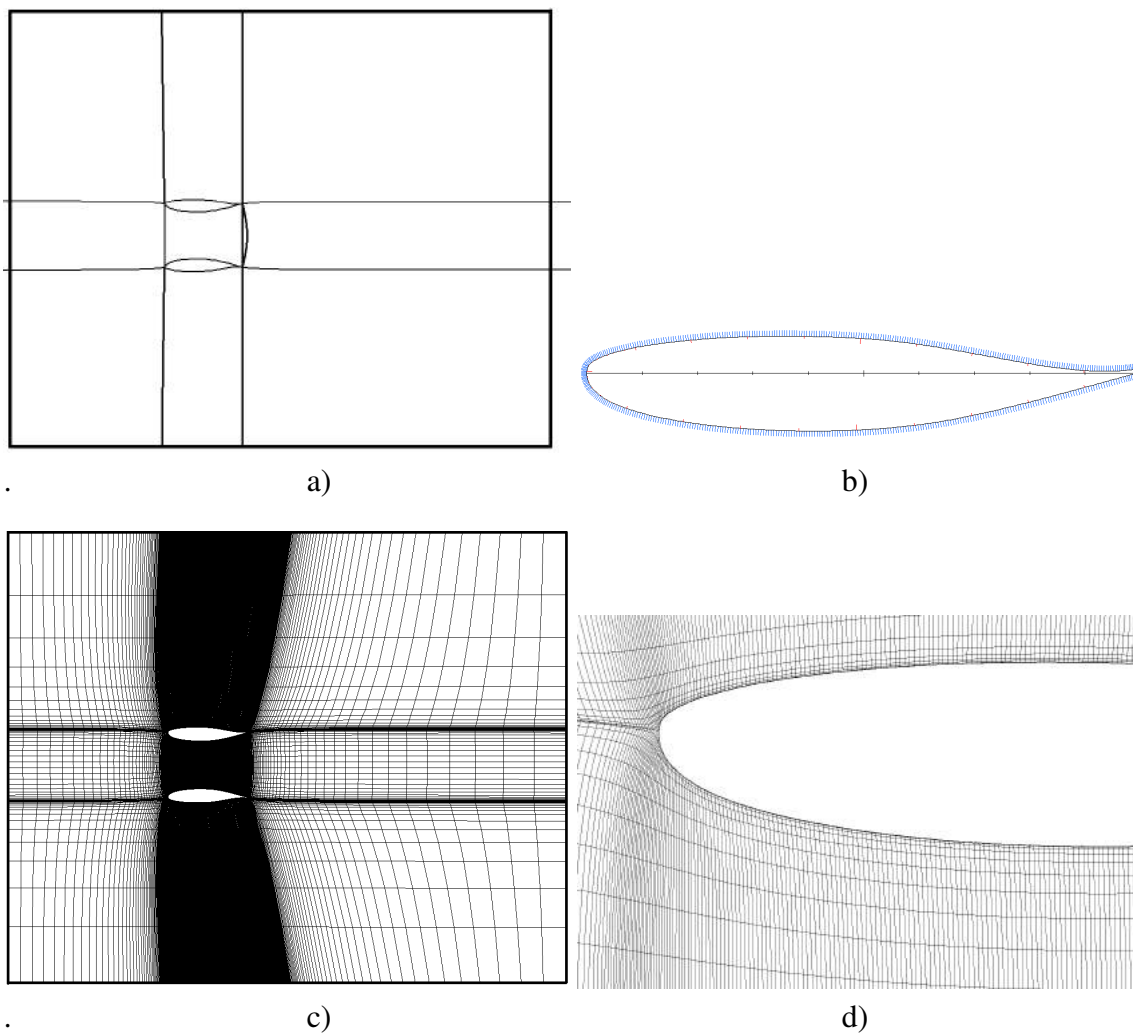


Figure 4.1: MSES grid generation

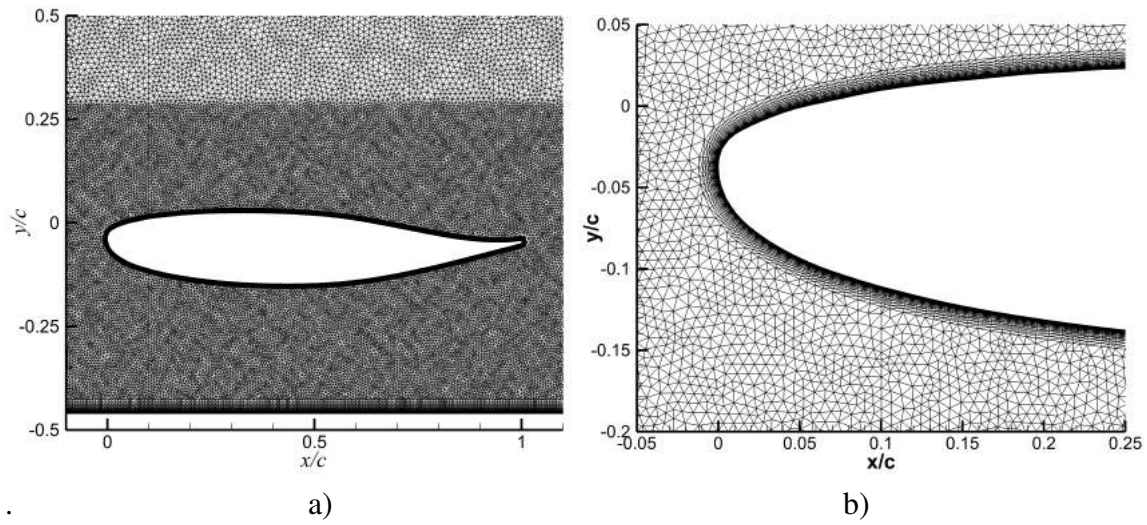


Figure 4.2: Slice of computational grid at symmetry plane for single-element wing

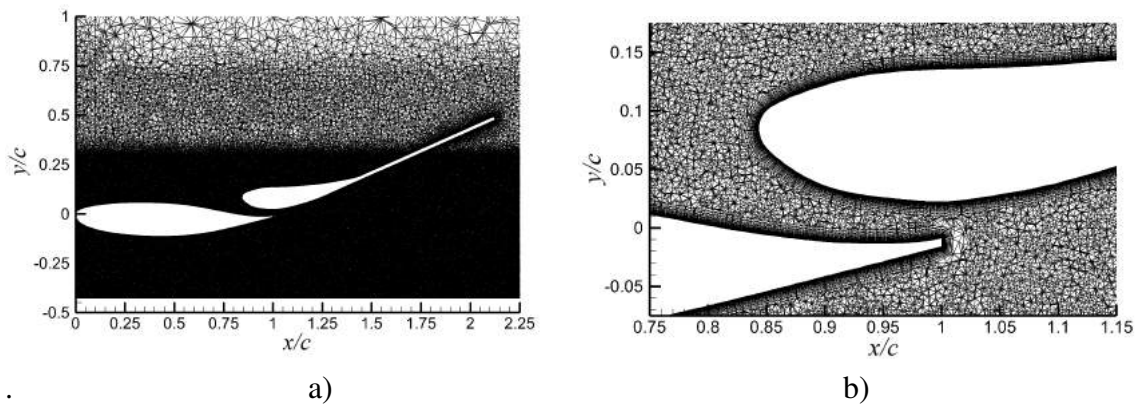


Figure 4.3: Slice of computational grid at symmetry plane for multi-element wing

Chapter 5

Results - Two-Dimensional Wing in Free Transition

In this section the characteristics of boundary-layer transition, and in particular the Reynolds-number sensitivity, of a GA(W)-1 aerofoil section are investigated using experimental and numerical techniques. This chapter contains observations of free transition in which remarks on the influence of incidence and ground proximity are made. A following chapter is then concerned with forcing laminar-to-turbulent transition at various chordwise locations so that the influence of laminar and turbulent boundary-layer states on the aerodynamic performance of the wing can be observed.

5.1 Experimental

5.1.1 Static Pressure Distribution

Static pressure measurements for the wing at zero incidence ($\alpha = 0^\circ$), at various ground clearances, and at various Reynolds numbers are given in Figure 5.1a-c. The ground clearance range $0.1 \leq h/c \leq 0.5$ is representative of a racing-car front wing operating range. Whilst a racing-car wing typically operates at $h/c \approx 0.3$, the three ground clearances given in Figure 5.1 cover the full working range. For the two higher ground clearances, a Reynolds number range of $2.88 \times 10^5 \leq Re \leq 7.19 \times 10^5$ was tested. As indicated in the introduction, this Reynolds number range is not as wide as the approximate operating Reynolds number range of a racing-car front wing, which is $3.5 \times 10^5 \leq Re \leq 1.4 \times 10^6$ (depending on the car's front wing and circuit). Although the results presented here are undoubtedly in the lower region of this range, they prove that the operating Reynolds number of a racing car is sufficiently low that laminar phenomena are present. Even at the highest Reynolds number achieved in this study for $h/c \geq 0.3$, $Re = 7.2 \times 10^5$, the laminar separation bubble still has a large effect on the static pressure distribution. Unfortunately, at the minimum tested ground clearance of $h/c = 0.1$, a maximum Reynolds number of only $Re = 4.79 \times 10^5$ could be tested due to the large load that the wing produced. Hence, in order to ensure structural integrity the maximum test speed was limited. At $h/c = 0.1$ the laminar separation bubble is prominent at all tested Reynolds numbers.

As discussed in the literature review, Knowles et al. [62] tested a GA(W)-1 aerofoil section at $Re = 4.28 \times 10^5$ for various incidences and ground clearances. Although there was low spatial resolution in their static pressure measurements, the influence of ground effect and incidence on the suction magnitude were clear. The results did not conclusively show the presence of a laminar separation bubble, although some regions of constant

suction, which indicate the presence of a bubble, could be observed. The static pressure measurements shown in Figure 5.1a-c, indicate that a laminar separation bubble is present across the entire Reynolds number and ground clearance ranges tested at $\alpha = 0^\circ$. As discussed previously, the presence of a bubble is indicated by the static pressure remaining constant as the flow moves downstream in the pressure recovery region. It is difficult to locate accurately the laminar separation points from the present measurements because the spatial resolution, although better than Knowles et al. [62], is still too low. Even if it were of a higher resolution, however, attempting to read the separation point from a static pressure distribution based on the criteria set out by Russell [49] (Figure 2.6) is still fairly arbitrary. Ideally a high spatial resolution hot-film array would be used. Despite this, general trends of the size and location of the bubble can be distinguished from the present results.

As previously explained, a consequence of ground effect is the increase in suction due to constraining of flow. In order to satisfy the continuity equation the velocity doubles as the ground clearance halves, which leads to considerably larger pressure gradients. As the ground clearance becomes increasingly smaller, therefore, the downforce increase becomes ever greater. For example, moving from both $h/c = 0.9$ to $h/c = 0.3$, and $h/c = 0.3$ to $h/c = 0.1$ would appear to require velocity to triple in order to satisfy continuity. In practice, however, the continuity argument does not hold true as the movement in stagnation point will alter the pressure field such that mass flow under the wing will reduce. Moreover, the boundary-layer thickness will change as ground clearance alters due to the stronger pressure gradients that are produced. As such, whilst the flow velocity under the wing increases, it is not enough to maintain mass flow for the reducing area. Moreover, the massflow under three-dimensional wings will be impacted by vorticity at the wing tips. Whilst in practice mass flow may not be maintained and as such continuity

must be viewed as an ideal case, the basis of the argument is valid. Therefore, between $h/c = 0.3$ and $h/c = 0.1$ in particular, the pressure gradients are large enough to have a substantial effect on boundary-layer development. This will be discussed further in the following sections.

The Reynolds-number sensitivity of the static pressure measurements, and thus downforce, appears to diminish as the Reynolds number increases i.e. the difference between the two lowest tested Reynolds numbers is larger than the difference between the two highest tested Reynolds numbers. It could be argued, therefore, that at sufficiently high Reynolds numbers there would be no Reynolds-number sensitivity, and downforce would remain constant. However, as a laminar separation bubble is still a dominant aspect of the static pressure distribution, this may occur at a Reynolds number higher than could be tested in this work. As the ground clearance reduces, the Reynolds-number sensitivity increases, with the spread of suction values becoming increasingly larger. Whilst the effect of Reynolds number on the laminar separation point appears small, even negligible, the length of the bubble is clearly dependent on the Reynolds number. As Reynolds number increases the bubble becomes increasingly shorter. This is shown in Figure 5.1 by the smaller constant pressure region as Reynolds number increases. Thus, by combining a reduction in ground clearance with an increased Reynolds number, a bubble is produced that is both shorter and occurs further upstream.

Figure 5.1 also shows the effects of Reynolds number and ground clearance on the wing at $\alpha = 3^\circ$ (Figure 5.1d-f) and $\alpha = 5^\circ$ (Figure 5.1g-i). At $\alpha = 3^\circ$ the suction peak has moved upstream such that it occurs prior to $x/c = 0.1$, and thus the pressure tappings do not record its presence. The measurements suggest that a laminar separation bubble still forms at all tested Reynolds numbers for $\alpha = 3^\circ$, however at $Re = 5.99 \times 10^5$ it is considerably smaller. Additionally, the Reynolds-number sensitivity is all but eliminated, as shown

by the results of the two highest tested Reynolds numbers being almost coincident. In comparison to $\alpha = 0^\circ$, the separation bubble occurs further upstream for $\alpha = 3^\circ$. This is to be expected as the adverse pressure gradient begins further upstream. It is, however, worth noting that the boundary-layer flow was capable of travelling much further through the adverse pressure gradient before the laminar separation occurred, despite the gradient being stronger. Similarly to $\alpha = 0^\circ$, the Reynolds-number sensitivity appears to increase, and a larger separation bubble forms, as the ground clearance reduces. Hence, at $h/c = 0.1$ the separation bubble has a significant effect on the static pressure distribution.

Greater angles than $\alpha = 5^\circ$ were also tested, however due to the large loads that these cases generated the maximum Reynolds number that could be tested was limited. Results of these tests are given in Figure 5.2. As incidence is increased the maximum suction generated by the wing also increases, and trailing-edge separation begins to occur. As incidence is increased above $\alpha = 5^\circ$ it becomes difficult to infer the presence of a laminar separation bubble. Although the static pressure measurements indicate that there could be a small bubble, this cannot be conclusively confirmed. For the highest tested incidences, the measurements suggest a leading-edge laminar separation bubble is forming at $x/c \approx 0.1$. At $Re = 3.59 \times 10^5$ this occurs for both the $\alpha = 13^\circ$ and $\alpha = 15^\circ$ cases, whilst at $Re = 4.79 \times 10^5$ it can only be observed for the $\alpha = 13^\circ$ case. It is postulated that at $\alpha = 15^\circ$ the bubble became shorter due to the higher Reynolds number, and so was not detected as it moved upstream of $x/c = 0.1$.

Figure 5.3 shows the influence of ground clearance at varying incidence for $Re = 4.79 \times 10^5$. This suggests that at $h/c = 0.9$, because of the difference to the $h/c = 0.7$ case, the wing is still influenced by the presence of the ground. Hence, the out-of-ground-effect condition was not found. The main difference is clearly the large increase in suction at smaller ground clearances. Figure 5.3 also shows that, as the ground clearance reduces,

the separation bubble moves upstream.

At $\alpha = 5^\circ$ there is no clear separation bubble at the higher ground clearances, but there is a clear bubble for the lower ground clearance of $h/c = 0.1$. This highlights how the effect of the bubble is more pronounced when the wing is situated close to the ground. The most noticeable difference, however, is that at $h/c = 0.1$ and $\alpha > 0$ the wing has a very different pressure distribution characteristic to those at the other ground clearances. At $\alpha = 0^\circ$ the wing at $h/c = 0.1$ behaves similarly to the higher ground clearances, in that the suction peak occurs at $x/c = 0.42$ (the apex of the wing), however the magnitude of suction has significantly increased. At $\alpha = 3^\circ$ and $\alpha = 5^\circ$, however, whilst the higher ground clearances show only an adverse pressure gradient in the tested range, the $h/c = 0.1$ case also shows a favourable pressure gradient and suction peak. Hence, the suction peak occurs at a point further downstream than it does at higher ground clearances.

5.1.2 Surface-Flow Visualisation

The laminar separation bubble is the defining flow feature of the wing. The relation between the separation and reattachment points given by the flow visualisation and corresponding points on the static pressure distribution (generated by MSES - see Section 5.2 for further detail) is shown in Figure 5.4.

Figure 5.5 contains photographs of flow visualisation carried out on the suction surface of the wing at Reynolds numbers in the range $3.59 \times 10^5 \leq Re \leq 7.19 \times 10^5$. The low shear stresses inside the bubble resulted in the paint not fully drying, despite being run in some cases for almost one hour in the tunnel. Consequently, there are drip marks visible in the photographs due to the wing being rotated to take photographs. Although this is inconvenient, the key surface-flow characteristics can still be observed. Separation of the laminar

boundary layer occurs at $x/c \approx 0.62$ for all tested Reynolds numbers. The reattachment point is more difficult to pinpoint. This is partly because the streamlines are much less defined in the reattachment region at lower Reynolds number than at higher Reynolds numbers. It is postulated that this is a result of the boundary layer being more turbulent in the higher Reynolds number cases due to increased turbulence production in the separated shear layer. It could also be that the reattachment point is wavering, moving up and downstream such that a static separation point is not found even in the averaged-flow field that the paint provides. Although the reattachment point may not be clear, it still appears that the bubble becomes shorter as the Reynolds number is increased. This is observed by considering the point at which the streamlines become apparent (highlighted in blue). All Reynolds numbers show a build up of paint behind the separation point, indicating the dead-air region that forms here (see Figure 2.4). At lower Reynolds numbers there is then a large area before the streamlines reappear, whilst at higher Reynolds numbers the streamlines begin much closer to the build up of paint.

The effect of incidence is shown in Figure 5.6. As incidence is increased from $\alpha = 0^\circ$ to $\alpha = 3^\circ$ the laminar separation bubble, which is still clearly visible, moves upstream from $x/c \approx 0.62$ to $x/c \approx 0.57$ and becomes shorter in length. At $\alpha = 5^\circ$ there is no indication of a laminar separation bubble, however flow visualisation gives a good indication of the flow characteristics. The streaklines upstream of $x/c \approx 0.55$ clearly differ from those downstream of $x/c \approx 0.55$. The streaklines upstream of $x/c \approx 0.55$ are similar to those upstream of the laminar separation bubble of $\alpha \leq 3^\circ$, and thus are assumed to represent the laminar boundary-layer state. The streaklines downstream of $x/c \approx 0.55$ are taken to indicate the turbulent state, as they appear similar to the post-bubble streaklines of the $\alpha \leq 3^\circ$ cases. It can, therefore, be concluded that transition occurs in the vicinity of $x/c \approx 0.55$. The streaklines appear thicker at $x/c \approx 0.55$, which indicates a lower, but

not zero, shear stress. The location of this apparent transition agrees with the expected transition location, based on the movement of the separation bubble between the $\alpha = 0^\circ$ and $\alpha = 3^\circ$ cases.

The $\alpha = 5^\circ$ case exhibits trailing-edge separation, showing that the adverse pressure is too severe and flow can no longer overcome it at this Reynolds number ($Re = 4.79 \times 10^5$). This was not detected by the static pressure measurements as the separation point is downstream of the last pressure tap.

Flow visualisation at the maximum incidence of $\alpha = 15^\circ$ is given in Figure 5.7. An extended region of trailing-edge separation, beginning at $x/c \approx 0.52$, is the primary characteristic. A short leading-edge separation bubble can also be observed. The bubble begins at $x/c \approx 0.05$ and is only $x/c \approx 0.05$ in length. This highlights how far the transition point moves forward as incidence is increased.

5.1.3 Hot-Film Anemometry

As explained in Section 3.4.2, the hot-film gauges give only semi-quantitative information on the state of the boundary layer. The low spatial resolution means that they can be used to indicate the state of the boundary layer at a given point, rather than pinpointing specific locations of separation and reattachment. The quasi-shear stress is a good measure of skin-friction drag, however it can be difficult to decipher whether the boundary layer is laminar or turbulent as both could yield a similar value. Frequency analysis, in the form of a Power Spectral Density (PSD) plot is generated to show at which frequencies the energy in the boundary layer is contained. The frequency response of the hot-film is approximately 100kHz but, as discussed in Section 3.4.2, measurements were taken at 2 kHz with a 1 kHz anti-aliasing filter applied. The PSD can be misleading on its own as

a plot may appear energetic, but have a low shear stress value; hence it is important also to look at the mean shear stress. But equally, laminar and turbulent boundary layers can have the same shear stress value but their PSDs will be different.

An understanding of the boundary layer can be gained from the mean and variance of the quasi-shear stress at each position. The quasi-wall-shear stress (τ_Q) and coefficient of variation (CV) for the wing at various Reynolds numbers with $\alpha = 0^\circ$ are given in Figure 5.8 for $h/c = 0.3$. In the region $0.4 \leq x/c \leq 0.6$ the shear stress reduces as flow moves downstream; this represents the laminar boundary layer. The region of zero shear stress is where the boundary layer has separated from the surface. When the boundary layer then reattaches in a turbulent state the shear stress initially increases, before then reducing again further downstream. As the Reynolds number is increased the shear stress also increases in both the laminar and turbulent regions.

Had the spatial resolution of the hot-film sensor array been higher, the results would have been capable of providing accurate positions of separation and reattachment. These positions can, however, still be seen within the spatial resolution limits of the sensor array. From Figure 5.8 it can be observed that as the Reynolds number increases the reattachment point moves upstream.

The boundary-layer state can be determined from the CV. In the laminar boundary layer the flow is smooth and the shear stress has low variance. In contrast, the turbulent boundary layer comprises eddies of differing length and time scales that result in high variance of the shear stress signal. The location of transition, therefore, is indicated by the point where CV rises sharply. Figure 5.8 shows that transition has occurred at $x/c \approx 0.7$ for the three highest tested Reynolds numbers, and $x/c \approx 0.75$ for the lower Reynolds number cases.

Figure 5.9 shows τ_Q and CV for three incidences (0° , 3° , and 5°) at $Re = 4.79 \times 10^5$. As incidence is increased from $\alpha = 0^\circ$ to $\alpha = 3^\circ$ the bubble shortens in length, as demonstrated by τ_Q being zero for a single point rather than two. At $\alpha = 5^\circ$, τ_Q never becomes zero, which indicates that no separation bubble is formed. This supports the flow visualisation conclusion that transition occurs in the attached boundary layer. In the laminar boundary layer, $x/c \leq 0.5$, the greater the incidence of the wing, the τ_Q . After transition, however, maximum τ_Q is much higher for the higher incidence cases initially, but as the flow moves downstream it reduces at a much greater rate. This leads to $\alpha = 5^\circ$ having the lowest shear stress at the last gauge. Figure 5.9b shows that transition has moved upstream, with a turbulent boundary layer first recorded at $x/c = 0.7$ for the $\alpha = 0^\circ$ case, at $x/c = 0.65$ for the $\alpha = 3^\circ$ case, and $x/c = 0.55$ for the $\alpha = 5^\circ$ case.

The effect of ground clearance is shown by comparing Figures 5.8 & 5.9 with Figures 5.10 & 5.11, which show various Reynolds numbers (Figure 5.8) and incidences (Figure 5.9), but at the lowest ground clearance of $h/c = 0.1$. The influence of Reynolds number at the lowest ground clearance is similar to that at $h/c = 0.3$. The transition point moves upstream, as shown by the point where CV increases, and τ_Q upon reattachment being greater as the Reynolds number is increased. In comparison to the higher ground clearance of $h/c = 0.3$ the magnitude of τ_Q is greater for a given incidence and Reynolds number at the lower ground clearance. For $\alpha = 0^\circ$ at $h/c = 0.3$ (Figure 5.8), transition occurs at $x/c \approx 0.75$ for the Reynolds number range $2.88 \times 10^5 \leq Re \leq 4.79 \times 10^5$. For the same Reynolds number range at $h/c = 0.1$ transition occurs at $x/c \approx 0.65$ for the two lower Reynolds numbers, and $x/c = 0.6$ for $Re = 4.79 \times 10^5$. It appears that the wing at $h/c = 0.1$ has higher Reynolds-number sensitivity than at $h/c = 0.3$, particularly in the laminar boundary layer ($x/c \leq 0.55$). The effect of incidence at $h/c = 0.1$ is shown in Figure 5.11. It shows similar trends to Figure 5.9 in that the transition point moves

upstream as incidence is increased.

Figure 5.12 shows a PSD of the output of each hot-film gauge, at various Reynolds numbers. As explained in Section 3.4.2, the spectra were all measured at the same time using individual amplifiers multiplexed to a single output. Figure 5.8b shows that the boundary layer will be laminar at $x/c = 0.45$ and turbulent at $x/c = 0.85$. Therefore, the difference between the laminar and turbulent boundary layer PSDs can be seen in Figure 5.12. In general, the energy in a laminar boundary layer is contained primarily in the low frequencies. For the lowest tested Reynolds number this is below 50 Hz. The turbulent boundary layer, on the other hand, contains energy which is more evenly distributed across the measured frequency spectrum. Observing the characteristics of the different tested Reynolds numbers at $x/c = 0.45$ and $x/c = 0.85$, it can be seen that the laminar and turbulent boundary-layer states are completely opposite in terms of Reynolds-number sensitivity. At $x/c = 0.45$, the $Re = 2.88 \times 10^5$ case contains very little energy and it is all contained in the low frequencies (below 50 Hz). In contrast, the $Re = 7.19 \times 10^5$ case contains significantly more energy and across a much broader frequency range. It is not the signal of a turbulent boundary layer at $x/c = 0.45$, as $x/c = 0.85$ shows that of a fully-turbulent boundary layer. However, the effect of increased ratio of inertial to viscous forces of the higher Reynolds number flow on the characteristics of the laminar boundary layer are clear. The PSDs of $Re = 2.88 \times 10^5$ and $Re = 7.19 \times 10^5$ at $x/c = 0.85$ appear very similar and the turbulent boundary layer that exists at $x/c = 0.85$ exhibits little Reynolds number sensitivity. This suggests that it is the laminar boundary-layer state that causes Reynolds-number sensitivity.

At $x/c = 0.7$, and to a lesser extent $x/c = 0.65$, a number of spikes are observed in the PSD at the three highest tested Reynolds numbers. For the lower Reynolds numbers these spikes, though much less prominent, occur at $x/c = 0.75$. The accelerometer tests, which

were described in Section 3.3.1, showed that these spikes are not caused by vibration and do not correspond to harmonics of the wind-tunnel-fan frequency. The spikes must, therefore, be physical within the flow. From Figure 5.8 it is known that at $0.6 \leq x/c \leq 0.7$ for $Re \leq 5.99 \times 10^5$ the friction coefficient is zero, indicating that a separation bubble is present at that location, and at $x/c = 0.75$ the boundary layer has reattached to the surface. It is postulated that the frequency spikes are transitional behaviour in which turbulence is being produced. It is proposed that the spikes are initial growth of disturbances followed by the generation and growth of harmonics and sub-harmonics of a fundamental frequency. Based on the observations of Dovgal et al. [120] this would be indicative of non-linear interactions between disturbances. Moreover, as the spikes are also observed at $\alpha = 5^\circ$ (Figure 5.14), for which the boundary layer has been shown to remain attached at all times, it demonstrates that these spikes are not the unsteady movement of the reattachment point, but instead a result of turbulence production in both the attached and detached shear layers.

The spikes are drastically reduced for the lower Reynolds number cases, where turbulence production would be reduced due to the velocity shear, which causes the Kelvin-Helmholtz instabilities, being smaller. Additionally, the frequency spikes correspond to the point where the boundary layer reattaches. For $Re \geq 4.79 \times 10^5$ the rise in τ_Q is at $x/c = 0.7$, the same point at which the frequency spikes are most prominent. Whereas at $Re \leq 3.59 \times 10^5$, the sharp increase in friction coefficient and the most prominent frequency spikes both occur at $x/c = 0.75$.

At $x/c = 0.7$ the frequency spikes occur at regular intervals, and could therefore be harmonics of a single fundamental frequency. This is in accordance with the findings of Hain et al. [14], who found that a single vortex-shedding frequency was present in Kelvin-Helmholtz instabilities above the separation bubble. Hot-wire measurements around a

NACA-0012 aerofoil at $\alpha = 3^\circ$ by Kim et al. [113] showed that the Tollmien-Schlichting wave frequency increased from 32 Hz to 88 Hz as the Reynolds number increased from $Re = 2.3 \times 10^4$ to $Re = 4.8 \times 10^4$. Similar phenomena are observed here, where the frequency spike rises from 65Hz at $Re = 3.59 \times 10^5$ to 99 Hz at $Re = 7.19 \times 10^5$. The development of this frequency can be seen in PSDs of upstream gauges. It is, however, clearest at $Re = 7.19 \times 10^5$ because of the higher energy it contains. As the flow moves downstream the amplitude of the spikes relative to the rest of the flow becomes increasingly greater, showing that the frequency is being amplified. This is to be expected as the adverse pressure gradient reduces the stability of the boundary layer causing amplification of Tollmien-Schlichting waves. Although similar phenomena can be observed for the lower Reynolds number cases, it is not noticeable at the most upstream gauges as the instability promoted by the adverse pressure gradient is not as great, hence there are fewer perturbations in the flow and the energy is still contained in the lowest frequencies.

Figure 5.8 shows that the shear layer is separated from the surface at $0.65 \leq x/c \leq 0.7$ for $Re = 5.99 \times 10^5$, and at $x/c = 0.65$ for $Re = 7.19 \times 10^5$. Given that frequency spikes can be observed in the laminar boundary layer prior to separation, then the models of natural transition (Figure 2.1) and a separation bubble (Figure 2.4) can be combined. The frequency spikes suggest that at higher Reynolds numbers the laminar boundary layer already contains instabilities that are promoting turbulence. Thus, when the laminar boundary layer separates, the turbulence production is greater as the existing instabilities are amplified. This is why transition sometimes occurs in the attached boundary layer, rather than through the separation-induced transition mechanism. When the instabilities in the laminar boundary layer are amplified sufficiently, there is enough turbulence to keep the boundary layer attached. Hence, no laminar separation occurs. In this case the boundary layer is no longer laminar, but is transitional prior to the usual laminar separation

point. Figure 5.13 shows schematics of the three instances observed in this work. The first is the same model presented in Figure 2.4, in which the laminar boundary layer separates, turbulence is generated in the separated shear layer, which reattaches as a turbulent boundary layer. The second bubble is shorter, this is because the laminar boundary layer contains some perturbations that are being amplified by the instability caused by the adverse pressure gradient. The perturbations are not, however, large enough to give the same wall-normal momentum transfer that makes a fully-turbulent boundary layer more resilient to adverse pressure gradients than laminar boundary layers, and so it still separates. In the separated shear layer the existing perturbations are amplified and sufficient turbulence to cause reattachment is produced further upstream than in the first case. In the third model, no separation bubble occurs and instead transition occurs in the attached boundary layer.

In Figure 5.12, the energy content at $x/c = 0.7$ for $Re = 7.19 \times 10^5$ appears larger than that of any other location, indicating that turbulence production is at a maximum at this location. This is unsurprising given that the output-voltage variance, and thus CV, is the integral of the PSD. Given that this is where the friction coefficient began to rise and CV is at a maximum, it is reasonable to conclude that turbulence production and energy content will be highest at this point. The boundary layer will then lose energy once it fully reattaches and must overcome the adverse pressure gradient once again. For all cases the frequency spikes still exist after τ_Q has become non-zero, showing that the boundary layer is not fully turbulent upon reattachment. Instead, the boundary layer becomes fully turbulent further downstream. Zhang et al. [16] came to a similar conclusion based on their hot-film measurements on a wing out of ground effect.

Figure 5.14 shows the PSD of hot-film outputs at $Re = 5.99 \times 10^5$ for incidences $\alpha = 0^\circ$, $\alpha = 3^\circ$ and $\alpha = 5^\circ$. The frequency spikes are again visible and occur at more upstream

locations as incidence is increased. This again leads to the conclusion that they must be an indication of transition as it has been shown that the location of transition moves upstream as incidence is increased. The three most downstream gauges all exhibit PSDs characteristic of the fully-turbulent-boundary-layer state. As discussed previously, at $\alpha = 0^\circ$ the boundary layer is laminar until $x/c = 0.65$, transition occurs at $0.65 \leq x/c \leq 0.7$, and the flow is turbulent for $x/c \geq 0.8$. Clearly the low spatial resolution of the hot-film gauges means that the exact positions of these three states cannot be determined. Even if the resolution were higher, however, it would still be difficult to determine where the boundary-layer becomes fully turbulent. The point of the fully-turbulent state could be seen as the point at which all frequency spikes have vanished, however this will still be a fairly arbitrary exercise. At the resolution of the present tests, it is easier to state once a fully-turbulent boundary layer is present at a gauge. For the $\alpha = 3^\circ$ case, a laminar boundary layer is evident up to $x/c = 0.6$, transition occurs in the region $0.65 \leq x/c \leq 0.7$, and a fully-turbulent boundary layer is evident from $0.75 \leq x/c \leq 0.85$. At $\alpha = 5^\circ$ all of the features occur further upstream. A laminar boundary layer is apparent at the $x/c = 0.45$ gauge before transition occurs in the region $0.5 \leq x/c \leq 0.6$, and a fully-turbulent boundary layer then observed at all remaining gauges.

The PSDs given in Figure 5.15 for the hot-film gauges at the lowest tested ground clearance, $h/c = 0.1$, are compared to $h/c = 0.3$ for varying Reynolds number. Two single frequency spikes are observed at 250 Hz and 300 Hz for the first four gauges which were not removed when the zeroes (wind-off values) were subtracted. This is not evidence of a physical flow feature, but rather signal or interference. In comparison to the $h/c = 0.3$ case, it is clear that the turbulent boundary layer occurs further upstream for the lower ground clearance. At $x/c = 0.65$ the results for $h/c = 0.1$ demonstrates characteristics of a turbulent boundary layer. The earlier transition point for $h/c = 0.1$ leads to the upstream

gauges ($x/c \leq 0.75$) showing greatly dissimilar results to that of $h/c = 0.3$. At the last two gauges ($x/c \geq 0.8$), however, the results at all Reynolds numbers and ground clearances are very similar. Although the distribution of energy throughout the frequency range is the same for these two locations, it is shown in Figure 5.11 that the shear stress is different and so the boundary layer is not the same. It does, however, indicate that similarity between cases may improve if a fully-turbulent boundary layer were present across the entire chord. This is investigated later.

No frequency spikes are evident at $h/c = 0.1$. At this ground clearance the local Reynolds number, at a given location, is higher because of the higher flow velocity caused by the closer proximity to the ground. This leads to the boundary layer becoming turbulent further upstream and, arguably, higher turbulence production. The latter is due to the higher velocity gradient that occurs at the edge of the boundary layer and the stronger pressure gradients that the boundary layer is subjected to. It could be, therefore, that the spikes are still occurring, but the low spatial resolution of the hot-film sensors means they are not being recorded.

At $h/c = 0.1$ for the gauges located at $x/c \leq 0.65$, prior to the formation of the fully-turbulent boundary layer, there is a greater difference between the signals of each Reynolds number than for $h/c = 0.3$. This indicates that as the ground clearance reduced the boundary layers Reynolds-number sensitivity increased.

5.1.4 LDA Velocity Profiles

Wake surveys of the x-component of velocity downstream of the wing at various Reynolds numbers for $\alpha = 0^\circ$ are shown for three locations: $x/c = 1.5$, $x/c = 2$, and $x/c = 3$, in Figure 5.16. As the fluid travels downstream, small-scale turbulence causes the wake

to become thicker and the velocity deficit to reduce. The upwash behind the wing, a consequence of the downforce being produced, leads to the wake moving upwards as it moves downstream. As the Reynolds number increases the wake becomes thinner and the velocity deficit is reduced. This is evident at all three locations.

The magnitude and location of the velocity deficit at various Reynolds numbers for each wake survey is given in Table 5.1. It can be seen that at all Reynolds numbers the velocity deficit reduces with downstream position. Although the velocity deficit is greatest for the lowest Reynolds number case, the recovery back towards freestream ($V_x/V_\infty = 1$) is greater. At $Re = 3.59 \times 10^5$ the velocity deficit recovers by $\Delta V_x/V_\infty = 0.069$ across the three given locations, whilst the $Re = 4.79 \times 10^5$ and $Re = 5.99 \times 10^5$ cases recover by $\Delta V_x/V_\infty = 0.066$ and $\Delta V_x/V_\infty = 0.063$ respectively.

Wake surveys for $\alpha = 3^\circ$ are given in Figure 5.17. As the flow moves downstream the velocity deficit reduces, the wake thickness increases, and the wake moves away from the ground. In comparison to $\alpha = 0^\circ$, the wake is higher due to the larger downforce the wing at incidence produces and the consequent increase in upwash.

The Reynolds-number sensitivity of the velocity deficit is reduced for $\alpha = 3^\circ$. In contrast to $\alpha = 0^\circ$, at $x/c = 1.5$ and $x/c = 2$ the higher the Reynolds number, the greater the velocity deficit. At $x/c = 3$, however, the velocity deficit is smaller for the higher Reynolds numbers. This shows that there is higher turbulence dissipation at higher Reynolds num-

Table 5.1: Magnitude and location of velocity deficit at varying Reynolds number and location behind wing

$\alpha = 0^\circ$	Min. V_x/V_∞			Location (y/c)		
	$x/c = 1.5$	$x/c = 2.0$	$x/c = 3.0$	$x/c = 1.5$	$x/c = 2.0$	$x/c = 3.0$
$Re = 3.59 \times 10^5$	0.857	0.901	0.926	0.414	0.416	0.451
$Re = 4.79 \times 10^5$	0.863	0.904	0.929	0.415	0.421	0.451
$Re = 5.99 \times 10^5$	0.873	0.911	0.936	0.416	0.425	0.451

bers for $\alpha = 3^\circ$. At $x/c = 3$ for $\alpha \geq 3^\circ$ the top of the wake profile is at a similar height for all Reynolds numbers, and the point of minimum velocity occurs further away from the ground. Similar trends can be observed in Figure 5.18 for $\alpha = 5^\circ$. However, the difference between the various Reynolds number cases has decreased, again showing a reduced Reynolds-number sensitivity. Overall, there is a trend of reducing Reynolds-number sensitivity with increasing incidence.

The velocity profiles underneath the wing on a line perpendicular to the chord for various Reynolds numbers for $\alpha = 0^\circ$ is shown in Figure 5.19. The profile is taken at $x/c = 0.375$; this is in the favourable pressure gradient but close to the point of maximum suction ($x/c = 0.42$). It can be seen that the velocity underneath the wing increases as the Reynolds number is increased, resulting in a lower static pressure as shown in Figure 5.1. Measurements were made to within $y/c = 0.085$ of the wing's lower surface, although this was not close enough to capture the entire boundary layer. This is due to the profile being taken at the end of the favourable pressure gradient where the laminar boundary layer is relatively thin. It does, however, show that maximum velocity, and therefore maximum suction, occurs close to the wing, as the velocity reduces further away from the wing. It also shows that the flow velocity has increased for the entire measured area underneath the wing, and not just close to the surface. Unfortunately, measurements could not be taken lower than $y/c = 0.15$ because the endplate blocked the laser. It is assumed, however, that the velocity will be increased in the entire area under the wing and that the boundary layer that forms on the ground due to the higher-than-freestream-velocity flow will be thinner.

5.2 Euler Simulations - MSES

In this section, results of the two-dimensional Euler code, MSES, are used to show boundary-layer characteristics of the wing to supplement the experimental results presented in the previous section. The Reynolds-number range is extended to higher values than could be achieved in the wind-tunnel experiments and the influence of incidence, ground proximity and freestream turbulence are observed. Finally, the dependence on geometry of the results is investigated by comparing the Reynolds-number sensitivity of the GA(W)-1 profile to that of the Tyrrell-026 profile, which was previously tested by Zerihaan & Zhang [69].

5.2.1 Influence of Reynolds number at Zero Incidence

From the experimental pressure distributions it was inferred that downforce coefficient increased as Reynolds number increased due to the increase in suction that was observed. Numerical simulation allows the downforce and drag coefficients for the profile to be found and their variation with Reynolds number to be determined. Figure 5.20a shows that as the Reynolds number is increased the downforce coefficient increases, drag coefficient decreases, and the point of transition moves upstream.

Pressure coefficient, friction coefficient, and boundary-layer displacement thickness distributions of the wing at $\alpha = 0^\circ$ are shown in Figure 5.21a-c. The first graph shows the static pressure distribution for the suction surface. The magnitude of suction increases across the chord as the Reynolds number increases, hence the increase in downforce observed in Figure 5.20a. As described in Section 2.1.3, the region of constant pressure in the distribution shows the presence of a laminar separation bubble. The bubble occurs at all Reynolds numbers, however the different suction values in the constant pressure region

have a direct influence on downforce generation. After the region of constant pressure, the suction value decreases as fluid is decelerated at a greater rate due to the curvature introduced by the bubble.

Figure 5.22a shows that the trends observed in Figure 5.21a are similar to the experimental pressure distributions shown in Figure 5.1. In particular, the regions of constant pressure crossing over one another and the suction increase for increasing Reynolds number. MSES, however, predicts a much higher suction value and a smaller separation bubble. The smaller separation bubble is a result of the greater suction value as it has been shown that when suction is increased (for higher Reynolds numbers) the bubble's size is reduced. Panel method simulations by Knowles et al. [62] also gave a significant over-prediction of suction in comparison to their wind-tunnel pressure distributions. There are a number of aspects that could contribute to this over-prediction. Firstly, the boundary layer shape predicted by the code may differ from that of the wind tunnel, due to aspects such as freestream turbulence intensity, and surface temperature and roughness of the wing. Factors such as the boundary-layer shape factor and separation bubble length and height can significantly impact the drag and lift coefficients of the wing. It is possible that the problem is due to the lack of turbulence modelling as the code works simply by using a turbulent-boundary-layer thickness (where applicable) to displace inviscid streamlines. Consequently, momentum losses that occur in turbulent flow due to the fluctuating-velocity components are not included. Additionally, MSES makes use of the mirror-image technique to simulate the moving ground. This means that no residual boundary layer forms on the 'ground' between the wing and its mirror image. This is because the 'ground' in this case does not remain at freestream velocity, but is accelerated and decelerated by the two wings. Although this effect might be small, it will also contribute to the higher suction values produced by MSES. Regardless of this shortcoming,

however, as the trends given by MSES agree with the trends observed in the experiments, the correlation can be deemed satisfactory.

Figure 5.21b shows the displacement thickness of the suction-surface boundary layer. It is the displacement thickness that the code uses to determine the effective section shape of the wing, and thus it has a direct influence on the force generated by the wing. For a given chordwise location the displacement thickness reduces as the Reynolds number increases.

The lower Reynolds numbers produce separation bubbles that are both longer and thicker than those at higher Reynolds numbers. This adds significantly more curvature to the effective aerofoil shape. It might be inferred that more suction would be generated by the wing at lower Reynolds numbers, because the effective wing's camber has been increased; but, in fact, the suction does not increase. The integral of the pressure coefficient distribution in the area bounded by the bubble (as detailed in Figure 5.23) is shown in Table 5.2. This shows that as Reynolds number is increased, force enhancement due to the bubble is reduced. The increase in downforce for increasing Reynolds number observed in Figure 5.20a cannot, therefore, be a result of the separation bubble. Instead, it can be attributed to the thicker boundary layer that forms at the trailing edge for the lower Reynolds number cases altering the effective shape of the wing such that the expansion ratio in the pressure recovery region (the area between the effective surface of the wing and the ground) is reduced. This causes the diffuser-effect of the wing to be diminished, and suction reduced as a result, as the Reynolds number is decreased.

The friction coefficient (C_F) distribution on the suction-surface of the wing, given in Figure 5.21c, shows an initial peak close to the stagnation point, after which it decreases as the fluid moves downstream in the laminar boundary layer. The region of zero skin

friction denotes the position of the separation bubble, after which reattachment of the turbulent boundary layer cause a rise in friction. It should be noted that, unlike in real flows where the boundary layer can be transitional upon reattachment, the MSES code only allows the boundary layer to be either laminar or turbulent.

There is a very small difference in the location of separation points for each Reynolds number, despite the increase in Reynolds number leading to lower shear stresses in the laminar boundary layer. The higher Reynolds numbers lead to the laminar separation point moving downstream slightly, a result of the boundary-layer flow having more energy at higher Reynolds number such that it is more capable of overcoming the adverse pressure gradient. The reattachment location is much more sensitive to Reynolds number, where the reattachment point moves upstream as Reynolds number increases. At higher Reynolds numbers the inertial forces become more dominant and instabilities are amplified at a greater rate, increasing turbulence production. Additionally, turbulent motions in a separation bubble are generated primarily through Kelvin-Helmholtz type instabilities, which originate from the velocity-shear of the dead-air region and outer flow - so higher Mach number flow, akin to a higher Reynolds number, causes greater production of these instabilities. This higher turbulence production is evident by the greater value of friction upon reattachment.

Table 5.2: Integral of static pressure distribution where $C_F = 0$ i.e. inside separation bubble (MSES, $h/c = 0.3125$)

Re ($\times 10^{-5}$)	$\int C_P$
2.85	-0.1469
3.56	-0.1361
5.94	-0.1044
8.32	-0.0819
10.69	-0.0677
11.88	-0.0632

5.2.2 Effect of Incidence

In addition to wing and boundary-layer characteristics at zero incidence, Figures 5.20 & 5.21 also show characteristics at $\alpha = 3^\circ$ and $\alpha = 5^\circ$. Similarly to the previous section, the static pressure distributions calculated for $\alpha = 3^\circ$ and $\alpha = 5^\circ$ show a similar shape and trend to the experimental results, but the values of suction are lower (Figures 5.21d & 5.21g). As incidence is increased the point of maximum suction moves upstream and the maximum suction value increases. The Reynolds-number sensitivities of the wing's downforce and drag coefficients reduce as incidence is increased, as shown in Figure 5.20, due to the similar pressure distributions. In contrast, as incidence is increased the variation of transition location with Reynolds number becomes larger.

The displacement thickness, given in Figure 5.21, shows that the effective aerofoil shape becomes less Reynolds-number dependent as incidence is increased, both in terms of the camber increase at the bubble's location and the boundary-layer thickness at the trailing edge. This explains why the pressure coefficient distributions, and by extension the aerodynamic force coefficients, are similar for all tested Reynolds numbers i.e. Reynolds-number sensitivity is reduced as incidence is increased. For a given incidence the bubble moves upstream as Reynolds number increases. Hence the bubble appears closest to the leading edge for $\alpha = 5^\circ$ at $Re = 11.88 \times 10^5$ and closest to the trailing edge for $\alpha = 0^\circ$ at $Re = 2.85 \times 10^5$. These two conditions also represent the smallest and largest bubbles respectively, both in terms of length and height. The displacement thickness is largest for $\alpha = 5^\circ$, which is a result of the stronger adverse pressure gradient. For $\alpha = 5^\circ$ the displacement thickness at the trailing edge shows an opposing trend to the other incidences, in that the higher the Reynolds number the thicker the boundary layer. It appears that it is the earlier transition to a turbulent state of the higher Reynolds number cases that leads to trailing-edge separation, as shown by the zero friction coefficient. It is the

higher momentum loss in the turbulent boundary layer, which begins further upstream at higher Reynolds numbers that leads to this separation. The effect of laminar and turbulent boundary-layer extent is discussed further in Chapter 7.

Table 5.3: Integral of friction coefficient distribution ($h/c = 0.3$, $\alpha = 0^\circ$)

$Re (\times 10^5)$	$\int C_F$
2.85	4.19
3.56	4.04
5.94	3.61
8.32	3.32
10.69	3.012
11.88	3.04

The friction coefficient for $\alpha = 3^\circ$ shows that the two lowest Reynolds number cases still exhibit a laminar separation bubble, as shown by the friction coefficient going to zero. At $Re \geq 5.94 \times 10^5$ no zero-friction region is observed, hence no separation bubble is present and transition occurs in the attached boundary layer. In MSES, this occurs because the amplification of instabilities has reached e^7 before the laminar separation point. At $\alpha = 5^\circ$ no separation bubble is observed at any Reynolds number but, as with other incidences, as Reynolds number is increased the transition point moves upstream. Unlike the other cases, however, the transition point is not shown by reattachment behind a separation bubble, but by the rise in shear stress.

Figure 5.20 shows that the drag coefficient is reduced as Reynolds number is increased. However, the friction distribution suggests that the higher the Reynolds number, the higher the skin friction drag, as the integral of the friction distribution is be larger. This is shown in Table 5.3. This means that the higher Reynolds number case must have a reduced pressure drag component such that the total drag of the wing is lower.

5.2.3 Effect of Ground Proximity

Figure 5.24 shows wing and boundary-layer characteristics for $0.15 \leq h/c \leq 1.0$ and an out-of-ground effect case. The smallest ground clearance modelled was $h/c = 0.15$ as a converged solution could not be obtained at $h/c = 0.1$. Comparison of the out-of-ground-effect to $h/c = 1$ shows that even at relatively high ground clearances, the wing is influenced by the presence of the ground. As the wing moves closer to the ground, significantly higher suction values are produced as flow is constrained between the wing and ground. The separation bubble can be observed at all tested ground clearances.

The displacement thickness shows that the separation bubble moves upstream and becomes increasingly shorter as the ground clearance is decreased. There is a small difference in the laminar boundary-layer thickness in that the closer the wing is to the ground, the thinner the boundary layer. This is a result of the higher flow velocity of the lower ground clearance. The difference in the laminar boundary-layer thickness, however, is relatively small in comparison to the difference in the turbulent boundary-layer thickness at the trailing edge. Although the boundary layer is thinner in the favourable pressure gradient for the lower ground clearance case, at the trailing edge it is dramatically thicker. So the higher flow velocity of the lower ground clearance case causes the laminar boundary layer to be thinner, however upon transition the stronger adverse pressure gradient and longer distance the turbulent boundary layer must travel between the reattachment point and trailing edge leads to a thicker boundary layer at the trailing edge. The thicker boundary layer is result of the much stronger adverse pressure gradient that the boundary layer is subjected to at lower ground clearances. It was suggested by Zerihaan & Zhang [69] that part of the force-reduction phenomenon at lower ground clearance is that a thicker boundary layer will encounter the adverse pressure gradient. The results given here, however, suggest that a wing closer to the ground will have a thinner boundary layer at the start

of the adverse pressure gradient. The thinner boundary layer at the start of the adverse pressure gradient would also appear to be the more sensible conclusion, as when the wing moves closer to the ground the flow velocity increases as it is constrained between the wing and the ground. For a given distance from the leading edge, the wing closer to the ground must have higher velocity flow over the surface, and as such a thinner boundary layer would be expected.

As the ground clearance reduces the friction coefficient in the laminar boundary layer increases and the separation point moves upstream due to the adverse pressure gradient becoming increasingly stronger. Additionally, the bubble becomes shorter in length. At the reattachment point the friction coefficient is greater for the lower ground clearances and occurs at a more upstream location. No trailing-edge separation is observed at any of the tested Reynolds numbers or ground clearances. However, given that the lowest ground clearance case has the lowest friction coefficient at the trailing-edge it can be inferred that this case is the closest to separation.

5.2.4 Effect of Freestream Turbulence

The effect of freestream turbulence was investigated by altering the transition criterion at $Re = 5.94 \times 10^5$, $\alpha = 0^\circ$ and $h/c = 0.3$. The freestream turbulence affects transition as the perturbations in the freestream flow are transmitted to the boundary layer. In the MSES code, freestream turbulence is altered by changing the critical amplification factor (N), the premise being that with higher freestream turbulence more perturbations would be transmitted to the boundary layer such that transition will occur sooner. A lower e^N value simulates this as perturbations would require less amplification inside the boundary layer. Values of N between 3 and 9 were trialled, the boundary-layer characteristics of which are given in Figure 5.25. As the freestream turbulence is increased, by the critical

amplification factor being reduced from 9 to 3, the downforce and drag coefficients reduce by 1.4% and 8.6% respectively. This is solely a consequence of the upstream movement of the transition point by up to 9.5%.

The pressure distributions for all cases align very closely, except in the region close to the separation bubble. It is clear that the lower freestream turbulence case gives the highest downforce because the integral of the static pressure distribution for this case ($N = 9$) is the largest. This is solely a consequence of the longer separation bubble giving a larger region of constant pressure. The separation bubble is both longer and taller at $N = 9$, as shown by the displacement thickness in 5.25b. The displacement thickness also shows that it is only the formation of the separation bubble that differs between the different freestream turbulence cases. The negligible difference in the boundary-layer thickness at the trailing edge means that all cases have equal diffuser effect, and thus the difference in downforce coefficient is solely a result of the size of the separation bubble and the curvature that it introduces to the wing's effective shape.

The friction coefficient distribution shows negligible difference in the favourable pressure distribution ($x/c < 0.42$). As the freestream turbulence is reduced (e^N increasing) the laminar separation point moves downstream; the exception is $N = 3$, for which transition occurs in the attached boundary layer. This is similar to that observed in the experimental work for $\alpha = 5^\circ$, where the boundary layer becomes so unstable from the stronger adverse pressure gradient that instabilities are amplified to the point of transition before laminar separation occurs. In this case, the boundary layer instability is a result of the higher freestream turbulence transmitting perturbations into the boundary layer. For the cases in which a separation bubble is formed, $e^N \geq 5$, the reattachment points occurs further downstream as freestream turbulence is reduced, hence a larger bubble is observed.

5.2.5 Geometry Dependency

The GA(W)-1 is an aft-loaded profile, where maximum suction occurs at $x/c \approx 0.42$, and as a result tends to promote laminar flow because of the long favourable pressure gradient. To observe whether laminar phenomena are still prevalent in more fore-loaded aerofoils, the Tyrrell-026 profile was also tested in MSES.

Boundary-layer characteristics on the suction surface of the Tyrrell-026 wing are shown in Figure 5.26. The static pressure distribution shows that the magnitude of suction is sensitive to the Reynolds number, just as the GA(W)-1 was. As the Reynolds number increases the magnitude of suction initially increases, however only up to a point, after which it then decreases again. The two highest tested Reynolds numbers then exhibit identical pressure distributions, thus leading to the conclusion that at $Re \geq 10.69 \times 10^5$ the wing is insensitive to Reynolds number. The static pressure distribution for the Tyrrell-026 aerofoil is significantly different to that of the GA(W)-1 as it contains two suction peaks, both of which are considerably higher ($C_p \approx -2.5$) than that of the GA(W)-1 ($C_p \approx -1.5$). These two suction peaks are a characteristic of the geometry that was also observed by Doig et al. [93] at a number of ground clearances. As the Reynolds number is varied the pressure distribution changes principally in the region close to the second suction peak ($0.05 \leq x/c \leq 0.5$). Moreover, based on the region of zero friction coefficient, a laminar separation bubble is formed for the lower Reynolds numbers.

From the friction distribution it can be observed that at the lower tested Reynolds numbers the boundary layer separates and then reattaches on two separate occasions, due to the two separate adverse pressure gradients that occur. Whilst reattachment of a separated shear layer is typically associated with the generation of a turbulent boundary layer, the boundary layer must be laminar during the second adverse pressure gradient in order for a bubble to be formed there also. The sequence of events, therefore, is that the boundary

layer is accelerated to the first suction peak, a small laminar separation bubble is then formed in the first adverse pressure gradient. Reattachment of the still-laminar boundary layer then occurs due to the flow again being accelerated. The laminar boundary layer then separates again in the second adverse pressure gradient before reattaching as a turbulent boundary layer. MSES provides the exact transition point (where instabilities have been amplified to the chosen critical amplification factor, N) for each surface. This shows that transition occurs at $x/c \approx 0.28$, which indicates that transition occurs in the second laminar separation bubble. Thus showing that the boundary layer remained laminar in the first bubble, and is turbulent downstream of the second bubble. Whilst the first small bubble is formed for all tested Reynolds numbers, its effect on the global flow-field is limited. The second bubble is formed only for $Re \leq 5.94 \times 10^5$. At $Re = 8.32 \times 10^5$ a similar trend to the GA(W)-1 at $\alpha = 5^\circ$ is observed in that transition occurs in the attached boundary layer, as shown by the rise in friction coefficient without it becoming zero. For $Re \leq 8.32 \times 10^5$ the displacement thickness shows how the effective shape the wing is altered by the presence of the bubble, with the effect reducing as the Reynolds number is increased.

For $Re \geq 10.69 \times 10^5$ the pressure distribution does not show the region of constant pressure, nor the bump in the displacement thickness, that characterise the separation bubble in the second adverse pressure gradient. Moreover, the friction distribution does not show the sharp fall towards zero, as the other cases do, where the second bubble forms. It can be confirmed from these two aspects, and the fact that MSES denotes the transition point as being $x/c = 0.035$, that transition has occurred in the first laminar separation bubble, and thus a fully-turbulent boundary layer is formed ahead of the second suction peak. It is because of this that some of the trends observed due to Reynolds-number sensitivity for the GA(W)-1 only hold true for the Tyrrell-026 profile at $Re \leq 8.32 \times 10^5$. For exam-

ple, the displacement thickness at the trailing edge becomes increasingly smaller as the Reynolds number is increased, however for $Re \geq 10.69 \times 10^5$ it becomes thicker. It can be observed that the boundary layer displacement thickness is the smallest of all tested cases for $Re \geq 10.69 \times 10^5$ after the first bubble, in the region $0 \leq x/c \leq 0.2$, but after the second separation of the lower Reynolds number cases the boundary layer for $Re \geq 10.69 \times 10^5$ is thicker than the $Re = 5.94 \times 10^5$ and $Re = 8.32 \times 10^5$ cases.

Between the observation of the Tyrrell-026 and GA(W)-1 profiles, examples of the two prolific profile styles i.e. fore- and aft-loaded, have been shown to be subject to laminar phenomena at Reynolds numbers representative of those at which a wing would operate when placed on a full-scale racing car. The Tyrrell-026 wing, however, displayed reduced Reynolds-number sensitivity in comparison to the GA(W)-1 section.

5.3 Summary & Discussion

It has been shown through a variety of experimental techniques that a laminar separation bubble will form on a wing in ground effect at Reynolds numbers equivalent to a full-scale racing car. It was shown that as the ground clearance reduces, the effect of the separation on the wing's static pressure distribution increases, thus showing that laminar and transitional effects become more influential in close ground effect. It is postulated that this is a result of the stronger favourable pressure gradient that occurs as the ground clearance reduces. This has the effect of suppressing perturbations in the boundary layer and stabilising the boundary layer prior to the adverse pressure gradient. The schematic of separation bubbles shown in Figure 5.13 describes how perturbations in the boundary layer prior to separation lead to a shorter bubble occurring. Thus, when a strong favourable pressure gradient is introduced, which has a stabilising effect, the bubble be-

comes longer. As the shape of the separation bubble causes an acceleration in the adverse pressure gradient such that a constant-pressure region occurs, as shown in Figure 2.6, a longer bubble will have more effect on the static pressure distribution. It is not only transitional characteristics that ground effect alters, however, as a study by Roberts et al. [85] of a racing car front wing found that yaw sensitivity increased as ground clearance reduced.

Whilst reducing ground clearance produced a longer bubble, increasing Reynolds number or incidence resulted in a shorter bubble. The wing's Reynolds-number sensitivity is unsurprising given that a full-scale racing car front wing is in the Reynolds number range where laminar characteristics are still evident. In general, as the Reynolds number increases a higher suction value is generated and thus the downforce coefficient increases. Work by Jacobs & Sherman [116] showed that many NACA profiles exhibited a higher lift coefficient as Reynolds number increased, which is synonymous with greater suction or higher pressure values being produced. Jacobs & Sherman [116] stated that lift coefficient is dependent entirely on boundary-layer behaviour, which in turn is a function of viscosity, and thus by extension the Reynolds number.

As incidence is increased the separation bubble becomes smaller and is eventually eliminated as transition occurs in the attached boundary layer. This is a result of the adverse pressure gradient, which has a destabilising effect on the boundary layer, becoming increasingly greater. The perturbations in the boundary layer lead to a shorter bubble being produced initially, but as incidence is increased further the perturbations are amplified sufficiently that the boundary layer does not separate. The Reynolds-number sensitivity of the static pressure distribution reduced as incidence increased. This can be attributed to both the separation bubble being shorter, and thus having a smaller impact on the static pressure distribution, and the laminar boundary layer covering a smaller chordwise extent.

Overall, it has been shown that the laminar boundary layer and separation-induced transition are important aspects of the aerodynamic characteristics of wings operating in ground effect at motorsport-relevant Reynolds numbers. Moreover, these aspects are not solely dependent on Reynolds number, as both incidence (or downforce capability) and ground clearance also have significant impact.

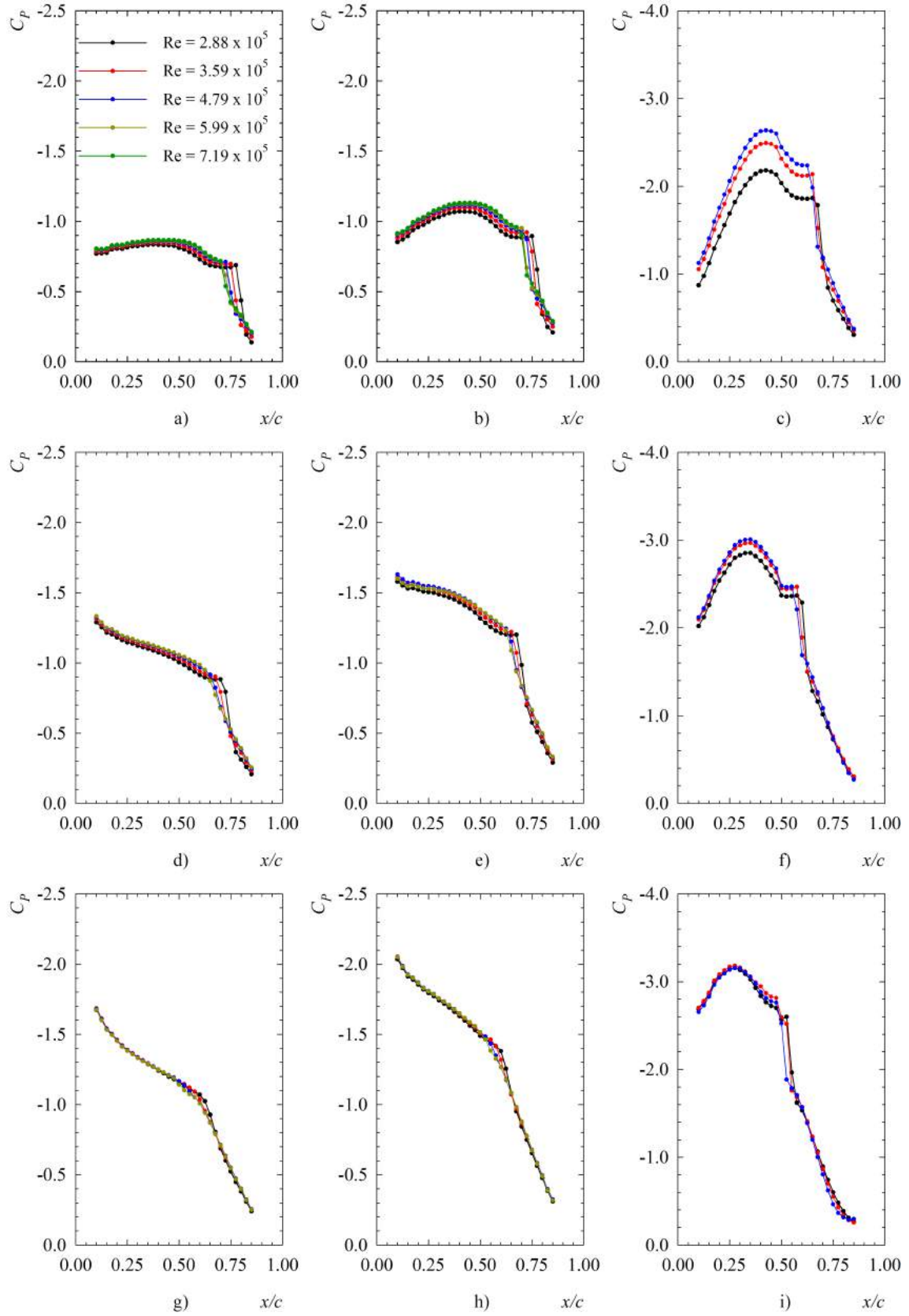


Figure 5.1: Suction-surface static pressure distribution at varying Reynolds number, a-c) $\alpha = 0^\circ$, d-f) $\alpha = 3^\circ$, g-i) $\alpha = 5^\circ$, and varying ground clearance of $h/c = 0.5$ (left column), $h/c = 0.3$ (centre column), and $h/c = 0.1$ (right column) (Experimental)

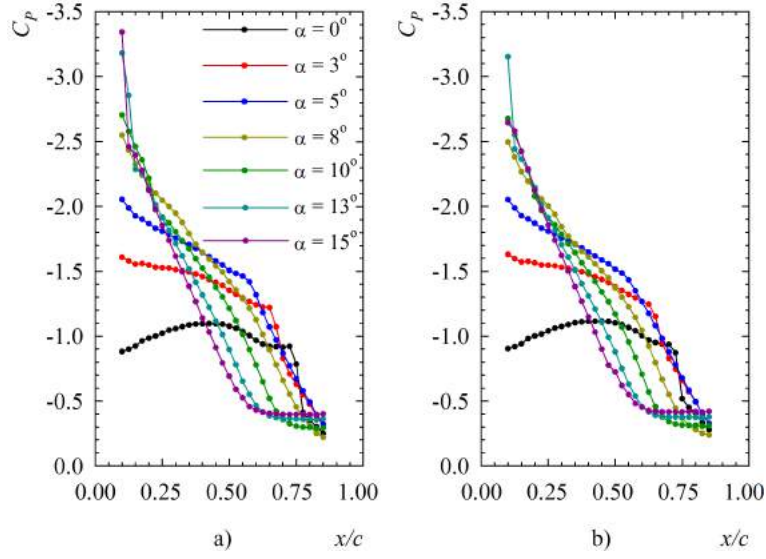


Figure 5.2: Suction-surface static pressure distribution at varying incidence $0^\circ \leq \alpha \leq 15^\circ$ at a) $Re = 3.59 \times 10^5$ and b) $Re = 4.79 \times 10^5$ (Experimental)

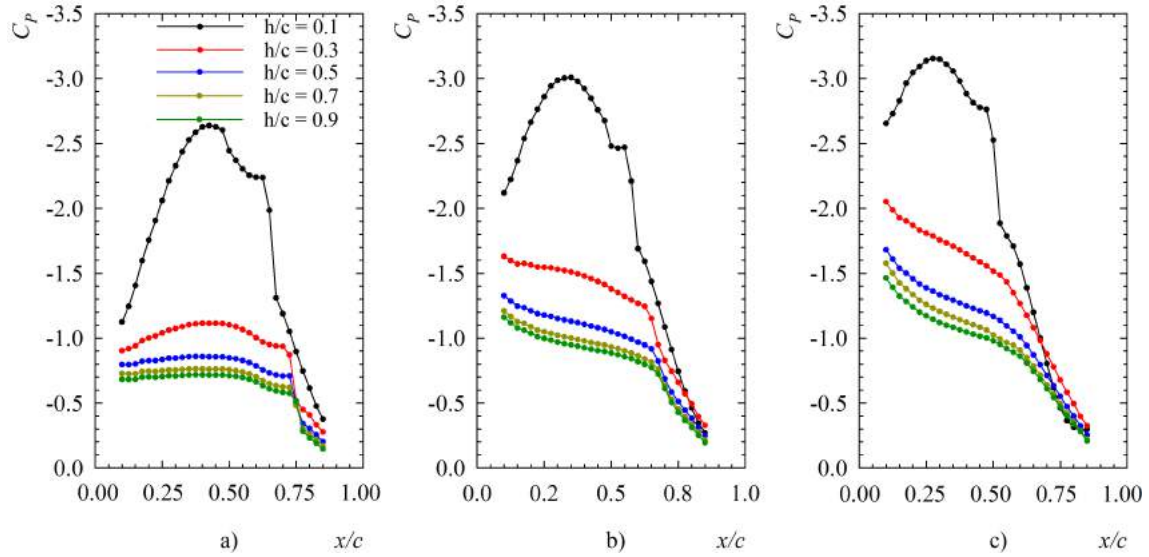


Figure 5.3: Suction-surface static pressure distribution at varying ground clearance $0.1 \leq h/c \leq 0.9$ at a) $\alpha = 0^\circ$, b) $\alpha = 3^\circ$, and c) $\alpha = 5^\circ$ at $Re = 4.79 \times 10^5$ (Experimental)

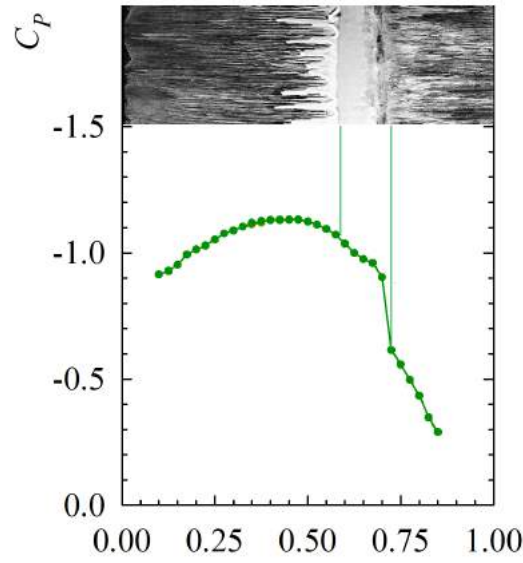


Figure 5.4: Suction-surface static pressure distribution with flow visualisation to show key flow features (Experimental flow visualisation and computational static pressure distribution, $Re = 7.19 \times 10^5$, $\alpha = 0^\circ$, $h/c = 0.3$, flow moving left to right)

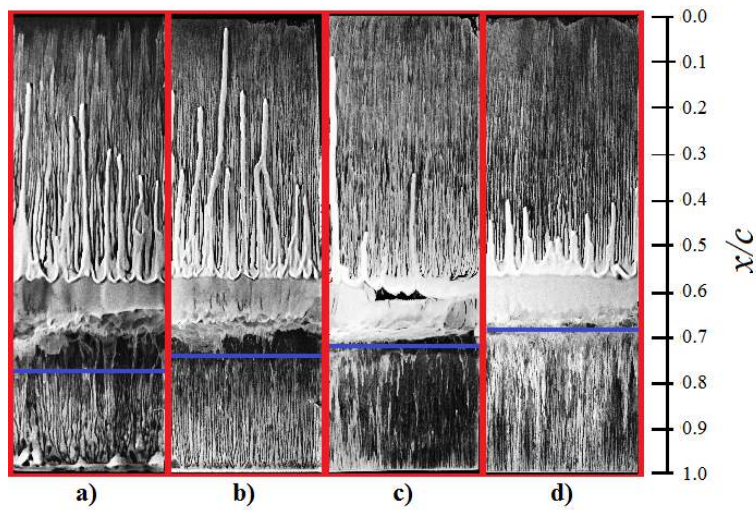


Figure 5.5: Suction-surface flow visualisation at a) $Re = 3.59 \times 10^5$, b) $Re = 4.79 \times 10^5$, c) $Re = 5.99 \times 10^5$ and d) $Re = 7.19 \times 10^5$ (Experimental, $\alpha = 0^\circ$, $h/c = 0.3$, flow moving top to bottom, blue line showing reattachment)

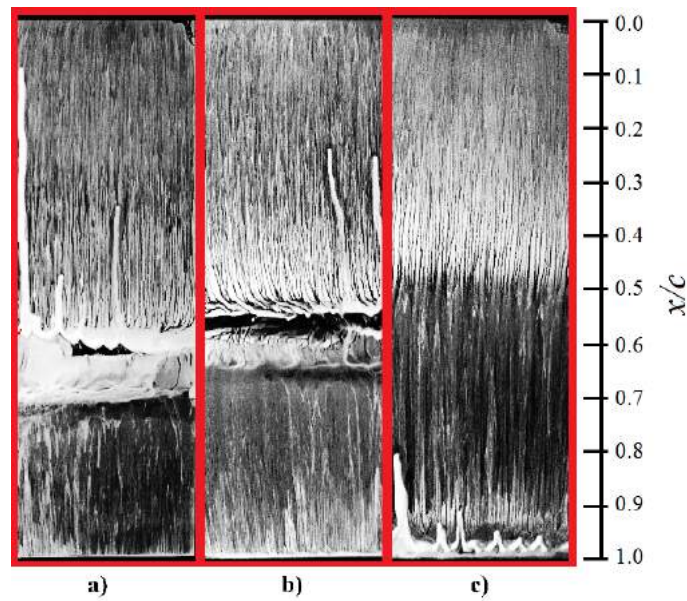


Figure 5.6: Suction-surface flow visualisation at a) $\alpha = 0^\circ$, b) $\alpha = 3^\circ$, and c) $\alpha = 5^\circ$ (Experimental, $Re = 4.79 \times 10^5$, $h/c = 0.3$, flow moving top to bottom)

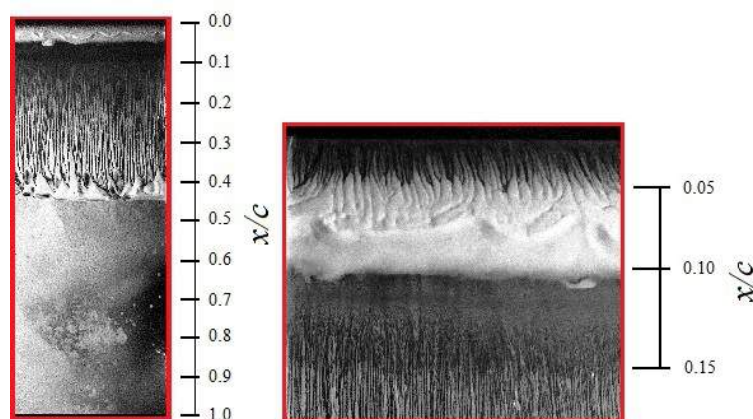


Figure 5.7: Suction-surface flow visualisation at $\alpha = 15^\circ$ (Experimental, $Re = 4.79 \times 10^5$, $h/c = 0.3$, flow moving top to bottom) - close up view of leading edge on the right

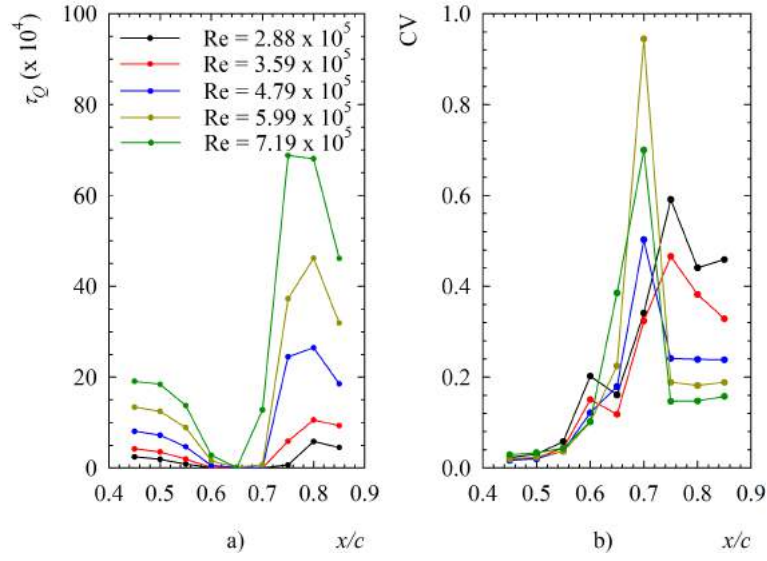


Figure 5.8: a) τ_Q and b) CV of hot-film gauges at varying Reynolds number (Experimental, $\alpha = 0^\circ$, $h/c = 0.3$)

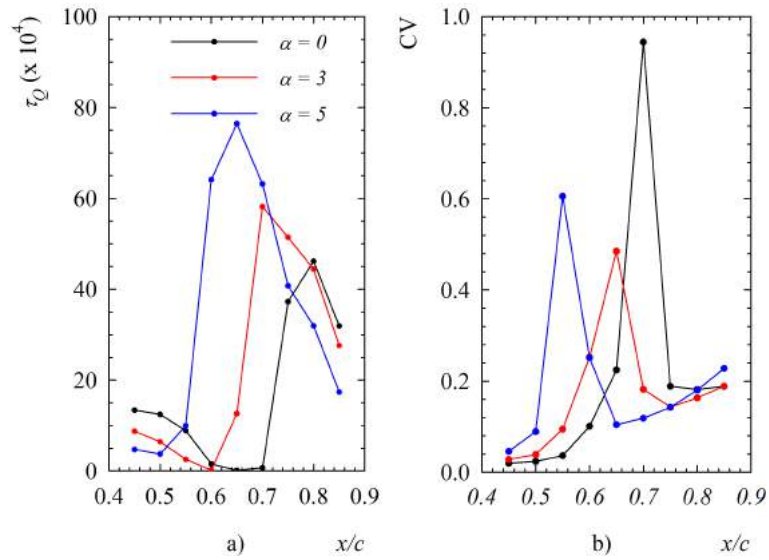


Figure 5.9: a) τ_Q and b) CV of hot-film gauges at varying incidence (Experimental, $h/c = 0.3$, $Re = 5.99 \times 10^5$)

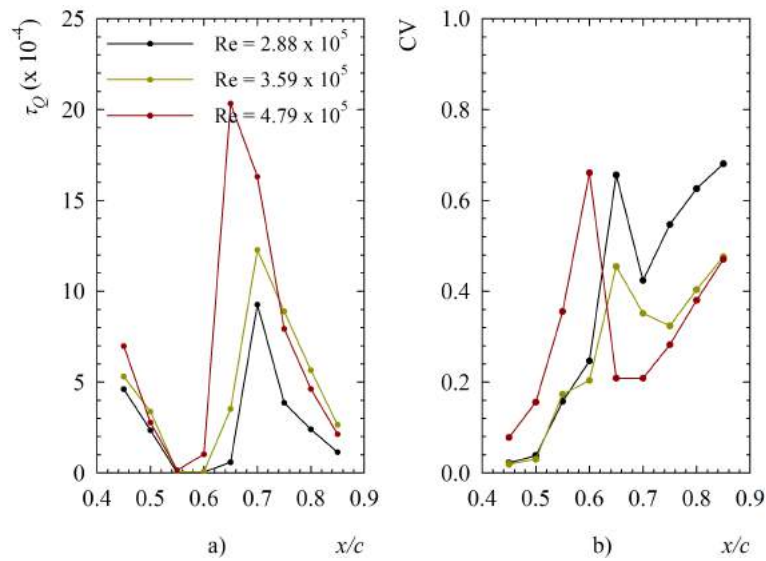


Figure 5.10: a) τ_Q and b) CV of hot-film gauges at varying Reynolds number (Experimental, $\alpha = 0^\circ$, $h/c = 0.1$)

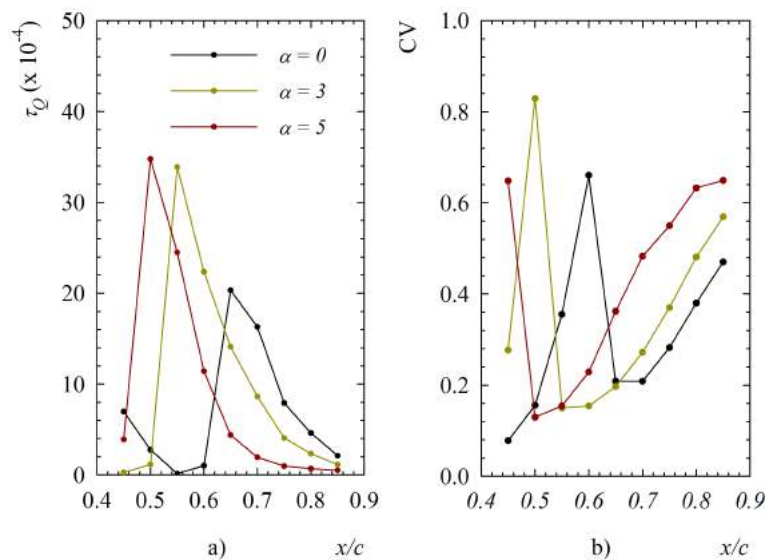


Figure 5.11: a) τ_Q and b) CV of hot-film gauges at varying incidence (Experimental, $h/c = 0.1$, $Re = 4.79 \times 10^5$)

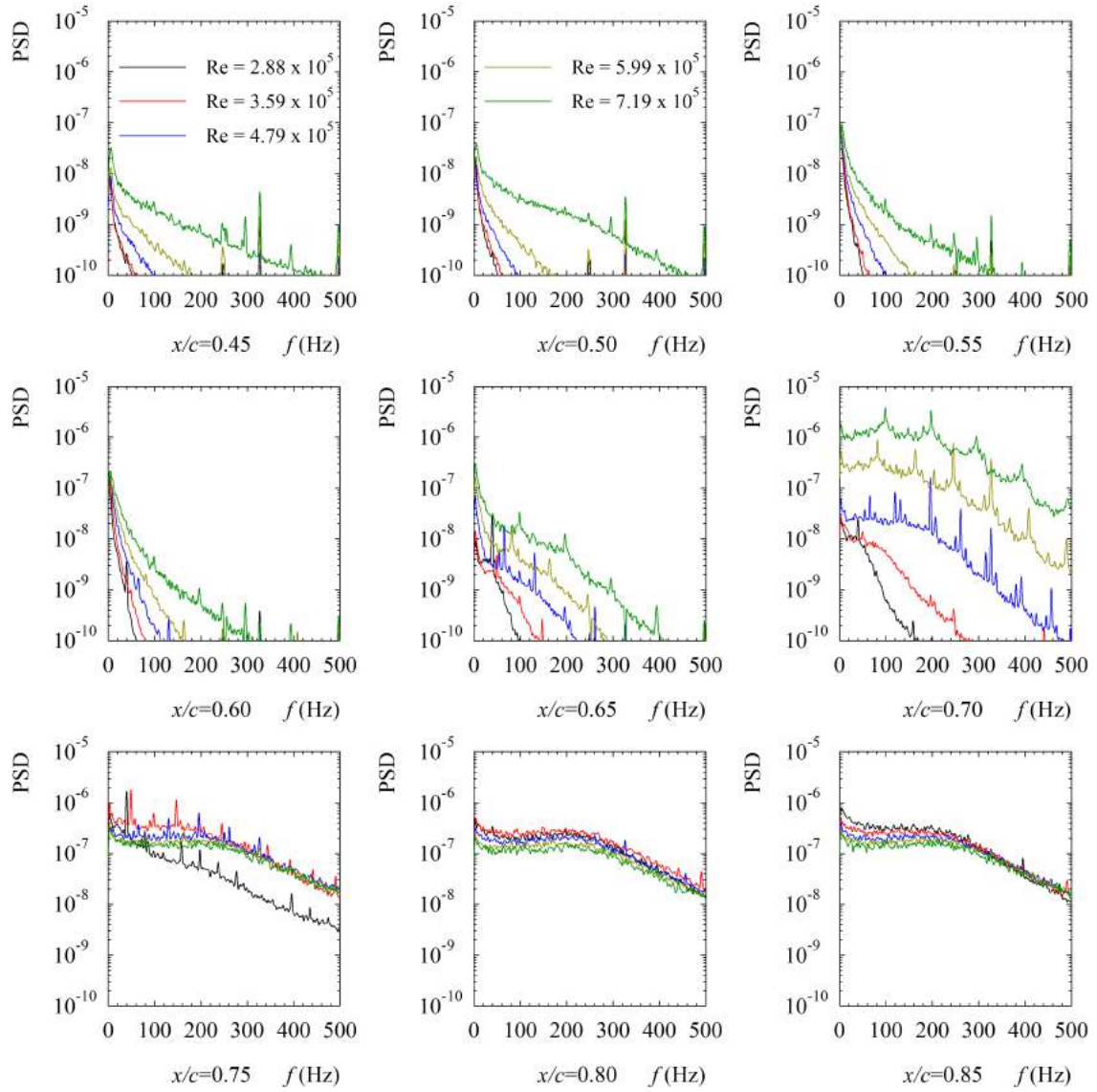


Figure 5.12: PSD of hot-film gauges for varying Reynolds number (Experimental, $\alpha = 0^\circ$, $h/c = 0.3$)

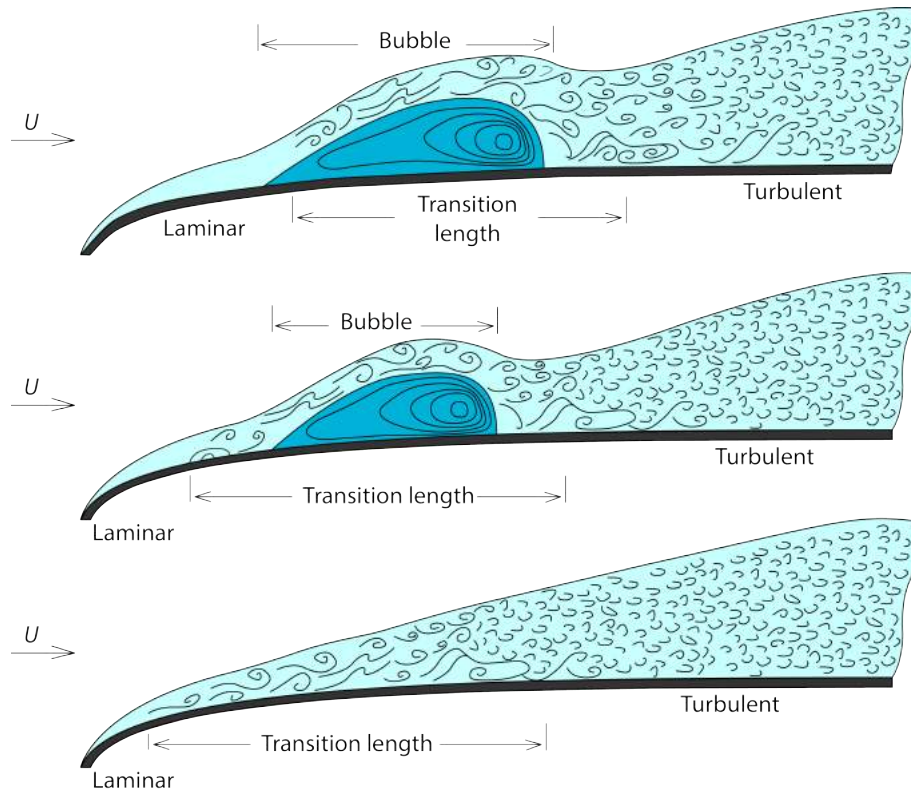


Figure 5.13: Diagram of separation bubbles of varying length due to growth of instabilities in the pre-bubble boundary layer

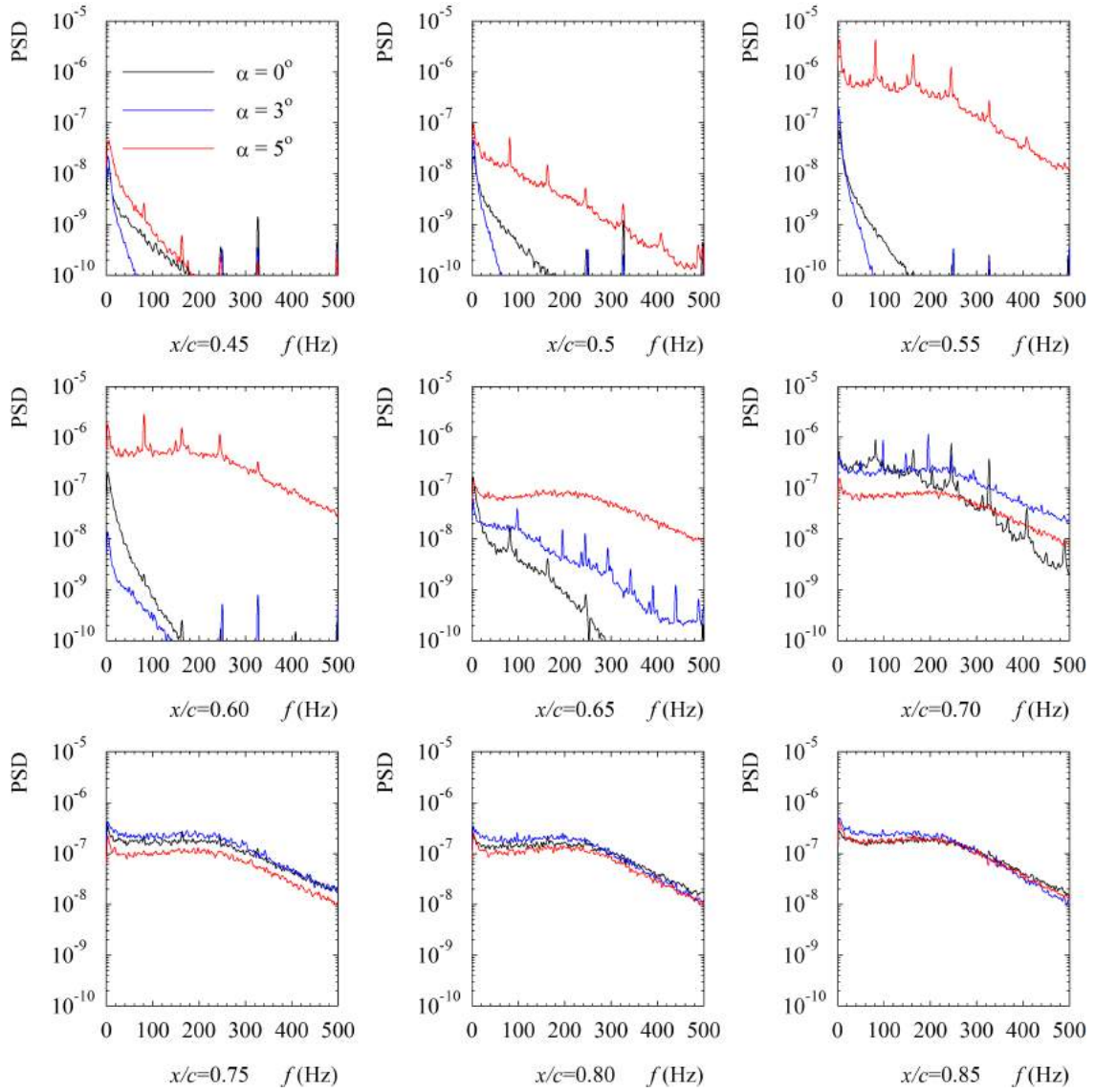


Figure 5.14: PSD of hot-film gauges for varying incidence (Experimental, $h/c = 0.3$, $Re = 5.99 \times 10^5$)

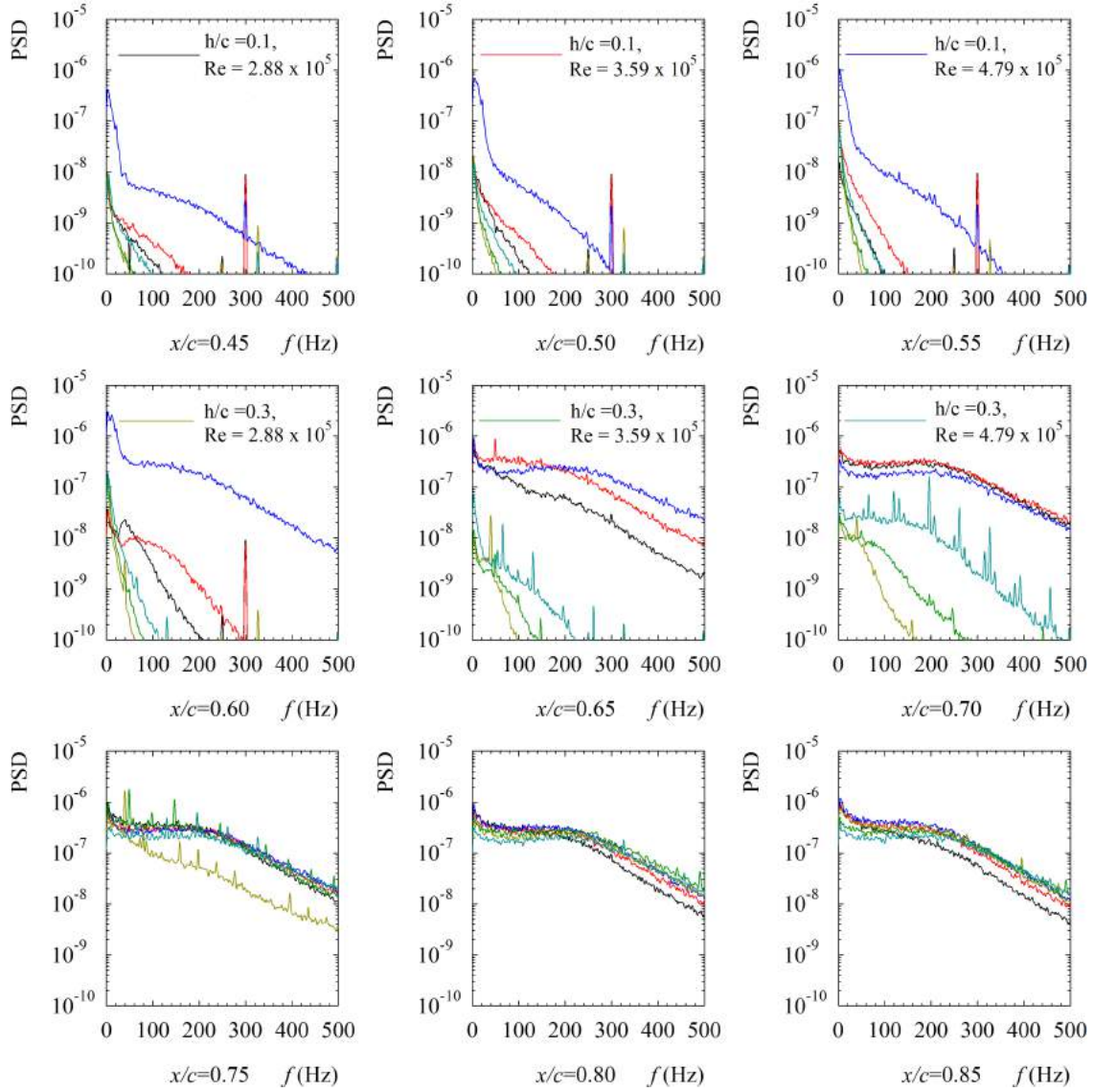


Figure 5.15: PSD of hot-film gauges for varying ground clearance and varying Reynolds number (Experimental, $\alpha = 0^\circ$)

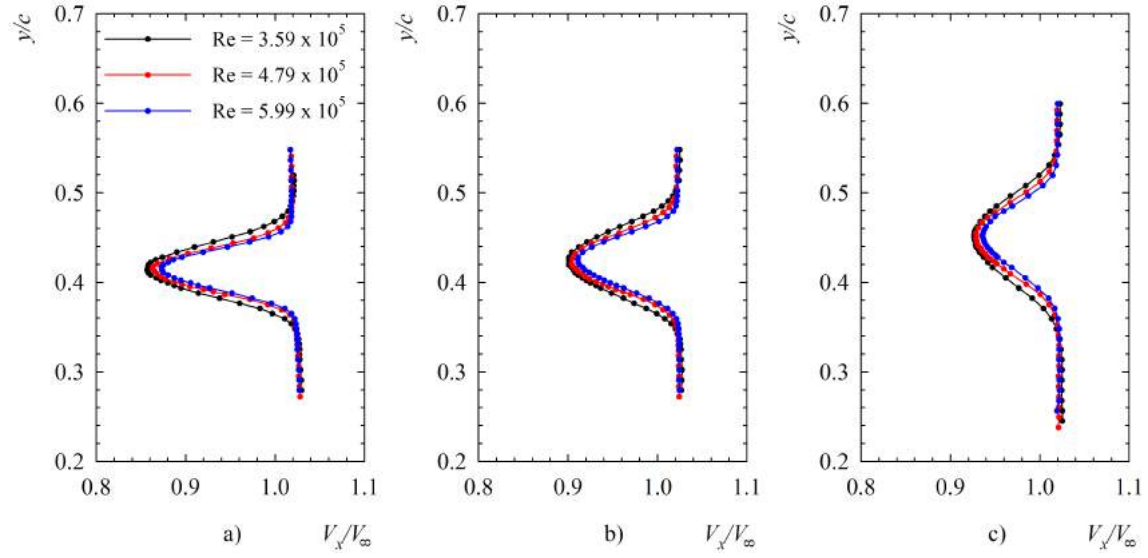


Figure 5.16: LDA wake survey of x-velocity at varying Reynolds number at a) $x/c = 1.5$, b) $x/c = 2.0$, and c) $x/c = 3.0$ (Experimental, $h/c = 0.3$, $\alpha = 0^\circ$)

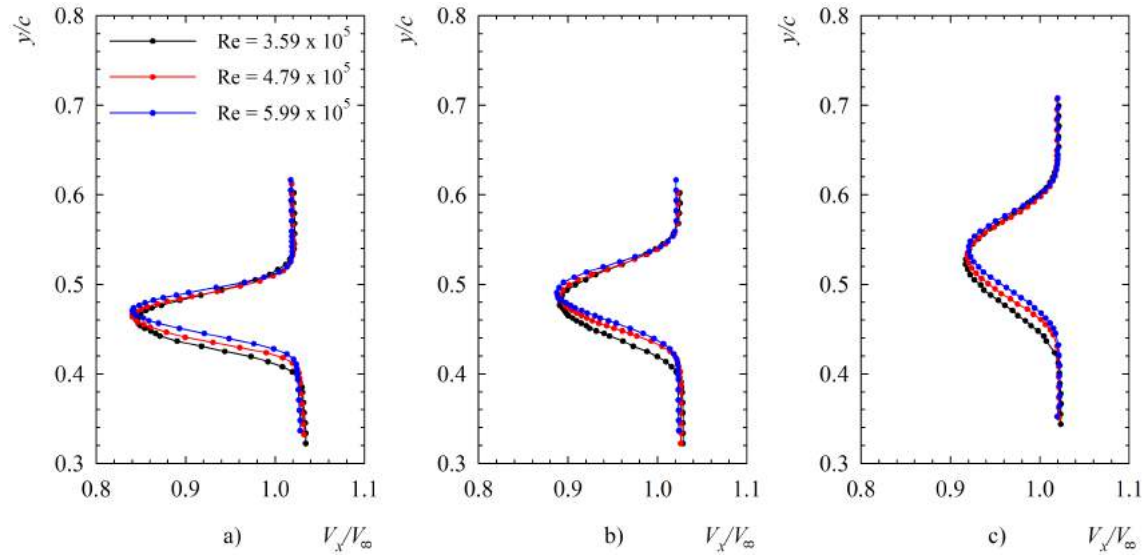


Figure 5.17: LDA wake survey of x-velocity at varying Reynolds number at a) $x/c = 1.5$, b) $x/c = 2.0$, and c) $x/c = 3.0$ (Experimental, $h/c = 0.3$, $\alpha = 3^\circ$)

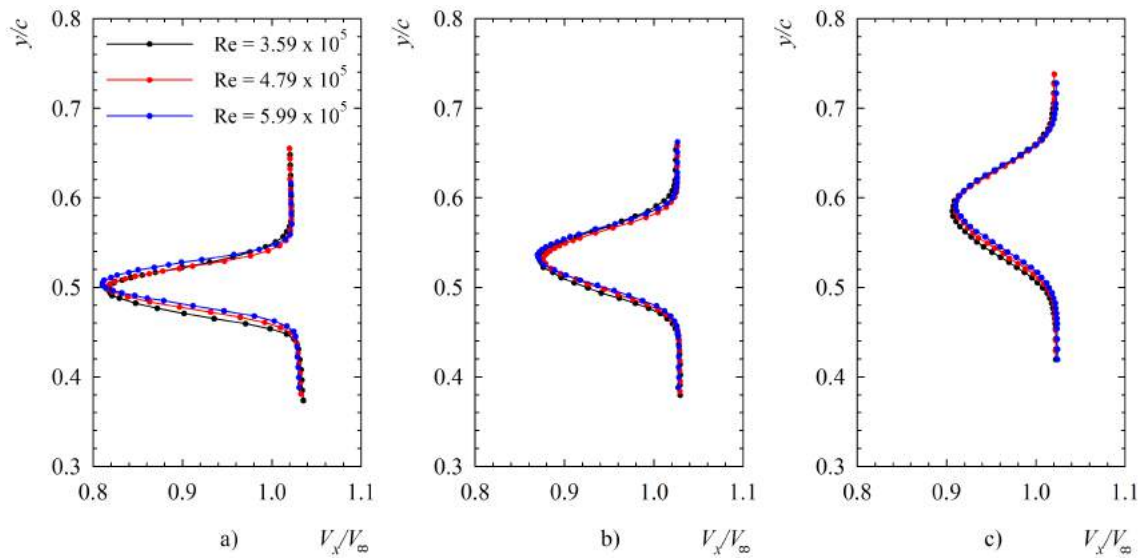


Figure 5.18: LDA wake survey of x-velocity at varying Reynolds number at a) $x/c = 1.5$, b) $x/c = 2.0$, and c) $x/c = 3.0$ (Experimental, $h/c = 0.3$, $\alpha = 5^\circ$)

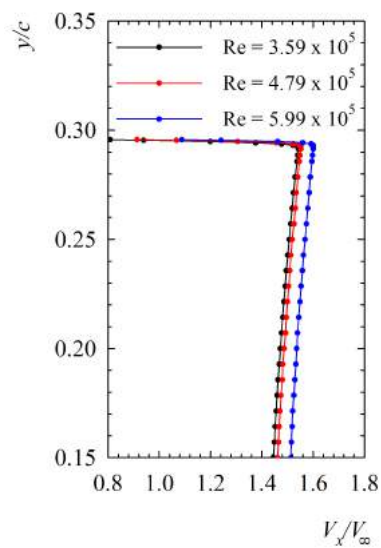


Figure 5.19: LDA survey of x-velocity under the wing at varying Reynolds number at $x/c = 0.375$ for $\alpha = 0^\circ$ (Experimental, $h/c = 0.3$)

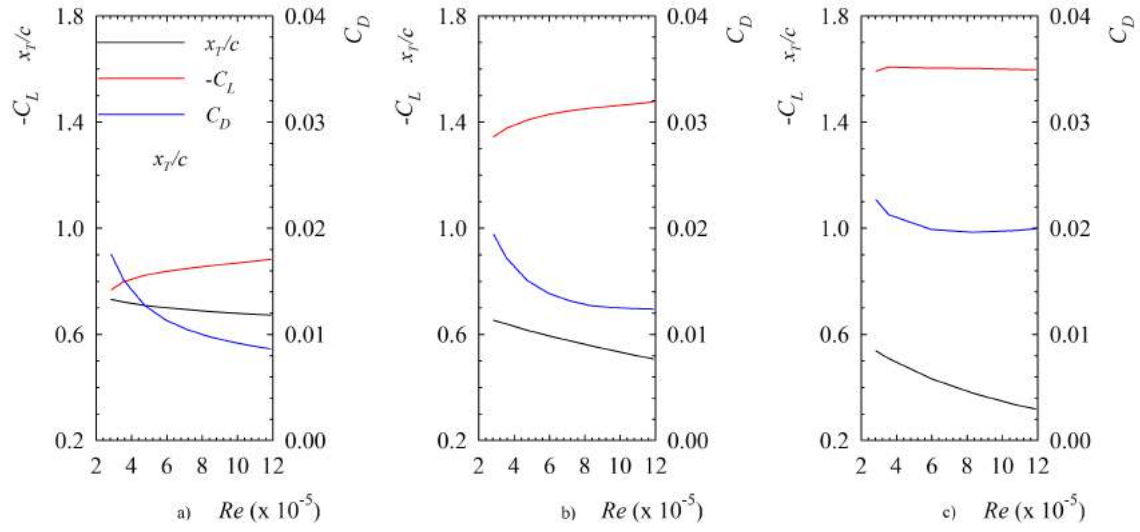


Figure 5.20: Variation of location of transition (x_{Trans}), downforce coefficient and drag coefficient with Reynolds number a) $\alpha = 0^\circ$, b) $\alpha = 3^\circ$, and c) $\alpha = 5^\circ$ (MSES, $h/c = 0.3$)

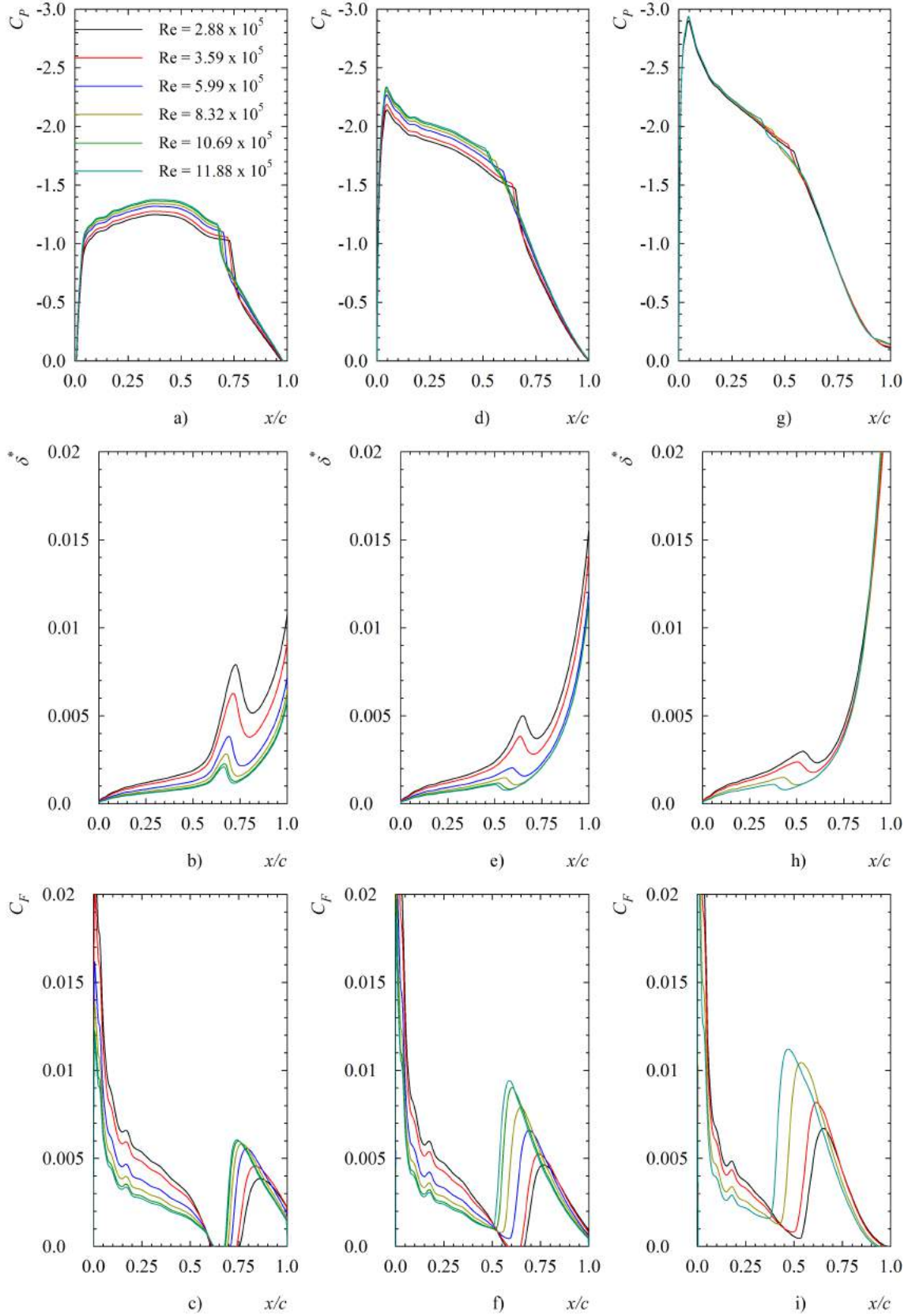


Figure 5.21: Chordwise distributions of pressure coefficient, boundary-layer displacement thickness and friction coefficient at varying Reynolds number and incidence a-c) $\alpha = 0^\circ$, d-f) $\alpha = 3^\circ$, and g-i) $\alpha = 5^\circ$ for GA(W)-1 aerofoil (MSES, $h/c = 0.3$)

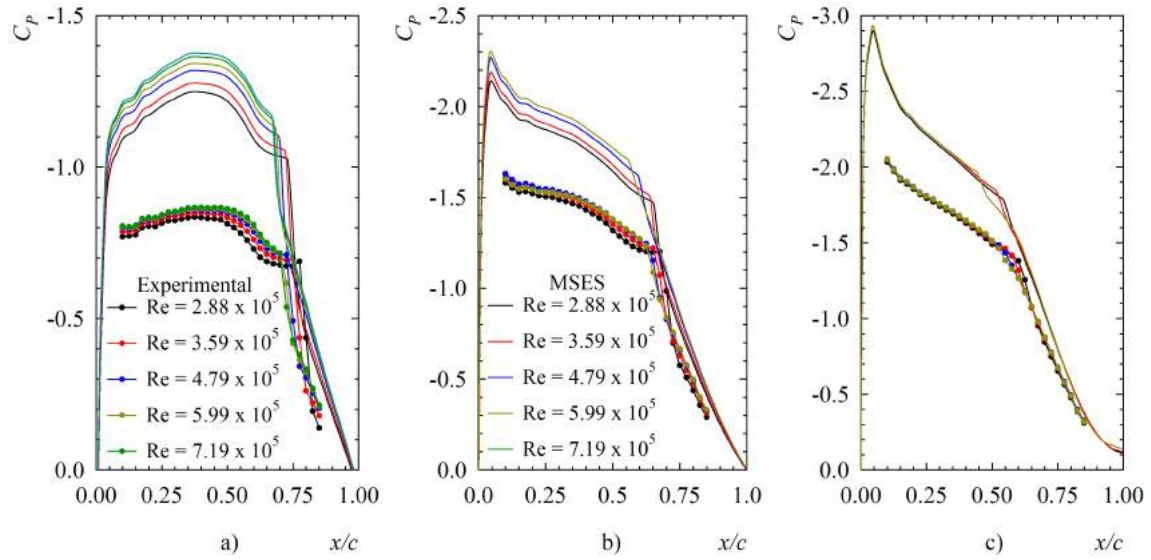


Figure 5.22: Comparison of pressure coefficient distributions from MSES (plain lines) and experiments (dotted lines) at a) $\alpha = 0^\circ$, b) $\alpha = 3^\circ$, and c) $\alpha = 5^\circ$ for GA(W)-1 aerofoil ($h/c = 0.3125$, $Re = 2.44 \times 10^5$)

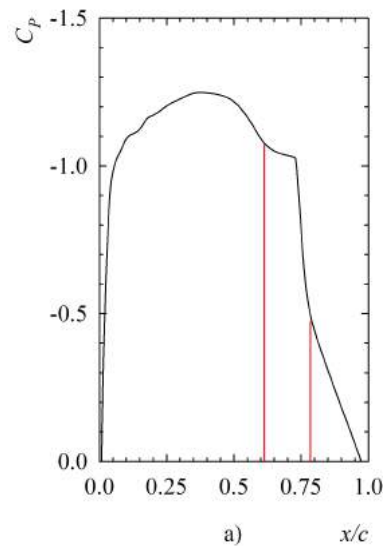


Figure 5.23: Diagram of measurement area for Table 5.2, red lines show the start and end of bubble (defined by $C_F = 0$)

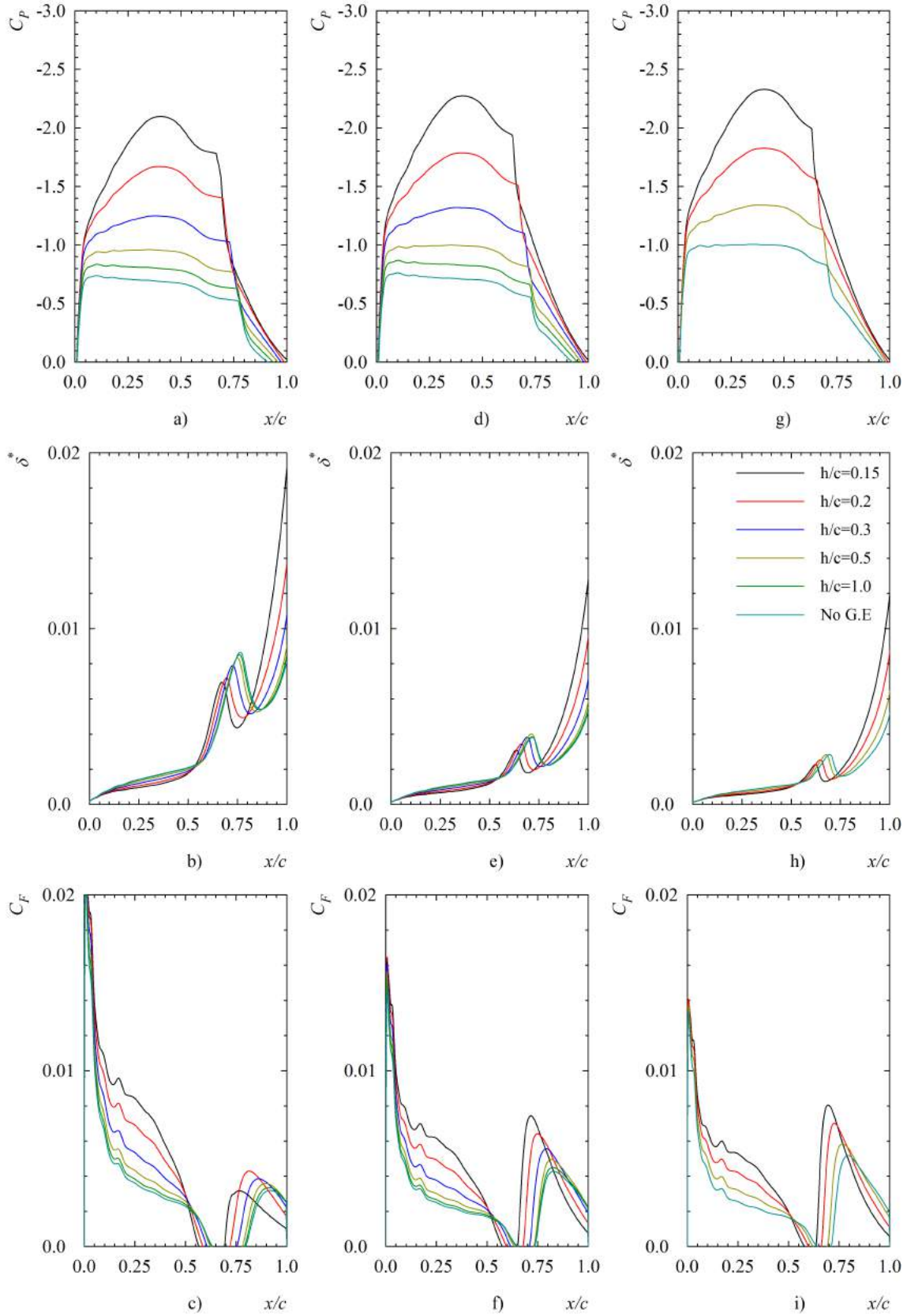


Figure 5.24: Chordwise distributions of pressure coefficient, boundary-layer displacement thickness and friction coefficient at various ground clearances for a-c) $Re = 2.85 \times 10^5$, d-f) $Re = 5.94 \times 10^5$, and g-i) $Re = 8.32 \times 10^5$ for GA(W)-1 aerofoil (MSES, $\alpha = 0^\circ$)

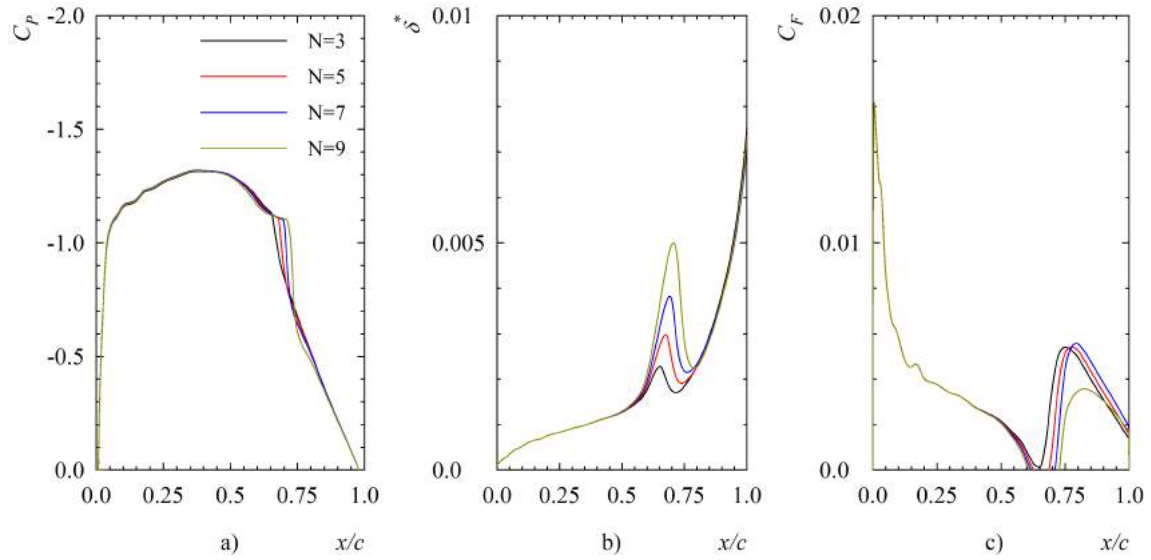


Figure 5.25: Chordwise distributions of pressure coefficient, boundary-layer displacement thickness, and friction coefficient at varying turbulence intensity for GA(W)-1 aerofoil (MSES, $Re = 5.94 \times 10^5$, $\alpha = 0^\circ$, $h/c = 0.3$)

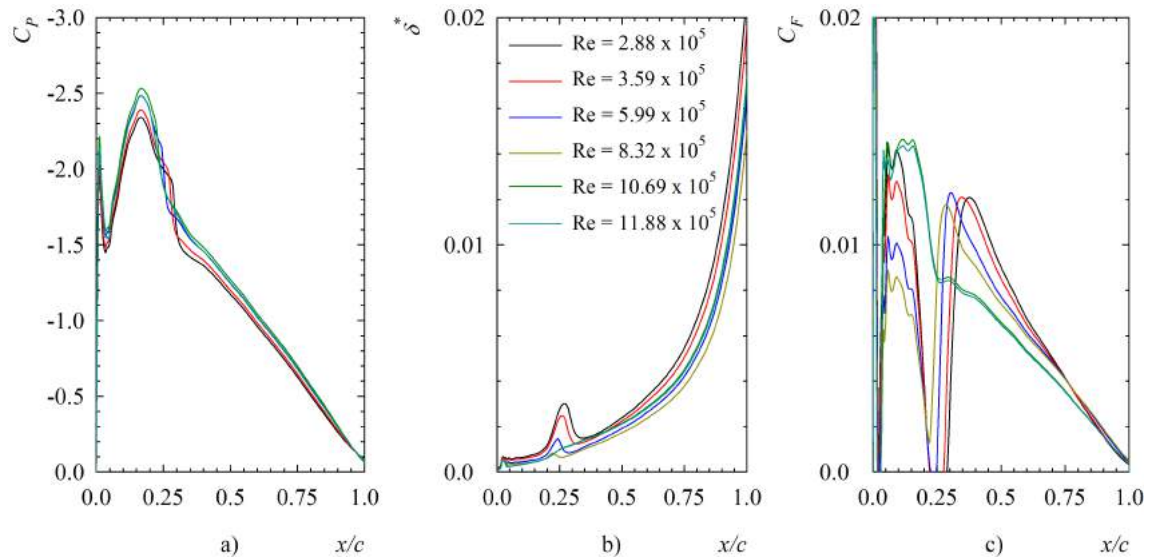


Figure 5.26: Chordwise distributions of pressure coefficient, boundary-layer displacement thickness, and friction coefficient at varying turbulence intensity for Tyrell-026 aerofoil (MSES, $h/c = 0.3$, $\alpha = 0^\circ$)

Chapter 6

Results - Three-Dimensional Wings in Free Transition

In the previous chapters the two-dimensional flow physics have been evaluated. In this chapter, the complexity of the geometry is incrementally increased. Three-dimensional single-element and multi-element CSW (Correia simplified wing [81]) configurations are used to observe the aerodynamic characteristics of what are essentially simplified versions of a racing car front wing before, finally, a 2009-specification F1 wing is investigated.

6.1 Single-Element Wing (CSW-SE)

6.1.1 Force Measurements

The graphs presented in Figure 6.1 show the curves of downforce and drag coefficient for varying ground clearance and Reynolds number. Previous studies have shown that as ground clearance reduces the downforce produced by the wing increases due to the

constraining of flow, which produces greater suction levels. This is true down to a critical ground clearance, after which the adverse pressure gradient is too large for the flow to overcome. The drag coefficient, on the other hand, typically increases with reducing ground clearance at all times, even in the force-reduction region.

Figure 6.1 shows that Reynolds number has a significant effect on both force-enhancement and force-reduction phenomena. Figure 6.1b shows how the rate of downforce increase or decrease changes with reducing ground clearance ($\frac{\delta C_L}{\delta h/c}$). A positive value of $\frac{\delta C_L}{\delta h/c}$ shows an increase in downforce as the wing moves towards the ground, where the higher the magnitude the larger the rate of increase, and a negative value indicates a decrease in downforce with reduced height.

At the lowest tested Reynolds number ($Re = 1.63 \times 10^5$) there is no noticeable force-enhancement, as at no point does the downforce increase with reducing ground clearance; instead there is only the force-reduction phenomenon, which begins to occur at $h/c = 0.5$. The downforce then continually decreases as the wing moves closer to the ground. The median Reynolds number ($Re = 2.24 \times 10^5$) displays what could be considered the typical downforce curve for a wing in ground effect in that it features a force-enhancement region, as downforce increases almost exponentially from $h/c = 1.67$ to $h/c = 0.208$, followed by a force-reduction region. Analysis of the rate of change of downforce with ground clearance ($\frac{\delta C_L}{\delta h/c}$) in Figure 6.1b infers that even at the maximum tested ground clearance the wing is operating in ground effect, as between the two highest test points at $Re \geq 1.83 \times 10^5$ the downforce increases. Although the amount is only very small, the ground is clearly still having an influence on the wing. The highest Reynolds number ($Re = 2.85 \times 10^5$) shows the complete opposite phenomenon to the lowest tested Reynolds number in that only the force-enhancement region is observed. Downforce continually increases from the highest ground clearance across all tested ground clearances, hence

giving the high downforce attained in the experiment. The $\frac{\delta C_L}{\delta h/c}$ curve, however, shows that the rate of downforce increase is starting to decrease and thus it can be assumed that the force-reduction region, where $\frac{\delta C_L}{\delta h/c} < 0$ is not too far away. It should be noted that the minimum tested ground clearance of $h/c = 0.125$ is equivalent to a full-scale wing operating 30 mm off the ground. Typically a wing, by regulation, must be 50 mm above the reference plane of the car (the floor); the floor usually only touches the ground at the end of a straight (under braking the ride height actually increases because downforce reduces with speed), and thus 30 mm is likely to be below the operating window of most racing car front wings.

The drag curves shown in Figure 6.2 exhibit similar behaviour in terms of ground-clearance sensitivity, in that as ground clearance reduces the drag increases at an increasingly greater rate. A second trend is that for any given ground clearance a higher Reynolds number gives a lower drag coefficient; this is no surprise as the same is true for a wing operating out of ground effect. It also appears that all the drag curves are converging towards a single point at very low ground clearance. Although this point was not reached in the tested ground clearance range, it is presumably where the point of separation has moved all the way upstream to the start of the adverse pressure gradient.

Reynolds-number sensitivity is typically eliminated at sufficiently high Reynolds numbers, as the boundary layer will be fully turbulent from close to the leading edge of the wing. This would be indicated by the force remaining constant as Reynolds number changes. Figure 6.3 shows the Reynolds-number dependency at a constant ground clearance; it can be seen that at highest tested Reynolds number the curve begins to asymptote, showing a reduced Reynolds-number sensitivity. At higher ground clearances the sensitivity is significantly reduced, which is not surprising given that for Reynolds numbers above $Re = 2.03 \times 10^5$ the force results are very similar in the $0.25 \leq h/c \leq 1.67$ region.

6.1.2 Surface-Flow Visualisation

Nine flow visualisation cases were conducted on the wing. These represented three ground clearances and three Reynolds numbers. This section is divided into three subsequent sections: firstly with the wing placed at a reference ground clearance and Reynolds number so that the fundamental flow features are presented. After these have been established, the individual effects of ground clearance and Reynolds number are explored.

At reference ground clearance and Reynolds number

For this section, a ground clearance of $h/c = 0.3125$ and Reynolds number $Re = 2.03 \times 10^5$ are considered. This was chosen as it represents a region that has been shown to be low in terms of Reynolds-number sensitivity and which is in the force-enhancement region. Moreover, $h/c = 0.3125$ is arguably the most representative ground clearance for a typical racing-car front wing.

The defining flow feature is the laminar separation bubble. At the centre of the wing (Figure 6.4 - red box) the flow appears highly two-dimensional as the streaklines remain streamwise as they stagnate on the bubble's leading edge; additionally, the separation and reattachment lines both appear spanwise. The reattachment line is very straight and there appears to be little unsteadiness in this area. At the wing tip (Figure 6.4 – blue box) the flow is much more three-dimensional due to the movement of air from the higher pressure outside the endplate, and on the upper surface, to the low pressure underneath the wing. This can be seen by the spanwise movement on the endplate. The tip vortex can be seen impinging upon the separation bubble where it gives turbulent mixing. The paint appears to move spanwise towards the wing's centreline, which demonstrates that the wing is most loaded in the central section (because the flow moves towards the lowest pressure).

Reynolds-number effect

Figure 6.5 provides flow visualisation at three Reynolds numbers, $Re = 1.63 \times 10^5$, $Re = 2.04 \times 10^5$, and $Re = 2.44 \times 10^5$ at a constant ground clearance of $h/c = 0.3125$. Hence, Figure 6.5c is the reference condition observed in Figure 6.4 and discussed in the preceding section. At the lower Reynolds numbers there is trailing-edge separation, to a varying degree, and noticeable reverse flow. As expected the lower the Reynolds number, the further upstream the separation point.

Observation of the laminar separation bubble in Figure 6.5 shows that the bubble's size and location are dependent on Reynolds number. Using Catia V5 the separation point (x_S/c), reattachment point (x_R/c), and length of the bubble (x_L/c) were manually measured at the centreline of the wing. Accurate readings were made possible by using known distances such as the wing's chord, distance between holes and hole diameters. The results are presented in Table 6.1. Overall, as Reynolds number increases the separation point moves downstream and the bubble becomes shorter.

Another remark can be made on the reattachment line. As the Reynolds number increases the line appears more defined, and it seems that there is less unsteadiness along the reattachment line. This is due to the instabilities in the boundary layer being greater at higher Reynolds numbers, as a result of viscous effects reducing and inertial effects becoming dominant such that turbulence production is increased. Hence, in the numerical simulation results presented previously for the two-dimensional wing, skin friction was greater

Table 6.1: Separation bubble size and location at the centreline of the wing at varying Reynolds number (Experimental, CSW-SE, $h/c = 0.3125$)

$Re \times 10^{-5}$	x_S/c	x_R/c	x_L/c
1.63	0.50	0.75	0.25
2.04	0.54	0.78	0.24
2.44	0.56	0.79	0.23

in magnitude after turbulent reattachment for higher Reynolds numbers.

Figure 6.6 gives the effect of Reynolds number at $h/c = 0.125$. This appears to show that for $Re = 1.63 \times 10^5$ a fully-laminar separation occurs i.e. no turbulent reattachment, as the adverse pressure gradient is so large. The vortex which forms directly behind the laminar separation point still occurs, however the boundary layer does not reattach to the surface. This would explain the severe lack of downforce for this case. The $Re = 2.03 \times 10^5$ case, however, is much more perplexing as it does not appear to be any different to the $Re = 2.44 \times 10^5$ case, other than the slight difference in bubble location, this is despite the force curves showing one to be acting in force-enhancement and the other to be in the force-reduction region. Correia et al. [81] cited the reduced effect of the separation bubble, due to it becoming thinner, as a possible reason for this. Unfortunately, without surface pressure measurements no further conclusion can be drawn as to the cause of force-reduction for $Re = 2.03 \times 10^5$.

Whilst the formation of the separation bubble and its Reynolds-number dependency is important due to the influence that they have on the wing, as Figure 6.7 shows, it is not only the separation bubble that is affected by the Reynolds number. Figure 6.7 shows that at lower Reynolds numbers the tip vortex bursts, this can be seen close to the trailing edge and by the lack of spanwise flow on the bottom of the endplate. As the Reynolds number is the only variable between the cases, the bursting of the wing-tip vortex must be due solely to the reduced Reynolds number. The other noticeable aspect of Figure 6.7 is that at higher Reynolds numbers there is spanwise flow inside the bubble; thus showing a larger spanwise pressure gradient. This is likely to be due to the trailing-edge separation of the lower Reynolds number case causing a loss of suction in the central region and thus making the pressure gradient negligible. The fact that flow is being drawn towards the centre, and not towards to tip, at the higher Reynolds numbers is shown by

the streamlines curving inboard at the separation line.

Ground proximity effect

Observation of Figure 6.8 and Table 6.2 shows that as the ground clearance reduces the separation point moves upstream and the bubble becomes shorter, a result of the pressure gradient becoming much larger. At the lowest tested ground clearance of $h/c = 0.125$ and at $Re = 2.44 \times 10^5$ the wing is still in the force-enhancement region, as at this Reynolds number no force-reduction phenomenon occurred. At this ground clearance the laminar boundary layer separates at $x/c = 0.52$, further upstream than at the higher ground clearances, due to the laminar boundary layer not being capable of overcoming the larger adverse pressure gradient generated when the wing is this close to the ground. Additionally, the shortest bubble is produced due to the higher flow velocity under the wing, which produces Kelvin-Helmholtz type instabilities and promotes turbulence production. The earlier formation of the bubble means that the suction value it retains (in the terms of the constant-pressure region on the pressure distribution) is greater than those of higher ground clearances, which aids in downforce production. Without static pressure measurements on the wing, however, the effect of the separation bubble cannot be quantified and so it is not known whether the influence of the bubble on the downforce production increases, decreases, or remains constant when varying ground clearance. It was observed in the 2D wing tests, however, that as ground clearance reduced the bubble moved upstream and the region of constant pressure held a greater suction value, thus contributing to force enhancement.

Overall, as ground clearance reduces the adverse pressure gradient becomes progressively greater, thus when the Reynolds number is kept constant the laminar boundary layer does not have enough energy to move further through the adverse pressure gradient before

Table 6.2: Separation bubble size and location at centre at varying ground clearance (Experimental, CSW-SE, $Re = 2.44 \times 10^5$)

h/c	x_S/c	x_R/c	x_L/c
0.125	0.518	0.713	0.195
0.225	0.548	0.757	0.209
0.3125	0.56	0.79	0.23

separating. This leads to the laminar separation point moving upstream as the ground clearance reduces. The reattachment point also moves upstream and the separation bubble becomes shorter at lower ground clearances. This is due to an increase in turbulence production from higher velocity shear increasing Kelvin-Helmholtz instabilities.

6.1.3 RANS Simulations

CFD simulations have been completed in order to understand the off-surface flow-field around the wing. Given the overall aim of understanding boundary-layer transition, it was important that the computational model be validated against wind tunnel experiments. The validated approach will then be used to understand the flow-field around the wing. Lastly, the transition model will be compared to a fully turbulent closure model so that the importance of using transitional models can be shown. It should be noted that the $k - k_L - \omega$ turbulence model was found to be sensitive to inlet conditions, as explained in Appendix A.3.

Validation of datum configuration

Investigation of the fundamental off-surface flow-field of the single-element wing was conducted at $Re = 2.44 \times 10^5$ and $h/c = 0.3125$. This configuration was chosen primarily as it was used as the datum configuration in the experimental tests, but also as the conditions are somewhat favourable to the computational model. RANS simulations

Table 6.3: Separation bubble size and location at the wing's centreline (Experimental, CSW-SE, $Re = 2.44 \times 10^5$)

	C_L	C_D	x_S/c	x_R/c	x_L/c
Experimental	0.627	0.0431	0.56	0.79	0.23
CFD	0.676	0.0399	0.611	0.831	0.221
$\frac{CFD-Exp}{Exp} (\%)$	7.81	-7.42	9.11	5.19	-8.26

have historically struggled with force-reduction phenomena such as trailing-edge separation and shear-layer unsteadiness, both of which were observed at the lower Reynolds numbers and ground clearances that were tested. Thus, this section will validate the computational model with the datum configuration only, the effect of Reynolds number will then be explored with the same method and any deficiencies in the prediction analysed in the applicable section.

The accuracy of the predictions of drag and downforce coefficients is shown in Table 6.3. Downforce coefficient is predicted to 7.81% of the experimental value and drag to -7.42%.

It was shown in the previous section that the occurrence of a laminar separation bubble was arguably the most important surface-flow feature of the wing. The CFD simulation was capable of predicting the separation bubble, as shown in Figure 6.9. A comparison of the bubble size and location is also given in Table 6.3. This shows that the overall bubble length is predicted to within $x/c = 0.09$, but the location is further downstream in CFD than in the wind tunnel. The poor drag estimate may be due to the later formation of the bubble in CFD compared to the experiments, as this results in a smaller portion of the boundary layer being turbulent, and consequently the shear stresses being under predicted. This is because at a given local Reynolds number a turbulent boundary layer will yield higher shear stress than its laminar counterpart.

Overall, it has been shown that the transitional model has been capable of reproducing both the measured forces and surface-flow structures from the experiments to within 9.1% of experimental values.

Flow field around datum configuration

As with any finite-span wing, there is a variation in spanwise loading. The centreline is free from end-effects and hence usually determined to have two-dimensional flow, as it is acted on equally by each end. Spanwise loading for a wing without endplates would feature maximum downforce generation at the centre of the wing. The geometry of the endplates, however, means that a vortex forms in the junction between the wing and the endplate that provides vortex-induced suction to affect the spanwise loading. Moreover, the nose section has an inherent effect on the flow field. The nose induces a downwash that increases the effective incidence of the central section of the wing. Therefore, as Figure 6.10 shows, the pressure distributions at different spanwise locations are significantly different. The downwash makes the wing act as if it were operating at a higher incidence and causes increased circulation. The change in stagnation point also means that a higher pressure is attained over the majority of the upper surface. Despite larger load being developed in the central region, the bubble position does not change significantly, as surface streamlines in Figure 6.9 and the pressure distribution in Figure 6.10 show. The fact that the adverse pressure gradient is similar for both cases, despite the difference in suction, clearly plays a part in this. This would appear to be at odds with the nose increasing the effective incidence of the central section of the wing. However, another consequence of the nose section is to reduce the wing's base pressure; hence the lower pressure at $x/c = 1$ for the $z/s = 0$ distribution. At $z/s = 2/3$ significantly lower suction and pressure values are produced at the front portion of the wing and the distribution appears similar to that

of the two-dimensional aerofoil of Section 5.2; again showing the influence of the nose on the wing.

The formation of the laminar separation bubble on the suction surface of the wing is shown by off-surface streamlines, generated by components of x- and y-velocity, in Figure 6.11. Within the dividing streamline, which shows how the flow has been displaced from the surface to alter the effective shape of the wing, the region of recirculating air inside the bubble is captured. The contours behind the streamlines show that the production of turbulent kinetic energy begins shortly after separation, before reaching a maximum during the reattachment process. This is expected as the instabilities amplified in the separated shear layer generate the turbulence that promotes reattachment. In the region ahead of the separation bubble, energy is contained entirely as laminar kinetic energy, then a portion of this energy is redistributed into turbulent kinetic energy in the separated shear layer. This occurs because the $k - k_L - \omega$ model is based on the redistribution of energy via a pressure-strain mechanism, as opposed to the production or dissipation of energy. Such characteristics are also shown in Figure 6.11, which plots turbulent kinetic energy and the x-component of shear stress (in the chordwise direction) for the centreline of the wing's suction surface. The separation bubble is shown by the area of zero shear stress. The x-component of shear stress is used instead of skin friction coefficient as the additional components, spanwise and vertical, influence the value; hence isolating the x-component makes it easier to observe the bubble's location. The maximum shear stress value occurs close to the leading edge; as the flow moves downstream shear stress decreases as the local Reynolds number increases. There is a small amount of turbulent kinetic energy produced at the leading edge, but this is quickly dissipated due to the strong flow acceleration that then occurs. There is then a lack of turbulent kinetic energy until transition occurs. Laminar separation occurs at $x/c = 0.611$; transition then begins to

occur in the separated shear layer as turbulent kinetic energy increases at $x/c \approx 0.675$ and reaches a maximum at $x/c \approx 0.779$. The turbulent boundary layer then reattaches at $x/c \approx 0.831$, as shown by the regrowth of shear stress.

Varying Reynolds number

The Reynolds number can be viewed as a measurement of the ratio between viscous and inertial forces in a given flow, where viscosity acts to dissipate turbulence and dampen instabilities whereas inertial forces promote turbulence production. Transitional phenomena are therefore directly dependent on the Reynolds number and, as such, it is important that a transitional turbulence model responds to these characteristics. The prediction of force coefficients given by the CFD simulations compared to those of wind tunnel tests at various Reynolds numbers are given in Table 6.4. A clear trend can be observed in that as the Reynolds number increases the accuracy of the force prediction improves. This is not particularly surprising as the flow characteristics are somewhat more complex at lower Reynolds numbers, as observed in Section 6.1.2 where the turbulent boundary does not fully reattach. Such an occurrence is seen at both the lower tested Reynolds numbers of $Re = 1.63 \times 10^5$ and $Re = 2.04 \times 10^5$, but not at the highest Reynolds number where the turbulent boundary layer remains fully attached.

Figure 6.12 shows the separation at various Reynolds numbers through both on- and off-surface streamlines. Whilst the force prediction of the lower Reynolds number cases was generally poor, the characteristics it shows in terms of flow structure are encouraging. The main problem with the lowest Reynolds number case is the size and location of the separation bubble. Similarly to the $Re = 2.44 \times 10^5$ case, which was observed in the previous section, the separation point for the bubble occurs at a more downstream location than that observed in the wind tunnel. As the Reynolds number is reduced the length of

Table 6.4: Comparison of experimental and computational downforce and drag coefficients, and errors as a percentage of full scale for CSW-SE at varying Reynolds numbers ($h/c = 0.3125$)

Re		C_L	C_D	C_L Error (%FS)	C_D Error (%FS)
1.63×10^5	Experiment	0.383	0.0639	-	-
	CFD	0.67	0.0473	41.92%	-35.26%
2.04×10^5	Experiment	0.587	0.0485	-	-
	CFD	0.689	0.0432	14.8%	-12.13%
2.44×10^5	Experiment	0.627	0.431	-	-
	CFD	0.676	0.0399	7.22%	-8.01%
2.85×10^5	Experiment	0.646	0.405	-	-
	CFD	0.692	0.380	7.20%	-4.07%

the bubble increases, as it should, but it becomes too long. This may be due to it occurring further downstream than the experiments, and therefore further into the pressure recovery region, where the flow velocity is lower and the Kelvin-Helmholtz instabilities that produce turbulence are inhibited. Consequently, turbulent reattachment occurs much further downstream because the adverse pressure gradient is smaller and leads to no trailing-edge separation occurring for the lower Reynolds number cases, as would be expected. The lack of trailing-edge separation leads to the over-estimate of downforce and under-estimate of drag. At $Re = 1.63 \times 10^5$ a long bubble is formed, as shown by the surface streamlines. The velocity streamlines show that a bubble is being produced that ends at approximately $x/c = 0.88$; however, the streamlines do not reattach to the surface behind the bubble and instead travel parallel to the surface. This leads to skin-friction drag being underestimated compared to the wind tunnel tests. Given that at a lower Reynolds number the friction drag makes up a larger proportion of the total drag, this contributes to the drag prediction becoming progressively poorer with reducing Reynolds number. For the higher Reynolds number cases a clearly defined bubble is produced, as the boundary layer correctly reattaches.

An additional deficiency of the lower Reynolds number RANS results is that the laminar

separation point is predicted too far downstream. The $Re = 1.63 \times 10^5$ case predicted a separation point farther downstream than its higher Reynolds number counterparts when it should have been a more upstream position. This is a considerable drawback because although the turbulence production characteristics of the closure model can be fairly easily tuned through turbulence quantities at the inlet boundary condition, it is difficult to alter the separation and reattachment points independently of one another. This could be achieved through alteration of the model constants but this is outside the scope of this study.

Whilst the problems with the lower Reynolds number results are clear, there are a number of positives with how the model behaved in such conditions. Firstly, although the location of the bubble may not be correct, it is clear that the bubble has increased in both length and height as the Reynolds number was reduced; as it should. Additionally looking at the point at which turbulent kinetic energy begins to be introduced to the flow-field, which essentially shows the transition location, it can be seen that as Reynolds number increases the transition point moves upstream; again as it should.

It has been shown that the $k - k_L - \omega$ model is capable of recreating wind tunnel results though the inclusion of transitional effects. Although the model struggled at lower Reynolds numbers, at the higher tested Reynolds numbers the estimation of sectional forces and surface-flow structures were very good. It is fairly common for RANS-based simulations to struggle with stall and force-reduction characteristics, and therefore the lack of prediction of trailing-edge separation for the lower Reynolds number cases is not necessarily a disadvantage of this specific closure model. The characteristics of the separation bubble becoming shorter and thinner, and the transition point moved upstream as Reynolds number increased were all captured. Given that the higher Reynolds numbers used in this study are for a 50% scale model at wind tunnel speeds, it shows that for

Table 6.5: Comparison of experimental and computational downforce and drag coefficients for CSW-SE for fully-turbulent and transition-sensitive turbulence models ($h/c = 0.3125$)

Case	C_L	C_D	C_L Error	C_D Error
Experiment	0.627	0.0431	-	-
$k - k_L - \omega$	0.676	0.0399	7.22%	-8.01%
$k - \omega$ SST	0.565	0.0442	-11.00%	2.52%

full-scale and full-speed car models the model could give very accurate results.

Comparison of transitional and fully-turbulent closure models

The previous section showed that the transitional $k - k_L - \omega$ model is capable of predicting the existence of a laminar separation bubble. In order to show the importance of simulating transitional effects through the use of transitional closure models, comparisons to a classical, fully-turbulent approach are made in this section. As the $k - k_L - \omega$ model is based on the $k - \omega$ framework it was deemed appropriate that $k - \omega$ SST be used as the fully-turbulent model for comparison. Additionally, $k - \omega$ SST was shown by Mahon & Zhang [91] to be the best turbulence model for predicting the sectional forces of a single-element wing in ground effect.

Table 6.5 shows the prediction of downforce and drag coefficient by the two closure models in comparison to the experimental results. The accuracy of the transitional $k - k_L - \omega$ model was discussed in the previous sections and so will not be discussed further. Although it is immediately clear that the fully-turbulent $k - \omega$ SST provides a better prediction of drag, but a poorer prediction of downforce.

It is clear that the separation bubble is an integral component of the surface-flow structure. Figure 6.13 shows on-surface streamlines from $k - k_L - \omega$ and $k - \omega$ SST. With the latter, a fully-turbulent boundary layer forms from the leading edge of the wing; this means

that no laminar separation can occur and, thus by extension, no bubble forms. Instead, turbulent trailing-edge separation occurs. As a turbulent boundary layer is more resilient to adverse pressure gradients, due to fluctuating velocity components giving wall-normal momentum transfer, the turbulent separation occurs much further downstream than the laminar separation bubble. There are two consequences of trailing-edge separation: a decrease in downforce due to the reduction of circulation, and an increase in drag. Additionally, greater shear stresses caused by the turbulent boundary layer, which stretches across the entire suction surface, gives an increase in friction drag.

From Table 6.5 and Figure 6.13 it is clear that there are significant differences in the force production and surface-flow structures given by the two models. Figure 6.14a shows the difference between the two predicted pressure fields around the wing. Significantly smaller suction values are produced underneath the wing and less pressure above it in the $k - \omega$ SST result, as would be expected from the lower downforce it produces. This leads to a considerably different pressure field between the two models, which has an impact on flow direction even quite far away from the wing; as such on a full-car model the choice of closure model could impact on how flow moves over downstream components. The maximum difference in pressure coefficient is $C_P \approx 0.24$, which occurs in two locations: at the suction peak, and at $x/c \approx 0.7$, where the bubble occurs for the transitional model. The separation bubble has been shown to be a force-enhancement mechanism as it alters the effective shape of the wing to give it more camber, hence the region of constant pressure in the free-transition pressure distribution (Figure 6.10), and increased circulation. This increase in circulation leads to the suction level across the entire lower surface being increased. Additionally, altering the aerodynamic forces that the wing produces, given that the wing is of a finite span, the increase in suction below the wing and reduction in pressure above gives a smaller pressure gradient between surfaces, which can affect the

formation of wing-tip vortices. Hence, the smaller pressure gradient between the upper and lower surfaces with the $k - \omega$ SST model weakens the formation of such vortices.

Figure 6.14b compares the wakes given by each closure model. The plot shows the difference in total pressure between the two results (ΔP_0); positive values show lower total pressure in the $k - \omega$ SST result, and negative values a lower total pressure for the $k - k_L - \omega$ result. The positive values of ΔP_0 upstream of $x/c \approx 0.6$ show the thicker boundary layer produced by the $k - \omega$ SST model, due to the turbulent boundary layer, which is thicker than the laminar counterpart, beginning at the leading edge. Negative values of ΔP_0 at $0.6 \leq x/c \leq 0.8$ show where the separation bubble occurs, as the region of dead air encompassed in the bubble is of low energy. Downstream of the bubble and close to the trailing edge, positive values of ΔP_0 are formed due to the trailing-edge separation of the $k - \omega$ SST result, indicating a thicker wake. Overall, the total pressure loss at the trailing-edge of the wing shows how the $k - \omega$ SST model produces a wake that is both thicker and of lower energy.

Turbulent kinetic energy (k) around the wing is given in Figure 6.14c. The two closure models clearly produce turbulent kinetic energy in completely different manners. The $k - k_L - \omega$ model includes both laminar and turbulent kinetic energy variables, thus prior to transition through the separation bubble there is no production of turbulent kinetic energy. Conversely, the fully-turbulent formulation of the $k - \omega$ SST model means that turbulent kinetic energy is produced immediately from the leading edge, due to the turbulent boundary layer. Upstream of $x/c \approx 0.65$, the negative values of Δk show this thick turbulent boundary layer. For the $k - k_L - \omega$ model turbulent kinetic energy starts to be produced at $x/c \approx 0.67$, showing the point that the transition process starts, before reaching a maximum at $x/c \approx 0.775$. This can be seen by the positive values of Δk in the aft-region of the wing. The key difference between the models, therefore, is that $k - \omega$

SST produces the majority of turbulent kinetic energy close to the leading edge and little at the trailing edge. The $k - k_L - \omega$ model, by contrast, produces none at the leading edge and a large amount near to the trailing edge. The fact that only positive values of Δk occur downstream of $x/c \approx 0.7$, shows that the wake of the $k - k_L - \omega$ model contains more turbulent kinetic energy; hence the turbulent flow behind the wing is predicted very differently by the two models.

6.2 Multi-Element Wing (CSW-ME)

6.2.1 Force Measurements

Curves of downforce and drag against non-dimensional ground clearance for Reynolds numbers $1.66 \times 10^5 \leq Re \leq 2.44 \times 10^5$ are shown in Figure 6.15. Higher Reynolds numbers could not be tested due to the large loads that the wing was producing. The presented curves are similar to those found by Zhang & Zerihan [72, 73] for a double-element wing. The downforce curves show the three regions that Zhang & Zerihan observed. At higher ground clearances ($h/c \geq 0.3$) the wing operates in the force-enhancement region ($-C_L$ increases as h/c is reduced) with concentrated tip vortices; in this region there is negligible change in downforce production with varying Reynolds number. As the ground clearance continues to reduce the downforce continues to increase, but at a slightly reduced rate, which can be attributed to the loss of force-enhancement from the tip vortices that have burst. Despite the loss of vortex-induced suction the wing continues to produce more downforce in this region due to the constraining of flow between the wing and ground outweighing the loss of vortex-induced downforce. There is more of a Reynolds number dependency for the downforce coefficient in this region. Although the wing continues to produce more and more downforce in this region at all Reynolds numbers, the

rate of increase of downforce with reduction in ground clearance is less for the lower Reynolds number cases. In both of the force-enhancement regions, with and without concentrated vortex cores, variation of downforce with Reynolds number is less than 2%. Hence, at the typical wing operating height, Reynolds-number sensitivity is minimal. At the lowest ground clearances the wing operates in the force-reduction region, which is characterised by trailing-edge separation. The Reynolds-number dependency of the wing is considerably higher in this region than any other, with the force reduction being larger for the lower Reynolds number cases. This is a consequence of the thicker boundary layer and less energy in the flow at lower Reynolds number. At all tested Reynolds numbers the wing entered the force-reduction region, however, if it had been possible to test at even higher Reynolds number then it is expected that, similarly to the single-element wing, that no force reduction would have occurred at the minimum tested ground clearance.

The drag curves, which are given in Figure 6.15b, display a Reynolds-number dependency at all tested ground clearances. In the force-enhancement region, the difference in drag with varying Reynolds number at a given ground clearance remains consistent, with less drag being produced for increasing Reynolds number. In the force-reduction region, however, larger discrepancies begin to occur. Given the significant difference in downforce at these ground clearances, this is not surprising. Unlike the drag curves of the single-element wing, the drag curves for the multi-element wing do not continually increase in the force-reduction region, as in some cases the drag slightly reduces. This is likely a result of the multi-element wing's total drag having a larger induced drag component because of the stronger vortices it produces at the main-plane wing tip and the additional vortices which shed from the inboard flap edge. Thus, the loss of downforce in the force-reduction region weakens these vortices such that the induced drag component is sufficiently reduced that it outweighs any increase in pressure drag, thus giving a lower

total drag.

The Reynolds-number sensitivity of downforce and drag can be seen more clearly in Figure 6.16, in which the percentage loss of force coefficients for each Reynolds number case is compared to the datum case of $Re = 2.44 \times 10^5$. For downforce, at $h/c \geq 0.75$ the difference between all cases is less than 1% of the value at $Re = 2.44 \times 10^5$. Below $h/c = 0.75$ the Reynolds number discrepancy becomes increasingly larger, where the lower the Reynolds number the greater the downforce loss. The difference in drag coefficient with varying Reynolds number becomes progressively smaller at ground clearance reduces from $h/c = 1.45$ to $h/c = 0.3$, albeit only very slightly. Below this ground clearance, however, the difference then increases. In both cases, prior to the force-reduction region the Reynolds-number sensitivity is relatively small, especially when compared to that of the single-element wing (CSW-SE).

6.2.2 Surface-Flow Visualisation

At reference ground clearance and Reynolds number

To understand the fundamental surface-flow structures, the results of flow visualisation tests at $Re = 2.44 \times 10^5$ and $h/c = 0.3125$ are first presented as a datum.

Both the suction and pressure surfaces of the CSW-ME are shown in Figure 6.17. The pressure-surface flow visualisation shows a number of separation bubbles forming: one on the flap, one on the outer main-plane region and a large one at the centre main-plane region. The bubble on the main-plane is no surprise because of the thick GA(W)-1 section, however the noticeable aspect is how much larger and unsteadier the bubble in the central region is than at the outer span. This bubble is bounded by the pylons and the nose section, so its pressure gradient is significantly different to that of the outer region. The

nose imposes a large pressure on the upper surface as the two surfaces act as a converging nozzle. Hence, a larger bubble is seen at the central region than at the outer-span region.

Similarly to the CSW-SE configuration, a separation bubble occurs on the main plane suction surface (Figure 6.17a) and clear regions of laminar and turbulent flow can be seen (Figure 6.18a). By comparison with the CSW-SE, the bubble appears much more three-dimensional. The chordwise length of the bubble changes significantly across the span, with a shorter bubble being present at the centre and a longer bubble in the outer region ahead of the flaps. Furthermore, the bubble does not appear as a build-up of paint (which indicates zero shear stress) as before, but instead flows laterally across the wing. This lateral shear stress component shows that the reverse-flow vortex that occurs inside the bubble is not two-dimensional and the flow is moving towards a point on the main-plane ahead of the flaps. This point can be seen by the build-up of paint at approximately mid-flap ($z/s = 2/3$). The flow likely converges towards this point as it is the point of greatest suction, given its alignment to the centre of the flap. There also appears to be a bubble on the flap suction surface, however, it is not as clearly defined as the one on the main-plane; the lack of paint and close-observation of the streamlines appears to confirm its existence. Unfortunately, the join between the two pieces of material that make up the flap is located precisely in this area, hence the bubble forms on top of the silver tape placed over this join. At the inboard edge of the flap, flow can be seen to have a large spanwise component. This is because a strong vortex is generated in this region due to the large pressure gradient between the upper and lower flap surfaces, and thus flow is being entrained into it.

Figure 6.18b shows a close-up of the bubble and turbulent reattachment at the centre of the main-plane. At the top of the image the bubble has very little spanwise movement because it is at the centreline of the wing, and thus equally acted upon by spanwise pres-

sure gradients. The reattachment line is fairly clear and downstream of this are a number of eddies of varying length scale (and presumably time scale but clearly this cannot be confirmed in the steady-state flow visualisation). A little further down the image the mixing has become so high that there are no clearly-defined eddies but instead just constant colour. In the previous chapter it was shown that skin friction rises after reattachment of the turbulent boundary layer, reaches a maximum, and then reduces towards the trailing edge. Thus, the constant-colour section is likely to be the point of highest skin friction. The size of eddies in the flow visualisation after reattachment appears to match the expected shear stress distribution, based on the results of the previous chapter. The eddies initially become smaller in size until they all blend into the constant-colour section as shear stress reduces. The eddies then become progressively larger as the shear stress reduces towards the trailing edge. Finally, moving further downwards to the trailing edge, larger and larger eddies start to appear. Although this does not give a perfect view of turbulent reattachment, given that turbulence is a subject which is still not fully understood, it does help us understand the development of the reattached boundary layer.

Reynolds number dependency at reference ground clearance

Flow visualisation on the main-plane and flap suction surfaces at the lowest, median and highest tested Reynolds numbers are given in Figure 6.19. Whilst the main-plane does appear to have some dependency on Reynolds number, the flap flow appears very similar in all instances. At the lowest tested Reynolds number trailing-edge separation occurs in the central region of the main-plane; it does not occur ahead of the flaps as their inclusion allows off-surface pressure recovery by lowering the the main-plane base pressure. At the centre of the main-plane for the higher Reynolds number cases, laminar separation occurs later, the overall bubble length is reduced and there is less unsteadiness in the

Table 6.6: Separation bubble size and location at the wing's main-plane centreline for varying Reynolds number (Experimental, CSW-ME, $h/c = 0.3125$)

$\text{Re} (\times 10^{-5})$	x_S/c	x_R/c	x_L/c
1.63	0.51	0.74	0.23
2.04	0.59	0.76	0.17
2.44	0.61	0.77	0.16

reattachment i.e. the reattachment line is more clearly defined. The bubble dimensions are shown in Table 6.6, these are taken at the centreline of the wing using the chord length and bolt holes for reference. Obviously the bubble is much more three-dimensional in this case than for the single-element wing, but these give a general indication of how the bubble changes. Table 6.6 also shows that as the Reynolds number is increased, the separation point moves downstream and the reattachment point moves upstream, hence the bubble becomes shorter in length.

Ahead of the flap, at the outer-span of the main-plane, the size and position of the bubble does not show any significant Reynolds-number sensitivity. This was confirmed by analysing the flow-visualisation photographs with image-processing software, again using geometric features as references. The reattachment line is, however, once again more clearly defined as Reynolds number is increased, thus indicating reduced unsteadiness.

Ground proximity effect

Flow visualisation was conducted at three non-dimensional ground clearances, and results are given in Figures 6.19 - 6.21. Considering constant Reynolds number, as the ground clearance is reduced the separation line moves upstream. At $h/c = 0.3125$, which was observed in the previous sections, the separation line showed three-dimensionality in that it moved downstream closer to the endplates. At the minimum ground clearance of $h/c = 0.125$ (Figure 6.21), however, it appears almost horizontal across the span. As the wing moves closer to the ground the adverse pressure gradient becomes increasingly larger and this causes the laminar boundary layer to separate almost immediately upon entering the pressure recovery region. This region begins at the apex of the wing i.e. the lowest point on the wing in relation to the ground. Trailing-edge separation on the main-plane can be observed at all Reynolds numbers for $h/c = 0.125$, albeit to a varying degree. At the lowest Reynolds number there is a complete separation of the main-plane boundary layer. The laminar boundary layer separates at the beginning of the pressure recovery region, as shown by the build up of paint shows the region of dead air that forms shortly behind the separation point, but the boundary layer does not reattach. Additionally, there is evidence of separation at the trailing edge of the flap. This is a consequence of the full separation of the main-plane reducing the upwash of flow onto the flap's lower surface, making it more difficult for the flow to stay attached. As the Reynolds number increases the amount of trailing-edge separation on the main-plane progressively reduces, and the flow remains fully attached on the flap.

The main-plane wing-tip vortex was strong enough to leave an imprint on the inner surface of the endplate, allowing it to be viewed. Figure 6.22 shows the inner surface of the endplate at $h/c = 0.3125$ and $h/c = 0.125$ and indicates that at the lower ground clearance the vortex has burst shortly after the initial roll-up in the aft region of the main-plane at

$h/c = 0.125$. This can be seen by the highly disrupted streamlines in Figure 6.22b. At $h/c = 0.3125$, a concentrated and stable vortex can be seen across the entire length of the endplate. The vortex also remains attached for the flow-visualisation tests conducted at $h/c = 0.225$ (for all Reynolds numbers). The region on the downforce curve where the rate of downforce increase becomes smaller was attributed to the bursting of the wing-tip vortex causing a loss of suction at the wing tip, based on the work of Zhang & Zerihan [71]. The fact that the vortex remains intact at $h/c = 0.225$, which is situated well into region b (downforce-enhancement region with vortex breakdown - see Figure 2.9), it must be assumed that this is not the case and some other unidentified phenomenon is causing the slowing of downforce increase as it approaches the critical ground clearance. Whilst the main-plane wing-tip vortex bursts at low ground clearance, the streamlines on the flap's lower surface, which curve due to the vortex that forms on its inner edge, show no difference at any ground clearance or Reynolds number. Hence, there is no evidence that the flap vortices are bursting in any test case. This is because the shedding edge is situated at a higher ground clearance for the flap-tip vortex than the main-plane-tip vortex; for which the shedding edge is the lower endplate surface.

6.2.3 RANS Simulations

Validation of datum configuration

In this section, the simulations of the multi-element wing at reference Reynolds number ($Re = 2.44 \times 10^5$) and ground clearance ($h/c = 0.3125$) are validated, and the flow-field explored.

The prediction of downforce and drag coefficients by the simulation are given in Table 6.7. It is clear that the estimation of the forces is highly accurate. This is probably due to the very close representation of the surface-flow structures, as shown in Figure 6.23,

Table 6.7: Comparison of downforce and drag forces for experiments and computations for CSW-ME ($Re = 2.44 \times 10^5$, $h/c = 0.3125$)

Case	C_L	C_D	C_L Error	C_D Error
Experiment	3.471	0.3886	-	-
CFD	3.336	0.3881	-0.14%	-3.12%

which compares the surface flows from CFD and wind-tunnel experiments. As Figure 6.23 shows, the size and shape of the separation on the main-plane suction surface is well predicted. At the centreline the bubble length is almost identical to the experiments, however, its location is slightly further downstream. The separation point correctly moves downstream at more spanwise locations (closer to the endplate) ahead of the flap. At these outer regions, however, the slightly shorter bubble that is predicted occurs further downstream than in the experiments. Overall, the prediction of the separation bubble is remarkably accurate. One deficiency of the computational surface-flow structures, however, is that trailing-edge separation occurs in the central region; this did not occur in this configuration in the wind tunnel. It appears that this small region of separated flow is enough to reduce circulation to give the underestimation of downforce shown in Table 6.7. The key features of the surface-flow on the flap are also accurately predicted by the CFD simulation. The turning of flow into the spanwise direction near to the inboard edge, due to the formation of a tip vortex, and outboard spanwise movement close to the endplate are both captured. The formation of a small separation bubble on the flap can be seen in both, and again the size and location of the bubble in the CFD simulation is very good.

Flow field around datum configuration

The static pressure and shear stress distributions for the suction surface of the wing at inboard (centreline, $z/s = 0$) and outboard (flap mid-span $z/s = 2/3$) locations are given

in Figure 6.24. The shear stress in the laminar boundary layer is fluctuating, which is likely to be a result of either numerical instability of the governing equations solved in the simulation. Figure 6.24 shows that the inclusion of the flap causes significant changes to the flow around the wing and the pressure distribution; this can be seen by comparing the two locations to the single-element wing that was observed previously. The flap introduces an additional favourable pressure gradient on the main-plane such that the point of maximum suction occurs at the mid-chord rather than near to the leading edge. The considerably greater suction generated by the main-plane in this region is due to the flap providing a lower pressure behind the wing to give off-surface pressure recovery; this leads to separation being less likely and the flow being more stable. This can be seen on the pressure distribution of the main-plane and flap together in Figure 6.25, where the C_p at the trailing-edge of the main-plane can be very low as the flap takes over the recovery. Separation does occur, however, in the central region as the adverse pressure gradient is still large, but there is no flap to assist with the pressure recovery. These differences lead to the shear stress upon reattachment of the boundary layer being quite different. At $z/s = 2/3$ the turbulent boundary layer produces high shear stress, which is what would be expected of a turbulent boundary layer compared to its laminar counterpart. At $z/s = 0$, however, the shear stress levels after reattachment are very small, and then the second separation event occurs. Despite the suction values being lower for the central region, which would lead to lower shear stress developing, it is clear that the model is not correctly replicating the turbulent reattachment phase and is, in a manner, over-estimating stall.

Figure 6.25 shows the pressure acting on the wing's upper and lower surfaces through contours of pressure coefficient (C_p) and a graph of pressure coefficient at $z/s = 2/3$ (in line with the mid-span of the flap). The upper surface is as expected, with positive

values of C_P being produced following flow stagnating on the surfaces. The presence of the flap causes an additional area of high pressure to be produced near to the trailing edge of the main-plane. The flap also causes the point of maximum suction to be moved outboard towards the endplate. Whilst for the single-element wing maximum suction was produced at the centre of the main-plane, for the multi-element wing it is produced ahead of the flap at the mid semi-span. In addition to altering the spanwise loading of the wing by producing more suction closer to the endplate, the pressure gradient across the endplate is much larger, causing stronger main-plane tip vortices to be produced. On the inner edge of the flap, the vortex-induced suction produced by the wing-tip vortex can be seen. The location and influence of these two vortices can be observed in the wake surveys of total pressure given in Figure 6.26. The low-total-pressure vortex cores can be clearly seen to occur at the wing tips of the main-plane and flap as previously discussed; an additional vortex is produced on the upper-outer edge of the endplate as flow moves from the high pressure on the flap's upper surface across the endplate to lower pressure outside the endplate. This endplate-top-edge vortex is substantially weaker than the other two, due to the much smaller pressure gradient that exists: from Figure 6.25 it can be seen that the high pressure at the centre of the flap's top surface is $C_P \approx 0.5$, so flow moves across the endplate to the freestream flow ($C_P = 0$) due to a pressure difference of $\Delta C_P \approx 0.5$. In comparison, the difference between the flap's upper and lower surfaces at its inboard edge is $\Delta C_P \approx 1.5$, and at the main-plane wing tip is $\Delta C_P \approx 2.25$. In the previous section it was stated that the flap vortex did not burst at the lowest tested ground clearance but the main-plane wing-tip vortex did. The wake surveys show that the flap vortex forms much further away from the ground, making it more stable at the lower ground clearances.

Figure 6.27 shows streamlines generated by x- and y-velocity components and contours of turbulent kinetic energy at $z/s = 2/3$ (flap centreline) as predicted by the $k - k_L - \omega$

RANS computations. The separation bubble is very thin, especially in comparison to that observed in the previous section for the single-element wing, which is a consequence of the higher flow velocities and thus higher local Reynolds number that occur underneath the wing. This can be observed through the graph of shear stress and turbulent kinetic energy on the suction surface, in that no turbulent kinetic energy exists prior to the laminar separation point. Turbulence production begins approximately half way along the bubble's length. The maximum turbulent kinetic energy occurs at the reattachment point, which is shown by the rise in shear stress. Similar trends are observed on the flap's suction surface.

Reynolds-number effect

Although the experimental results showed that the multi-element wing was predominantly insensitive to Reynolds number changes, it is still important to observe whether the RANS model responds correctly. The force coefficients found from wind tunnel testing and computations are given in Table 6.8 for each Reynolds number case. In all cases the estimation of both drag and downforce are very accurate.

At the lowest tested Reynolds number the surface-flow characteristics, shown in Figure 6.28b, are again well predicted, but not quite as well as at the higher Reynolds number

Table 6.8: Comparison of downforce and drag coefficients from experiments and $k - k_L - \omega$ RANS computations for CSW-ME at varying Reynolds numbers ($h/c = 0.3125$)

Re		C_L	C_D	C_L Error	C_D Error
1.63×10^5	Experiment	3.408	0.396	-	-
	CFD	3.495	0.413	2.51%	4.10%
2.04×10^5	Experiment	3.453	0.392	-	-
	CFD	3.522	0.4070	1.96%	3.72%
2.44×10^5	Experiment	3.471	0.3886	-	-
	CFD	3.536	0.4026	1.85%	3.50%

(Figure 6.28a). The location of separation on the main-plane is again well predicted, however the length of the bubble is over-estimated by the CFD simulation, particularly in the central region. The experimental flow visualisation shows that in the central region a bubble forms as usual, but then trailing-edge separation occurs. It is difficult to see by the surface streamlines alone what is happening in the CFD case, as no defined reattachment line can be seen. However, the direction of the streamlines, particularly close to the flap where they curve around, appears similar to the separation characteristics seen in the flow visualisation. In the outboard region ahead of the flap the separation line is well predicted, although observation of shear stresses in that region shows that full reattachment does not occur. The reattachment appears to be further downstream in CFD, due to a longer bubble. The structures on the flap are again well predicted, although there is a clear bubble in the CFD case which is not particularly clear in the experiments. Unfortunately, the join between the two sections that were bonded together to form the flap is very close to where the bubble is expected to form (based on the CFD result); this was not known when designing the model.

Comparison of transitional and fully-turbulent closure models

Comparison between the $k - k_L - \omega$ and $k - \omega$ SST turbulence models was made for the single-element wing and shown to be significant. For the multi-element wing, it has been shown in this section thus far that transitional effects may not be as significant due to the thinner bubble and lack of Reynolds-number sensitivity. It is, however, still important to compare the transitional and fully-turbulent closure models on the multi-element wing. A comparison of the sectional forces predicted by each closure model and the wind tunnel experiments is given in Table 6.9. Although the $k - \omega$ SST model gives a good estimate of the drag force, an estimate that is better than the transitional model, it greatly under-

Table 6.9: Downforce and drag coefficients generated by each closure model compared to the experimental result ($Re = 2.44 \times 10^5$, $h/c = 0.3125$)

Case	C_L	C_D	C_L Error	C_D Error
Experiment	3.471	0.3886	-	-
$k - k_L - \omega$	3.336	0.3881	-0.14%	-3.12%
$k - \omega$ SST	2.985	0.3842	16.28%	-1.17%

predicts downforce again. Observation of the surface-flow structures given by the two models in Figure 6.29, for which the transitional model has been validated in the previous section, again shows the elimination of the separation bubble and occurrence of trailing-edge separation with $k - \omega$ SST. The large region of separated flow at the central region of the main-plane clearly contributes to the under-prediction of downforce. The good estimate of drag is, therefore, likely to be due to the over-prediction of skin friction drag from the fully-turbulent boundary layer compensating for the loss of pressure drag due to reduced circulation. The flow directions on the flaps are almost identical to those of the transitional model, the lack of the separation bubble is the only difference.

Comparisons of the flow-field around the wing at the flap centreline ($z/s = 2/3$) in terms of static pressure, total pressure and turbulent kinetic energy are given in Figure 6.30. The under-prediction of downforce by $k - \omega$ SST is shown in Figure 6.30a to be primarily due to it generating insufficient suction underneath the main-plane. There are only small differences in the C_p values on the upper surface, two of which arise at the locations of small separation bubbles. The primary difference occurs underneath the wing, where the magnitude of ΔC_p is three times higher than on the upper surface. This is a consequence of the separation bubble increasing the effective camber of the $k - k_L - \omega$ case which, as discussed previously, acts to increase circulation. The positive values of ΔP_0 close to the trailing edge in Figure 6.30c show the thicker boundary layer and trailing-edge separation that occur in the $k - \omega$ SST simulation, which acts to reduce circulation. It is clear at

the trailing edge of the flap that there are significant differences in the thickness of the wake. Unlike the lower surfaces, the upper surfaces give similar total pressure losses; this is a result of the large flow acceleration through the slot gap and the lack of large adverse pressure gradients on these surfaces.

Differences in turbulent kinetic energy is given in Figure 6.30b. The positive values of Δk show areas where $k - k_L - \omega$ has higher turbulent kinetic energy than $k - \omega_{SST}$. Negative values of Δk only occur in the front portion of each element, prior to the transition to the turbulent boundary layer for the $k - k_L - \omega$ model. The transitional model does not produce turbulent kinetic energy prior to the transition point, whereas the fully-turbulent model produces it from the leading edge, where the turbulent boundary layer begins. The flow-field is dominated by positive values of Δk in the aft-portion of the main-plane and entire flap, meaning that $k - k_L - \omega$ gives a wake with higher turbulence, but of higher total pressure. Given that the results of the $k - k_L - \omega$ model have been shown to be much more representative of the experimental results than the fully-turbulent model in terms of force prediction and surface-flow structures, it can be inferred that the flow-field is also much closer. Therefore, it can be seen that using a fully-turbulent closure model causes turbulent kinetic energy to be under-predicted and total pressure loss over-predicted in the wake.

6.3 F1 Wing

6.3.1 Force Measurements

The F1 wing is the most complex of all the configurations tested in terms of both geometry and flow structures; additionally, it produces the most downforce of all the wings. Downforce and drag curves for the F1 wing at varying ground clearance and Reynolds number

are given in Figure 6.31. The shape of the downforce curve is similar to that of the previous wing configurations, in that downforce increases as ground clearance reduces until a critical point, after which it enters the force-reduction region. Unlike the previously-tested wing configurations, however, the F1 wing shows minimal Reynolds-number sensitivity at all tested ground clearances. As a racing car front wing must work at a large range of Reynolds numbers, due to cornering speed differences of over 220 kph, this aspect is most likely by design. The drag curves also show little Reynolds-number sensitivity; given the lack of sensitivity in downforce coefficient it must be assumed that the difference in drag is not due to either pressure drag or induced drag changes, but primarily due to skin friction. Skin friction drag would decrease due to the higher velocity flow that was used in the higher Reynolds number test cases, whereas induced drag is reliant on the downforce coefficient, which, as shown, does not change with Reynolds number. The pressure drag will change slightly, however, due to the thickness of the boundary layer altering the effective shape of the wing. The shape of the drag curve is very different to that observed for the CSW configurations, for which drag continually increased with reducing ground clearance. For the F1 wing, the drag coefficient initially increases as ground clearance reduces, until $h/c \approx 0.55$, after which it then decreases. This reduction in drag coefficient occurs as the wing is still operating well within the force-enhancement region, and thus causes a dramatic increase in its aerodynamic efficiency. Given that F1 wings are designed to operate in a very specific ground clearance range it is likely that this drag reduction is in that region. Due to the complex design of the wing, the trends displayed in the force graphs could be due to a number of features; identifying such features is not part of the scope of this work and hence will not be explored further.

6.3.2 Surface-Flow Visualisation

Flow visualisation at the lowest and highest tested Reynolds numbers on the F1 wing's suction surfaces are given in Figure 6.32. As expected, because of the lack of Reynolds-number sensitivity in the force curves, there is little difference between the two Reynolds number cases. The separation line across the main-plane remains unchanged, hence it is likely that this represents the start of the adverse pressure gradient. Given the large amount of downforce produced by the wing, it is reasonable to expect that the adverse pressure gradient is very strong and thus the laminar boundary layer separates almost immediately, leading to the formation of a laminar separation bubble. The fact that the bubble begins at the same point at varying Reynolds numbers, is likely contributing to the lack of Reynolds-number sensitivity. The higher Reynolds number case clearly displays a shorter bubble, as would be expected, however this does not have a significant effect on the overall force production. This may be an indication that the location of the bubble is more significant than the length; this would make sense as it means that for all Reynolds numbers the same magnitude of suction is present in the region of constant pressure that characterises a separation bubble in the C_p distribution. Moreover, it is reasonable to assume that the bubble height is very small, as the local Reynolds number underneath the wing will be even higher than for the CSW-ME configuration, and thus the cambering effect is insignificant in terms of overall force production.

There is much more variation in the spanwise shape of the bubble that forms on the F1 wing in comparison to the CSW single- and multi-element configurations. This is because the CSW wings used a constant-section main-plane, whereas the aerofoil profile of the F1 wing varies across the span. At the centre of the main-plane the profile is a symmetrical 'neutral' aerofoil shape, and is quite similar in shape to the suction surface of the GA(W)-1; hence the bubble occurs at a similar location to the CSW wings. The profile ahead of

the flaps, however, is much more 'aggressive' and much more fore-loaded (apex closer to the leading edge), which causes the bubble to begin further upstream due to the adverse pressure gradient starting earlier. This shows that even on the aggressive aerofoil sections that F1 designers make use of, which includes two flap elements behind it, a separation bubble of significant size is still formed. Moreover, as the flow upstream of the bubble must be laminar, otherwise it would not form, there is clearly a large laminar boundary layer portion that exists on the wing.

Flow visualisation of the wing at constant Reynolds number for $h/c = 0.3125$ and $h/c = 0.125$ is shown in Figure 6.33. Although the size of the bubble does not appear to alter much in the central region, it is clearly shorter in the outer region at the lower ground clearance. At $h/c = 0.3125$ there is a clear separation on the first flap, but this does not occur at the lower ground clearance. Given that the bubble has been shown to increase effective camber and aid downforce production, this could be a contributor to the downforce loss. The main loss, however, appears to be the bursting of the tip vortices. An F1 wing relies heavily on vortex-induced suction to generate downforce, but it is clear from the inner-endplate flow visualisation that at the lower ground clearance the tip vortex has burst. Moreover, the footplate channel, which is designed to hold the tip vortex to let the wing operate closer to the ground, shows signs of separated flow at the lower ground clearance. This indicates that the flow moving around the endplate is not rolling into the tight, concentrated vortex it was at the higher ground clearance, thus reducing its force-enhancement capability.

Overall, the F1 wing has shown that even for highly-complex geometry and high-downforce capability wings, the laminar boundary layer and separation bubble still form significant aspects of the flow characteristics. Although the high downforce produced by the F1 wing means that the separation bubble itself may have only a small influence on the over-

all aerodynamic force being generated, it shows that the boundary layer is laminar for a significant portion of the wing.

6.4 Summary & Discussion

The results presented in this chapter have shown that the Reynolds-number sensitivity of a wing configuration corresponds to the amount of downforce that it produces. For the same Reynolds-number range of $1.63 \times 10^5 \leq Re \leq 2.44 \times 10^5$ the single-element wing displays up to 320% difference in the downforce coefficient;. Whilst this maximum difference does occur at the lowest test ground clearance, the minimum difference at any ground clearance is still 15% and increases almost exponentially as ground clearance reduces. In contrast, the multi-element and F1 wings display less than 2% difference in downforce coefficient across the majority of the test ground clearances, and have a maximum difference of less than 6% at the minimum ground clearance. The Reynolds-number sensitivity of drag coefficient shows a similar trend, with the single-element wing having a maximum and minimum difference of 35% and 28% respectively across the Reynolds-number range, but the multi-element and F1 wings displaying less than 3% in all instances.

Diwan & Ramesh [32] showed that a separation bubble's length, height and aspect ratio all reduce as Reynolds number increases. If these results hold true for the present work in terms of bubble height, as the flow-visualisation results in the present work have shown a shorter bubble occurring with increasing Reynolds number, then at higher Reynolds numbers the influence of the bubble on the wing's effective shape will be reduced. If a Reynolds number based on the local flow velocity under the wing, rather than that of the freestream air, were considered the multi-element and F1 wings would be operating at much higher Reynolds numbers than the single-element wing; this is evidenced by the

much higher suction generated to produce the higher downforce levels. Therefore, based on the argument that the bubble aspect ratio decreases at higher local Reynolds number, it is proposed that the bubble's influence on the effective shape of the wing is significantly reduced for the multi-element and F1 wings. This leads to the notion that the Reynolds-number sensitivity of the wing is linked to its downforce capability; in which the more downforce is produced, the less significant scale effects are. This is supported by the observation that the F1 wing produced the highest downforce of the wings tested and had the lowest Reynolds-number sensitivity. Although it cannot be conclusively shown in the present study, due to the lack of off-surface measurements, it is postulated that the increased upwash component in the flow from the higher-downforce-producing wings also aids in suppression of the bubble's influence, in addition to the higher flow velocity.

Despite the differences in Reynolds-number sensitivity, the wing configurations display the characteristic force curves that previous wing-in-ground-effect studies have shown, in which as ground clearance reduces the downforce and drag increases until a critical point, below which downforce reduces and drag continues to increase. The only exception to this was the F1 wing, for which drag reduces below a critical ground clearance; explanation of this phenomenon, however, requires further investigation with additional flow measurements beyond force acquisition and surface-flow visualisation.

Comparison of transitional ($k - k_L - \omega$) and fully-turbulent ($k - \omega$ SST) closure models for both the single-element and multi-element wing configurations demonstrated that the transitional model offered a much more representative solution compared to experimental results. Moreover, significantly different off-surface flow structures were observed between the two models. The $k - k_L - \omega$ predicted a similar surface-flow structure to that observed in the wind tunnel, where a laminar separation bubble at $x/c \approx 0.6$ was the defining feature. Moreover, for the multi-element wing the $k - k_L - \omega$ model was capable

of predicting downforce and drag coefficients to within 2.5% and 4% respectively. Given that the $k - k_L - \omega$ results closely matched the wind-tunnel experiments in terms of force production and surface-flow structure, it must be assumed that the off-surface flow-field given by $k - k_L - \omega$ will also be more accurate.

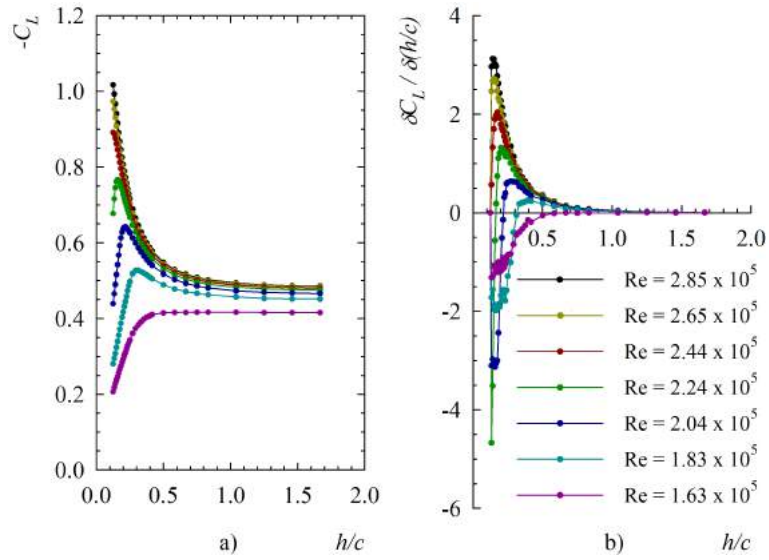


Figure 6.1: Curves of a) downforce and b) rate of change of downforce change with non-dimensional ground clearance at varying Reynolds number (Experimental, CSW-SE)

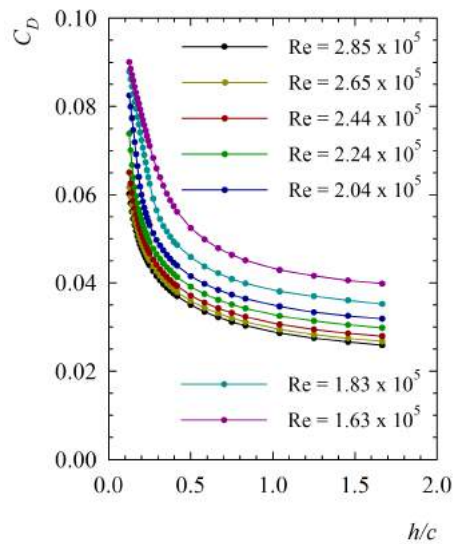


Figure 6.2: Variation of drag coefficient with ground clearance for varying Reynolds number (Experimental, CSW-SE)

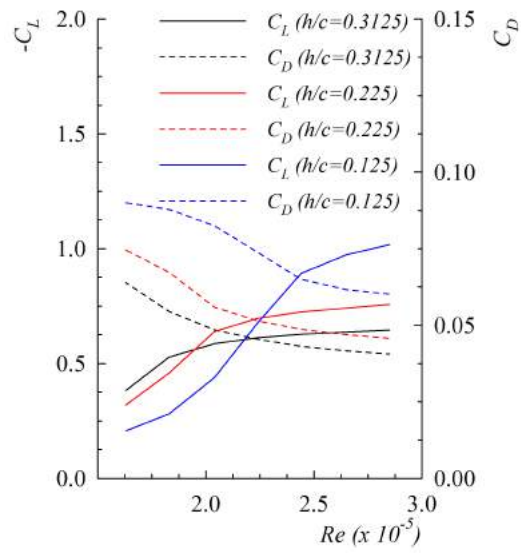


Figure 6.3: Reynolds-number dependency at constant ground clearances (Experimental, CSW-SE)

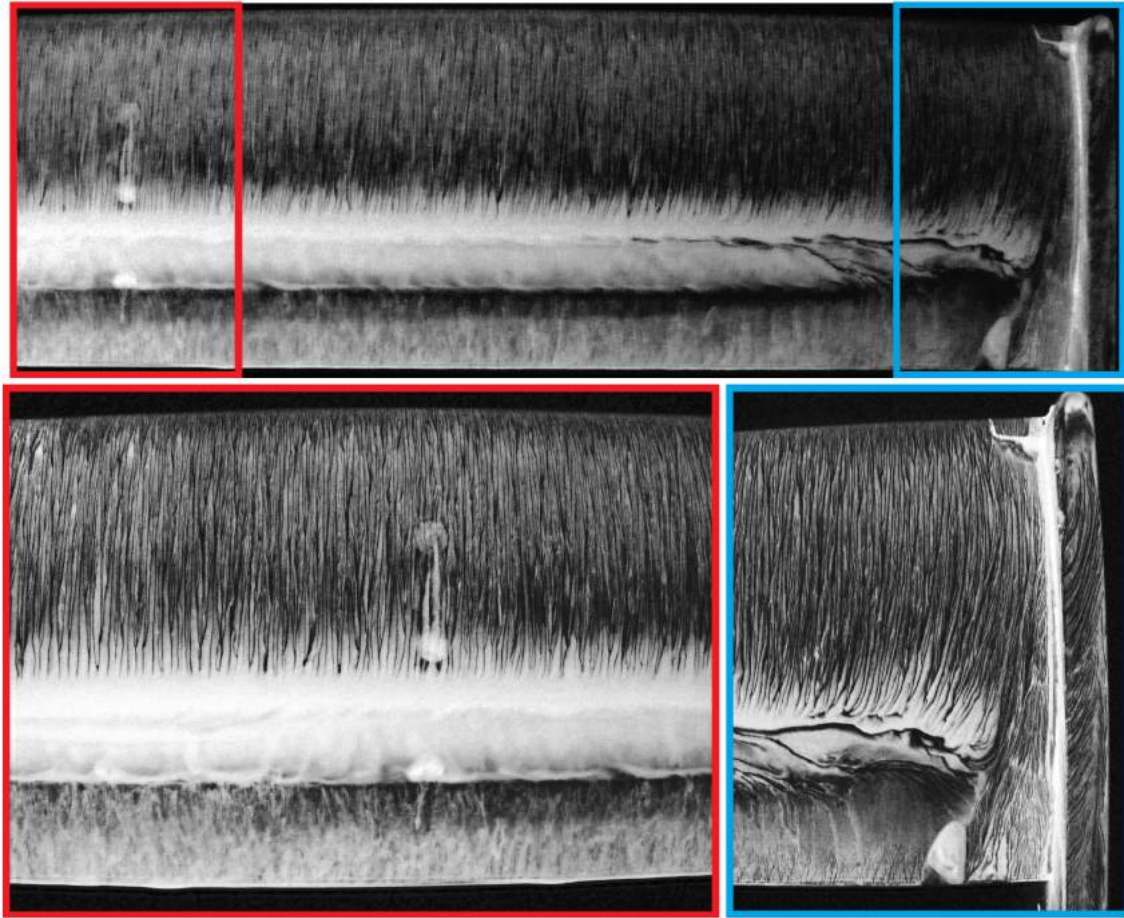


Figure 6.4: Suction-surface flow visualisation (Experimental, CSW-SE, $h/c = 0.3125$, $Re = 2.44 \times 10^5$, flow moving top to bottom)

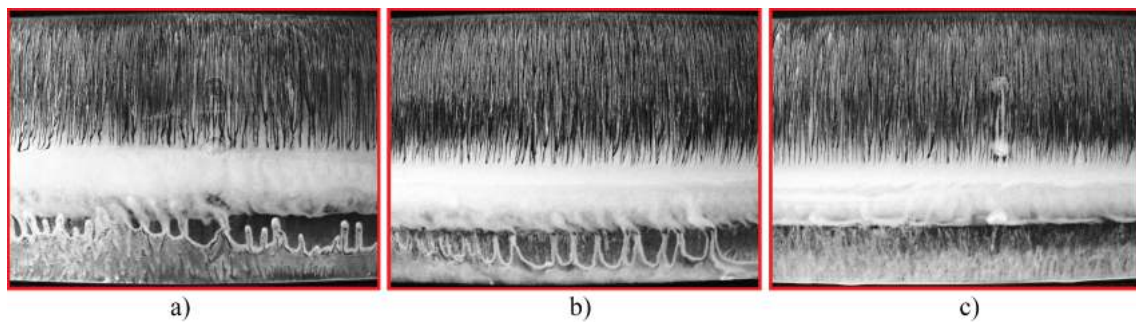


Figure 6.5: Suction-surface flow visualisation at Reynolds numbers a) 1.63×10^5 b) 2.04×10^5 and c) 2.44×10^5 (Experimental, CSW-SE, $h/c = 0.3125$, flow moving top to bottom)

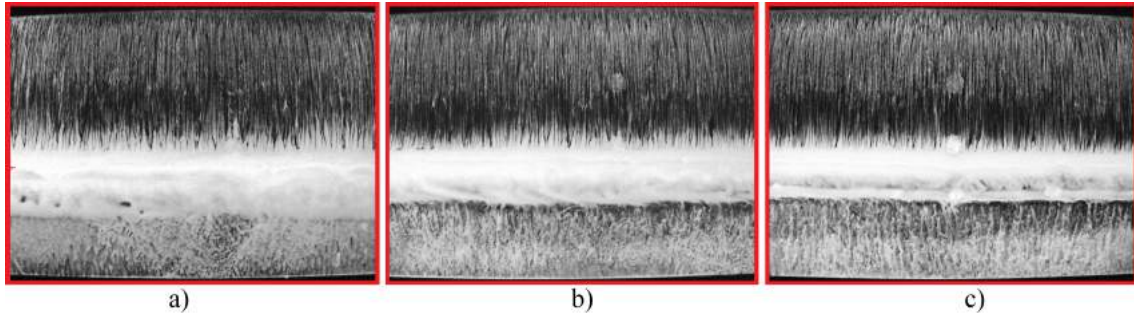


Figure 6.6: Suction-surface flow visualisation at Reynolds numbers a) 1.63×10^5 b) 2.04×10^5 and c) 2.44×10^5 (Experimental, CSW-SE, $h/c = 0.125$, flow moving top to bottom)

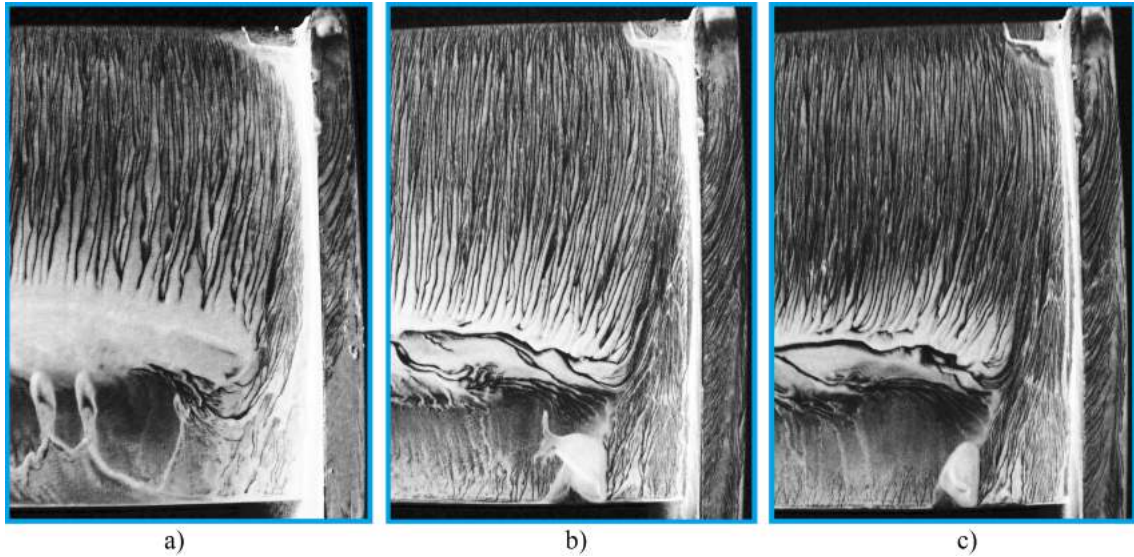


Figure 6.7: Suction-surface flow visualisation at the wing tip at Reynolds numbers a) 1.63×10^5 b) 2.04×10^5 and c) 2.44×10^5 (Experimental, CSW-SE, $h/c = 0.3125$, flow moving top to bottom)

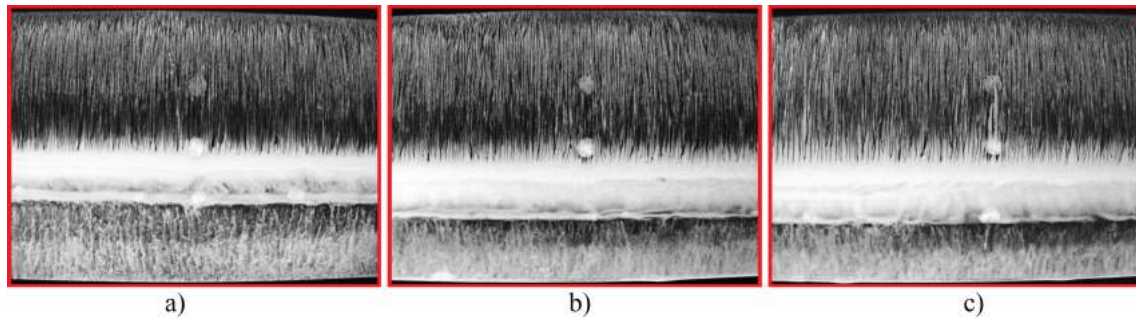


Figure 6.8: Suction-surface flow visualisation at normalised ground clearances of a) $h/c = 0.125$ b) $h/c = 0.225$ and c) $h/c = 0.3175$ (Experimental, CSW-SE, $Re = 2.44 \times 10^5$, flow moving top to bottom)

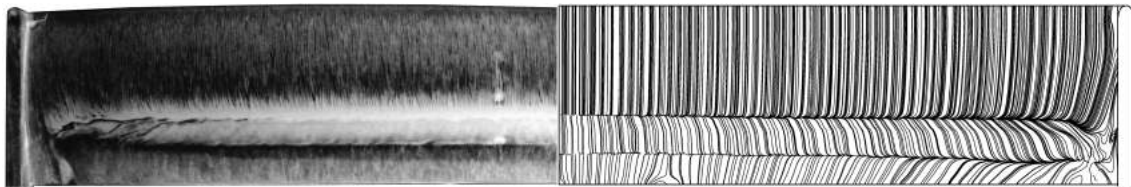


Figure 6.9: Streaklines on the suction surface of the single-element wing from a) wind-tunnel testing and b) RANS $k - k_L - \omega$ CFD (CSW-SE, $h/c = 0.3125$, $Re = 2.44 \times 10^5$, flow moving top to bottom)

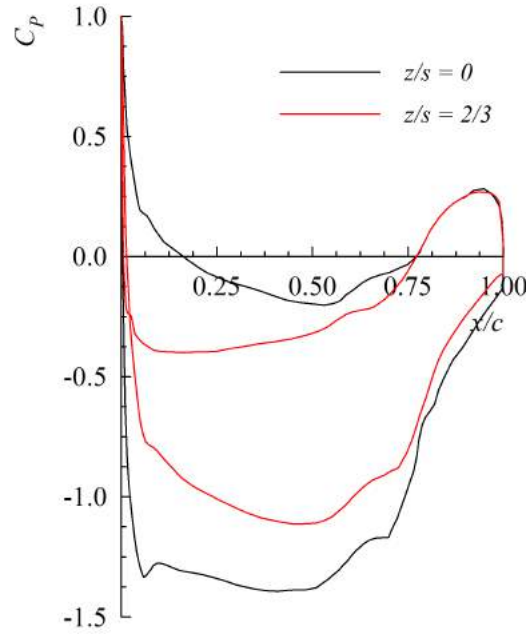


Figure 6.10: Static pressure distribution at centreline ($z/s = 0$) and part-semi span ($k - \omega$ RANS, CSW-SE, $h/c = 0.3125$, $Re = 2.44 \times 10^5$)

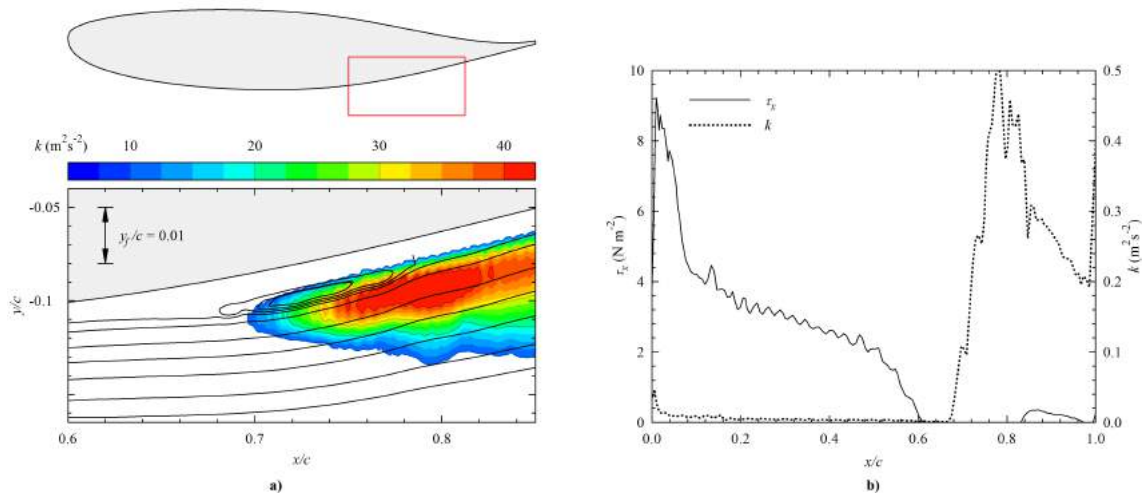


Figure 6.11: a) separation bubble shown by streamlines generated by velocity, and b) shear stress and turbulent kinetic energy on the lower surface ($z/s = 0$, $k - \omega$ RANS, CSW-SE, $h/c = 0.3125$, $Re = 2.44 \times 10^5$, flow field distorted for clarity¹)

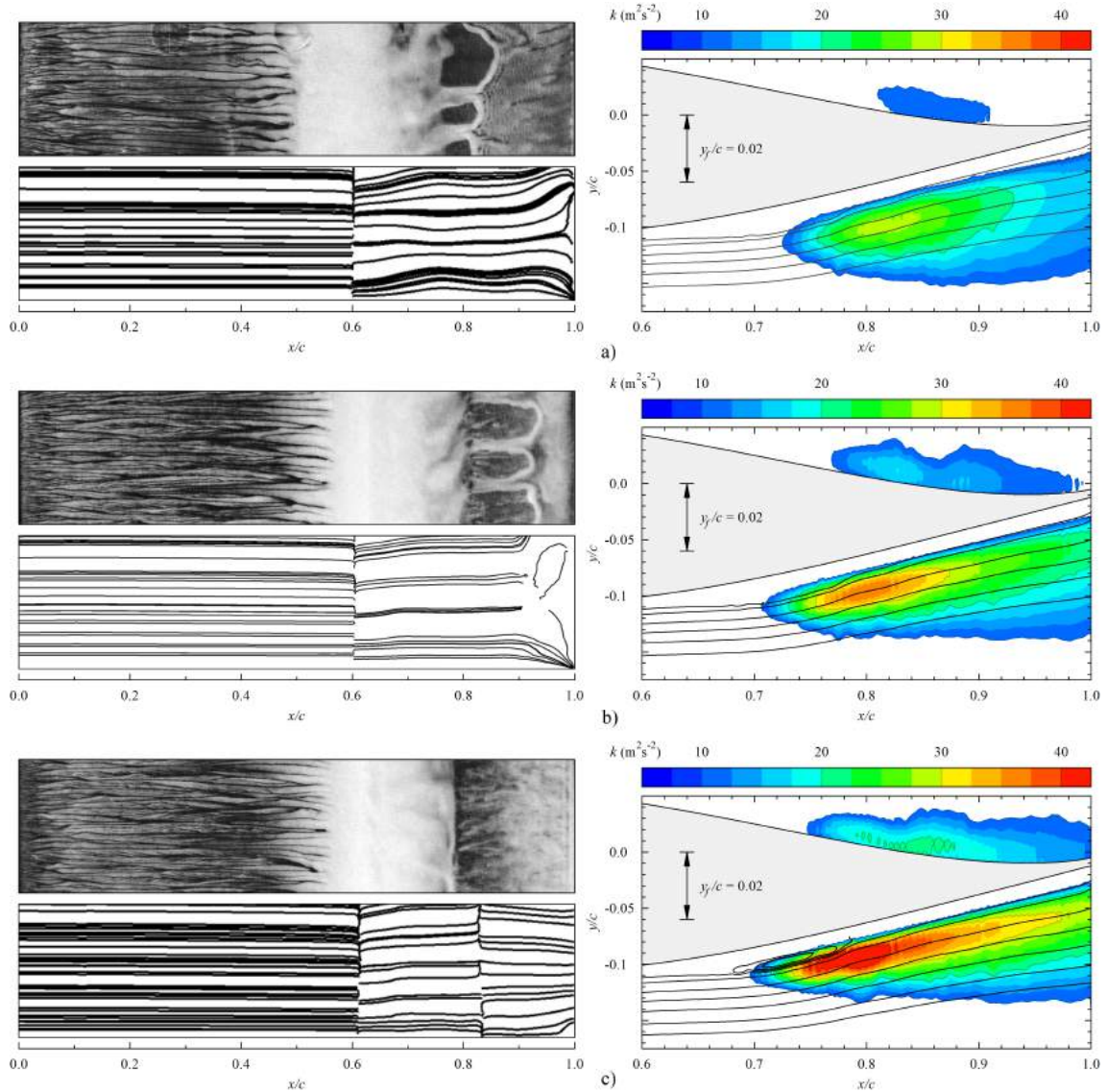


Figure 6.12: Comparison of experimental suction-surface flow visualisation (top left in each case) and computational surface streamlines generated by shear stress (bottom left in each case), and contours of turbulent kinetic energy with streamlines generated by velocity (right in each case) at a) $Re = 1.63 \times 10^5$, b) $Re = 2.04 \times 10^5$, and c) $Re = 2.44 \times 10^5$ ($k - k_L - \omega$ RANS, CSW-SE, $h/c = 0.3125$, flow moving left to right, cross-section of flow field distorted for clarity¹)

¹Note that for the contour plots in Figures 6.11, 6.12 & 6.27 the flowfield (but not the body) has been distorted by stretching it in the y-direction by a factor of 3 to make the bubble easier to see. (Merely magnifying the entire axis would have exaggerated the slope of the wing; magnifying both axes would have made the plot unacceptably large.) Thus, the y/c coordinate applies to the body; the size of the y-increment in the fluid is indicated by the y_f/c dimension.

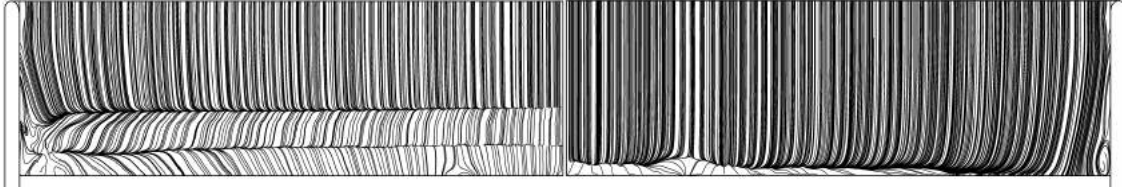


Figure 6.13: Comparison of surface streamlines generated by $k - k_L - \omega$ (left) and $k - \omega$ SST (right) (RANS, CSW-SE, $h/c = 0.3125$, $Re = 2.44 \times 10^5$, flow moving top to bottom)

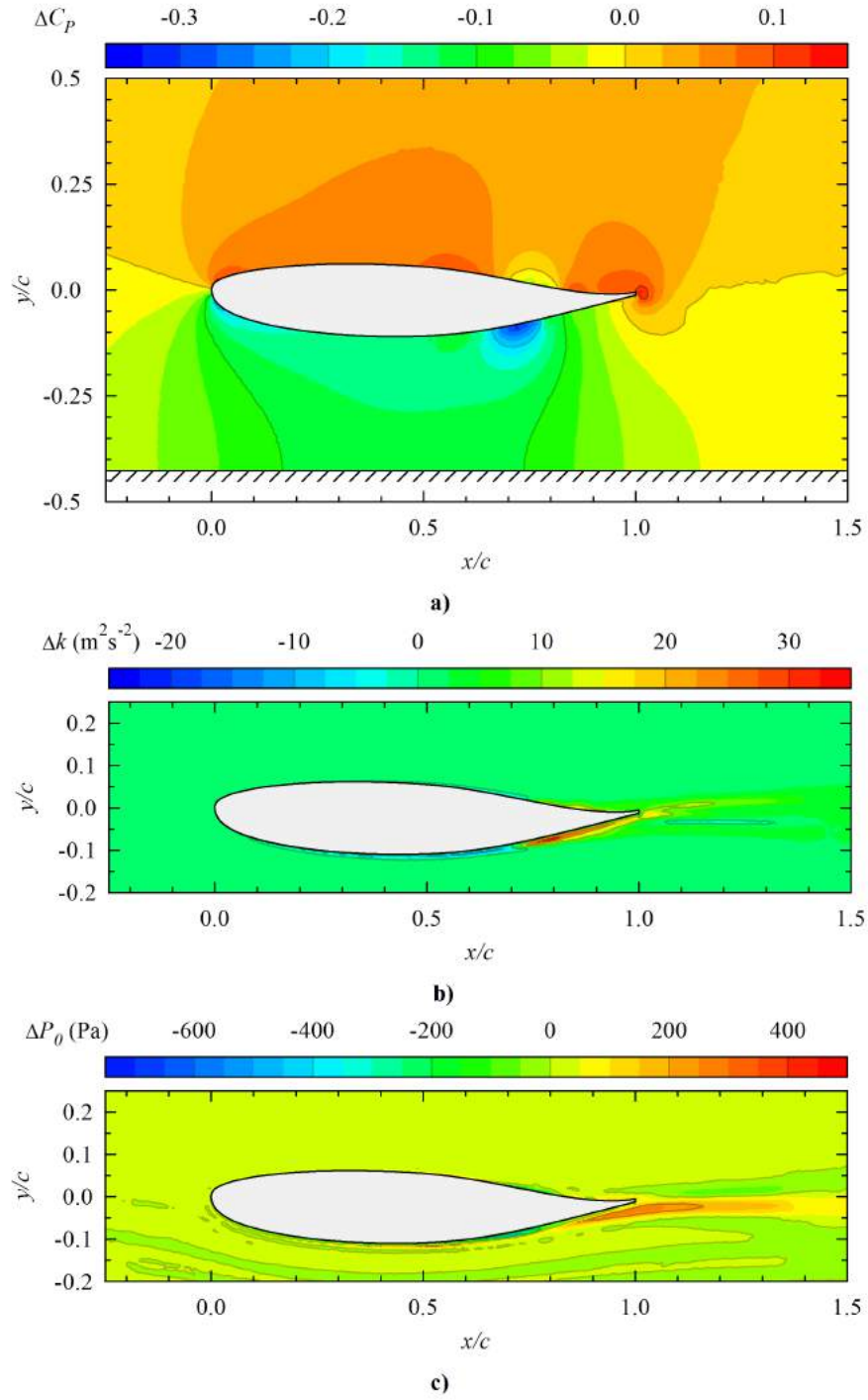


Figure 6.14: Comparison of closure models at main-plane centre line: contours of a) ΔC_p , b) ΔP_θ loss, and c) Δk (RANS, CSW-SE, $h/c = 0.3125$, $Re = 2.44 \times 10^5$)

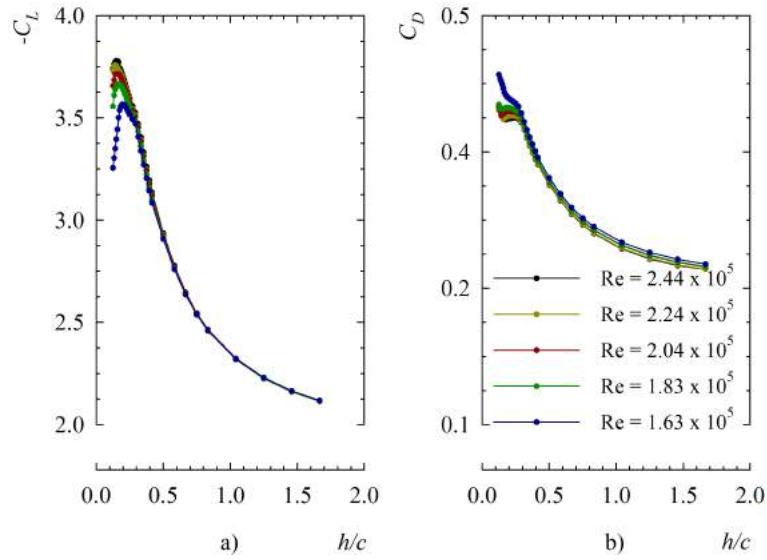


Figure 6.15: Variation of wing forces with ground clearance for varying Reynolds numbers a) downforce b) drag (Experimental, CSW-ME)

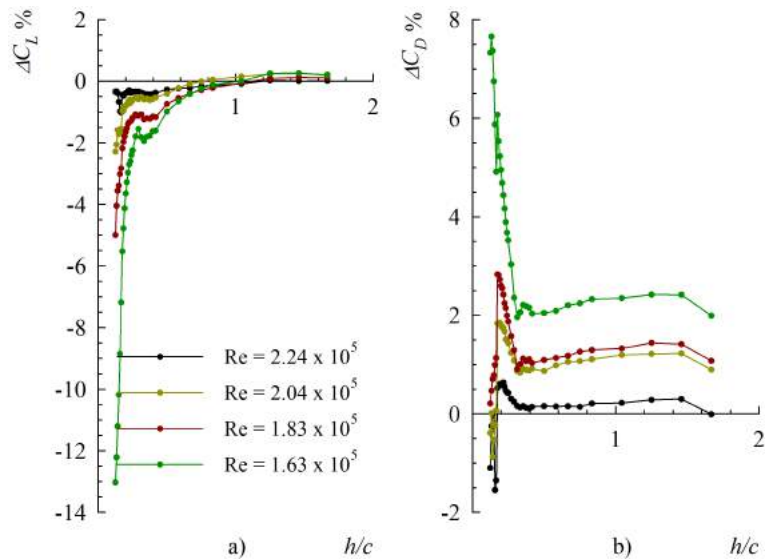


Figure 6.16: Percentage loss of a) downforce and b) drag of each Reynolds number compared to $Re = 2.44 \times 10^5$ (Experimental, CSW-ME)

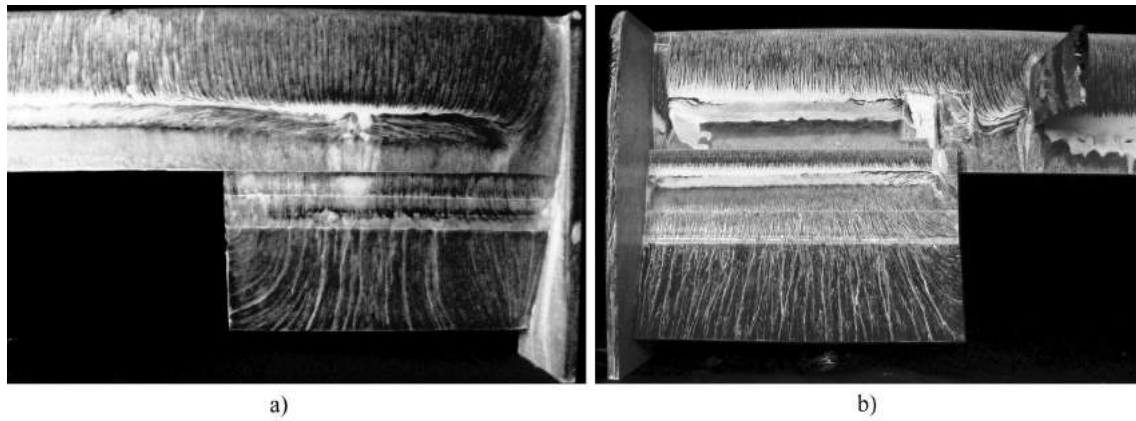


Figure 6.17: Flow visualisation of a) suction surfaces and b) pressure surfaces (Experimental, CSW-ME, $h/c = 0.3125$, $Re = 2.44 \times 10^5$, flow moving top to bottom)

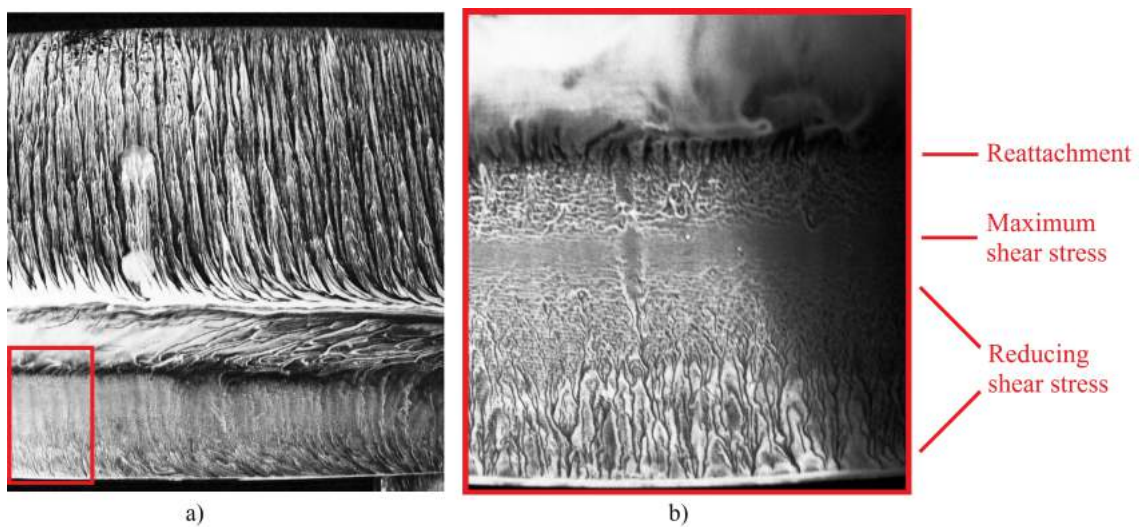


Figure 6.18: Suction-surface flow visualisation of a) flow regimes at main-plane centre and b) close-up of turbulent reattachment at main-plane centre (Experimental, CSW-ME, $h/c = 0.3125$, $Re = 2.44 \times 10^5$, flow moving top to bottom)

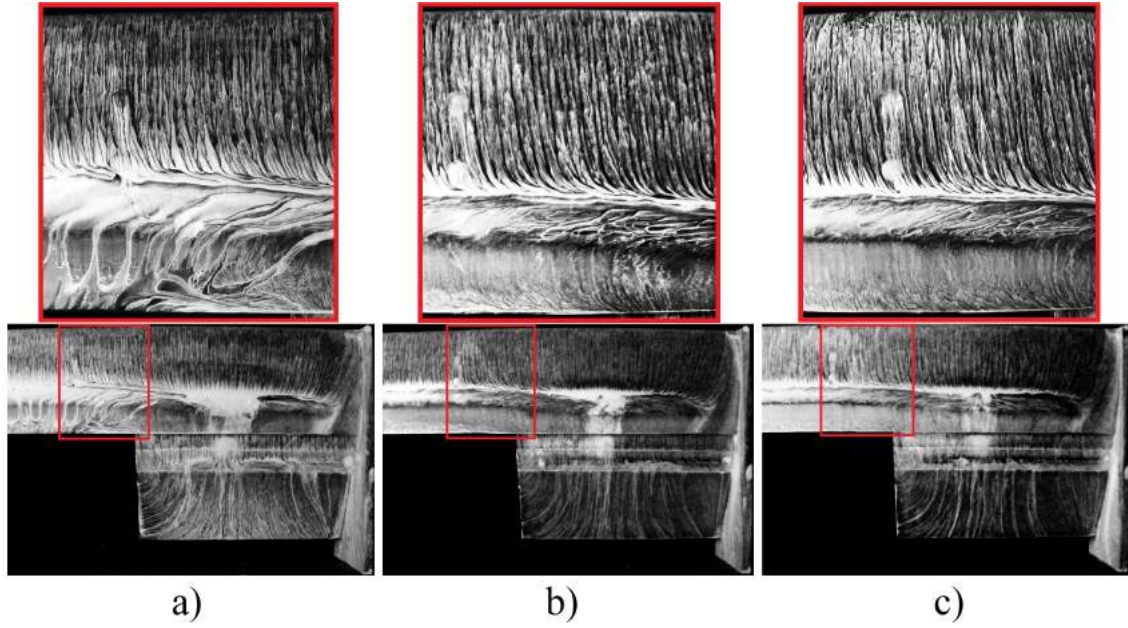


Figure 6.19: Suction-surface flow visualisation for $h/c = 0.3125$ at Reynolds numbers a) 1.63×10^5 , b) 2.04×10^5 , and c) 2.44×10^5 (Experimental, CSW-ME, flow moving top to bottom)

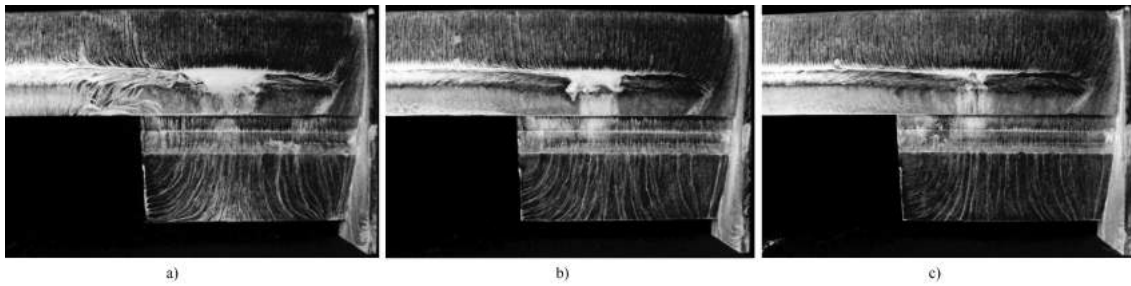


Figure 6.20: Suction-surface flow visualisation for $h/c = 0.225$ at Reynolds numbers a) 1.63×10^5 , b) 2.04×10^5 , and c) 2.44×10^5 (Experimental, CSW-ME, flow moving top to bottom)

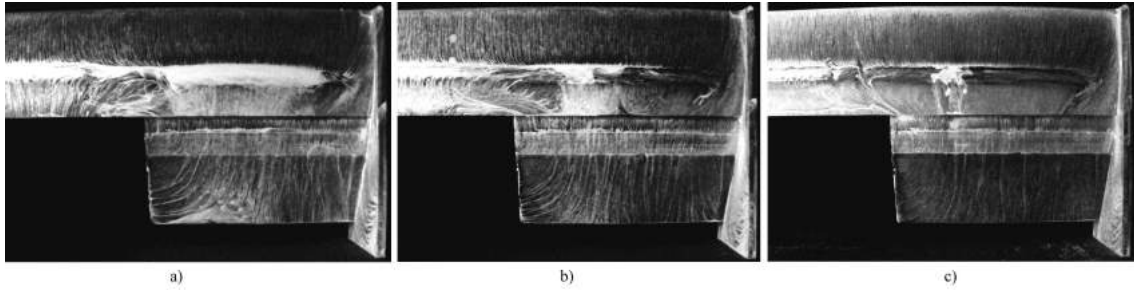


Figure 6.21: Suction-surface flow visualisation for $h/c = 0.125$ at Reynolds numbers a) 1.63×10^5 , b) 2.04×10^5 , and c) 2.44×10^5 (Experimental, CSW-ME, flow moving top to bottom)

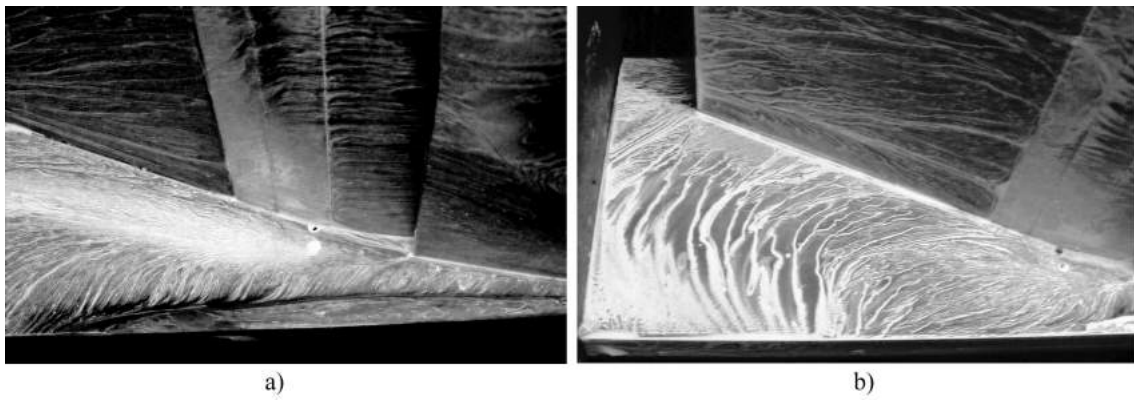


Figure 6.22: Flow visualisation of inner endplate surface at a) $h/c = 0.3125$, and b) $h/c = 0.125$ (Experimental, CSW-ME, $Re = 2.44 \times 10^5$, flow moving right to left)

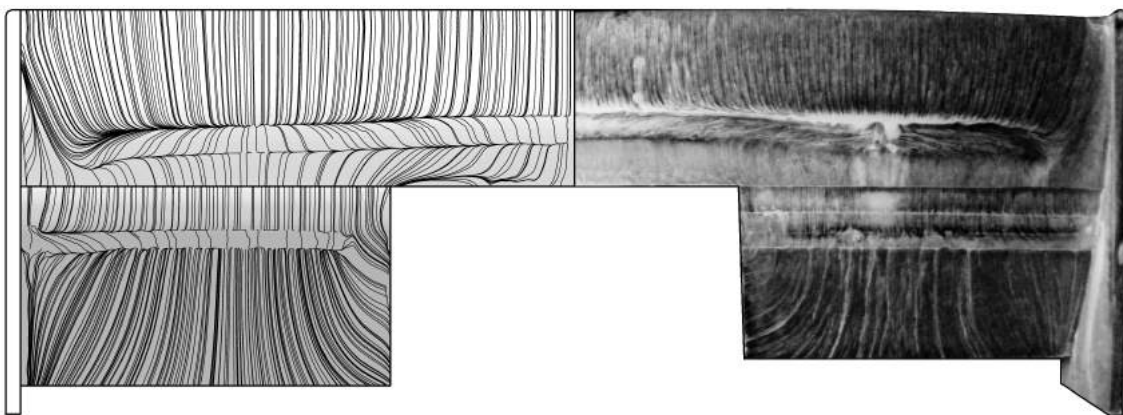


Figure 6.23: Comparison of experimental suction-surface flow visualisation and CFD surface streamlines on suction surface (CSW-ME, $Re = 2.44 \times 10^5$, $h/c = 0.3125$, flow moving top to bottom)

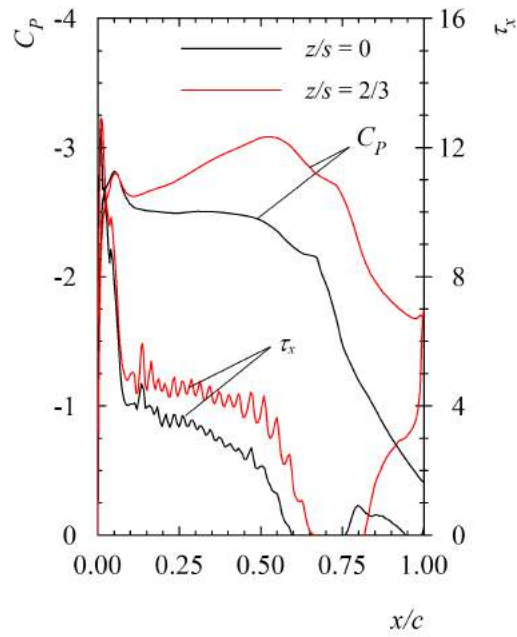


Figure 6.24: Static pressure and x-component of shear stress distributions on the suction surface of multi-element wing main-plane at $z/s = 0$ and $z/s = 2/3$ ($k - k_L - \omega$ RANS, CSW-ME, $h/c = 0.3125$, $Re = 2.44 \times 10^5$)

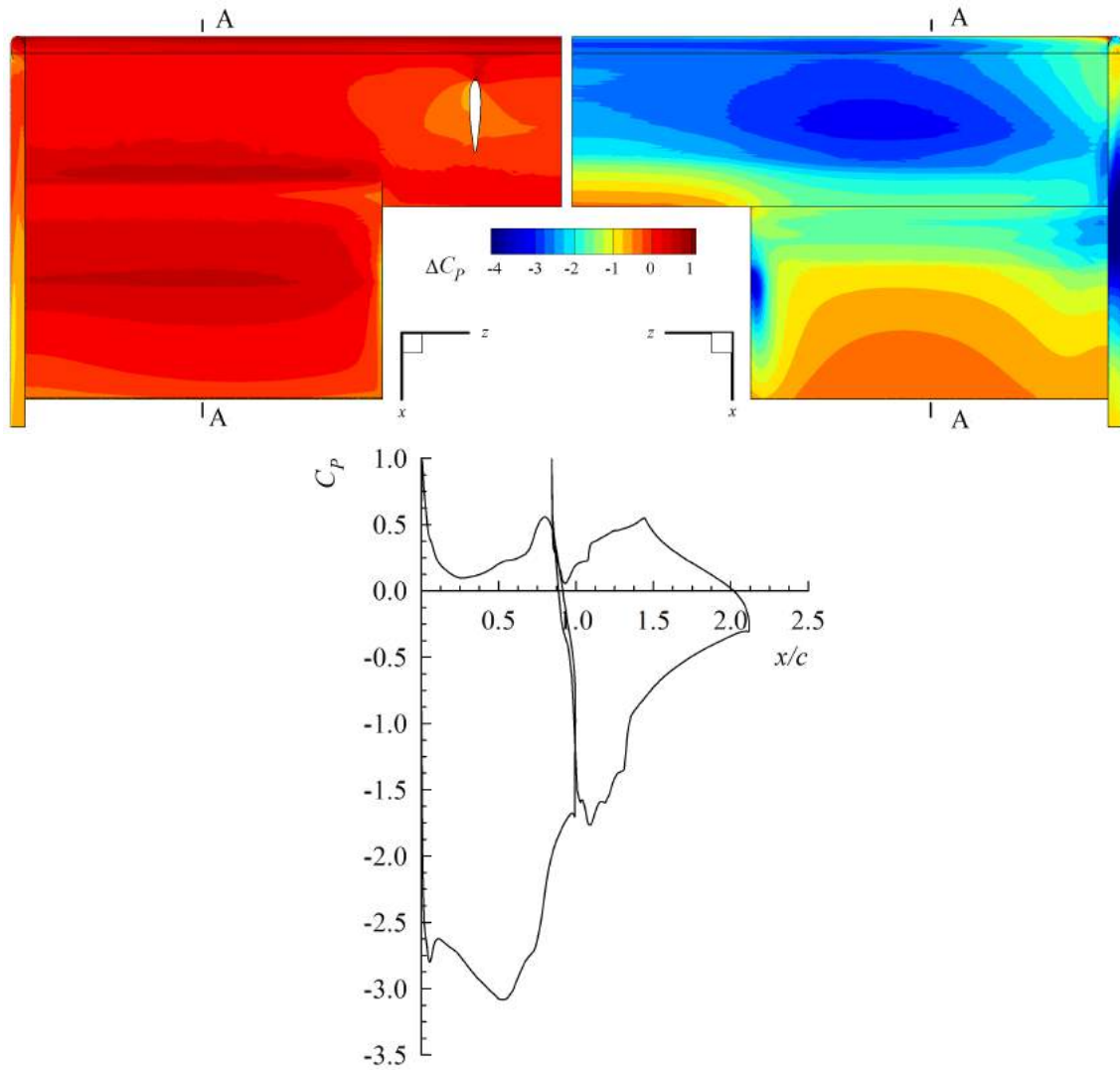


Figure 6.25: Contours of ΔC_p on the pressure (left) and suction (right) wing surfaces, and C_p distribution at $z/s = 2/3$ (plane A-A) ($k - k_L - \omega$ RANS, CSW-ME, $h/c = 0.3125$, $Re = 2.44 \times 10^5$)

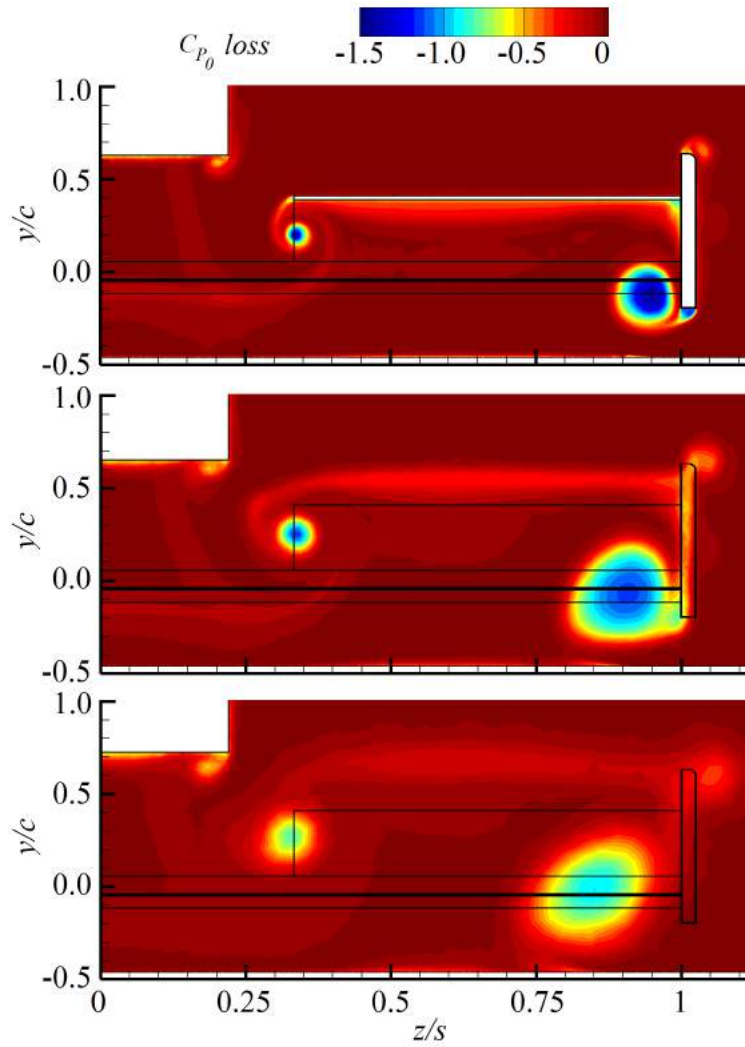


Figure 6.26: Contours of total-pressure coefficient loss in the flow-field behind the wing at a) $x/c = 2.25$, b) $x/c = 2.5$, and c) $x/c = 3$ ($k - k_L - \omega$ RANS, CSW-ME, $h/c = 0.3125$, $Re = 2.44 \times 10^5$)

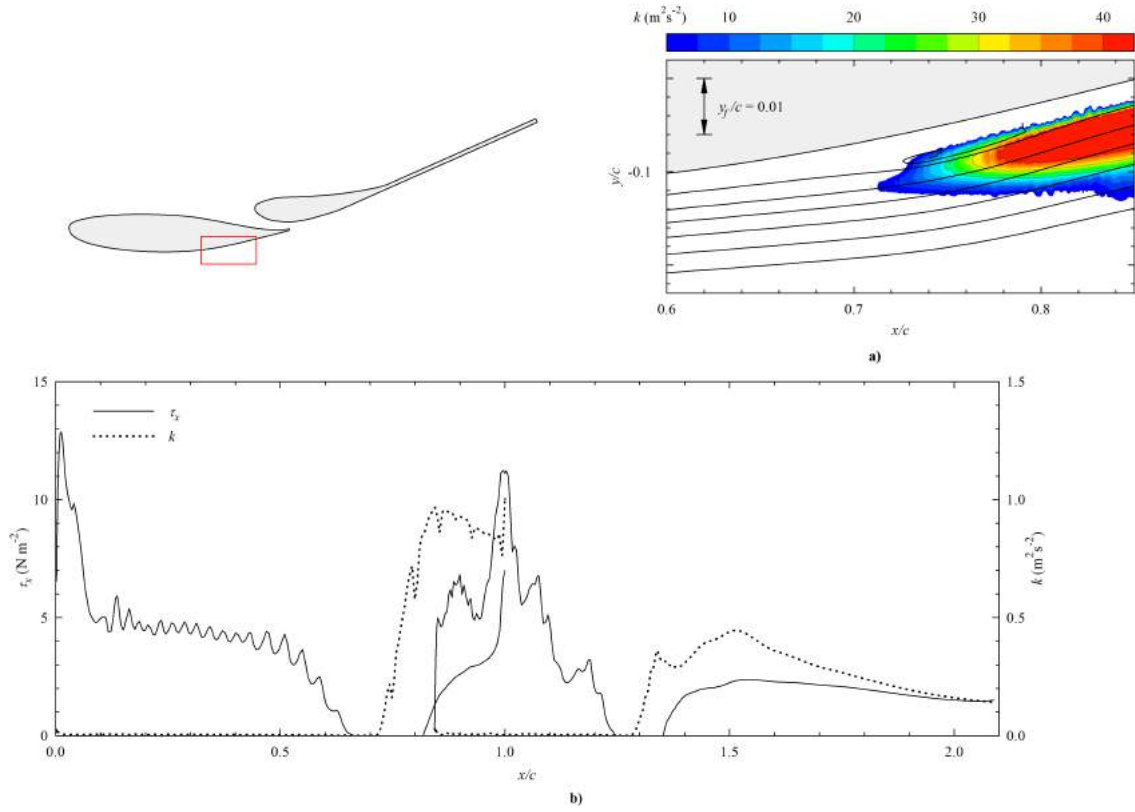


Figure 6.27: Contours of turbulent kinetic energy and streamlines of velocity at $z/s = 2/3$, and graph of x-component of shear stress and turbulent kinetic energy on suction surfaces ($k - k_L - \omega$ RANS, CSW-ME, $h/c = 0.3125$, $Re = 2.44 \times 10^5$, flow-field distorted for clarity - see footnote ¹ on p200)

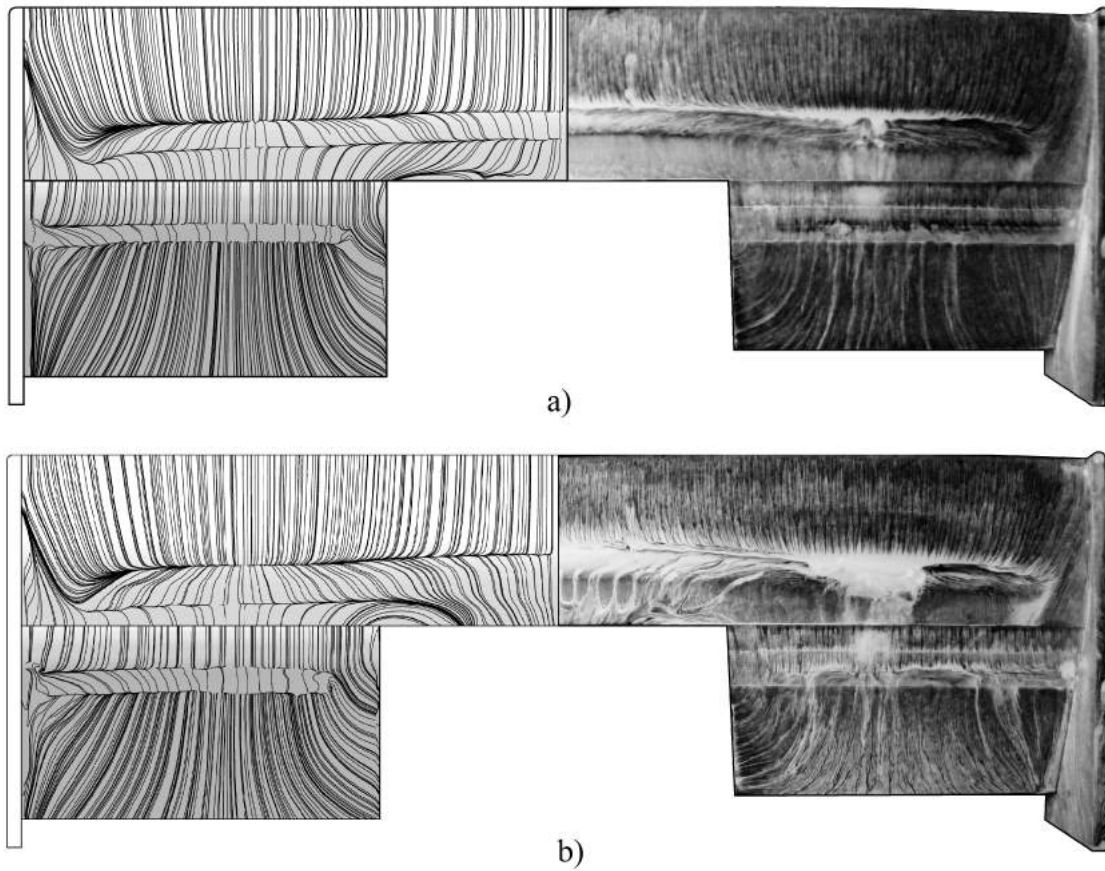


Figure 6.28: Comparison of experimental flow visualisation CFD surface streamlines at a) $Re = 2.44 \times 10^5$ and b) $Re = 1.63 \times 10^5$ ($k - k_L - \omega$ RANS, CSW-ME, $h/c = 0.3125$, flow moving top to bottom)

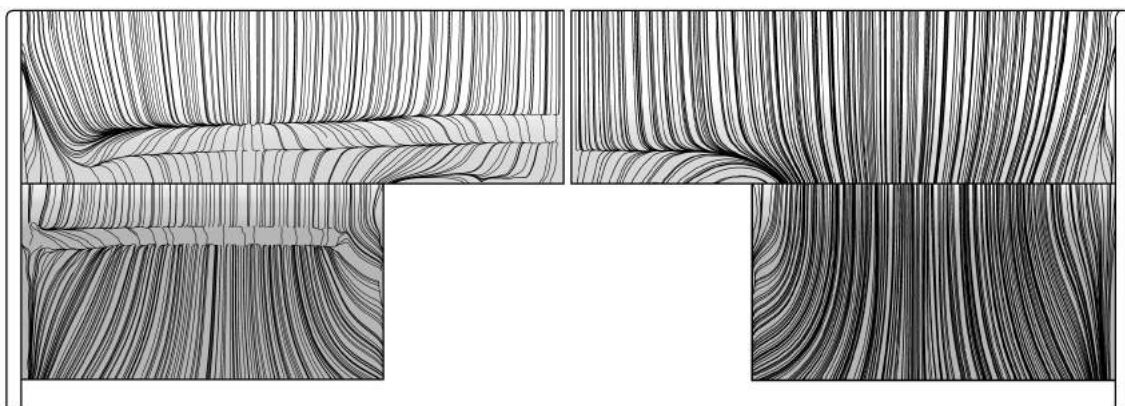


Figure 6.29: Comparison of surface streamlines (flow moving top to bottom) generated by $k - k_L - \omega$ (left) and $k - \omega$ SST (right)(RANS, CSW-ME, $Re = 2.44 \times 10^5$, $h/c = 0.3125$)

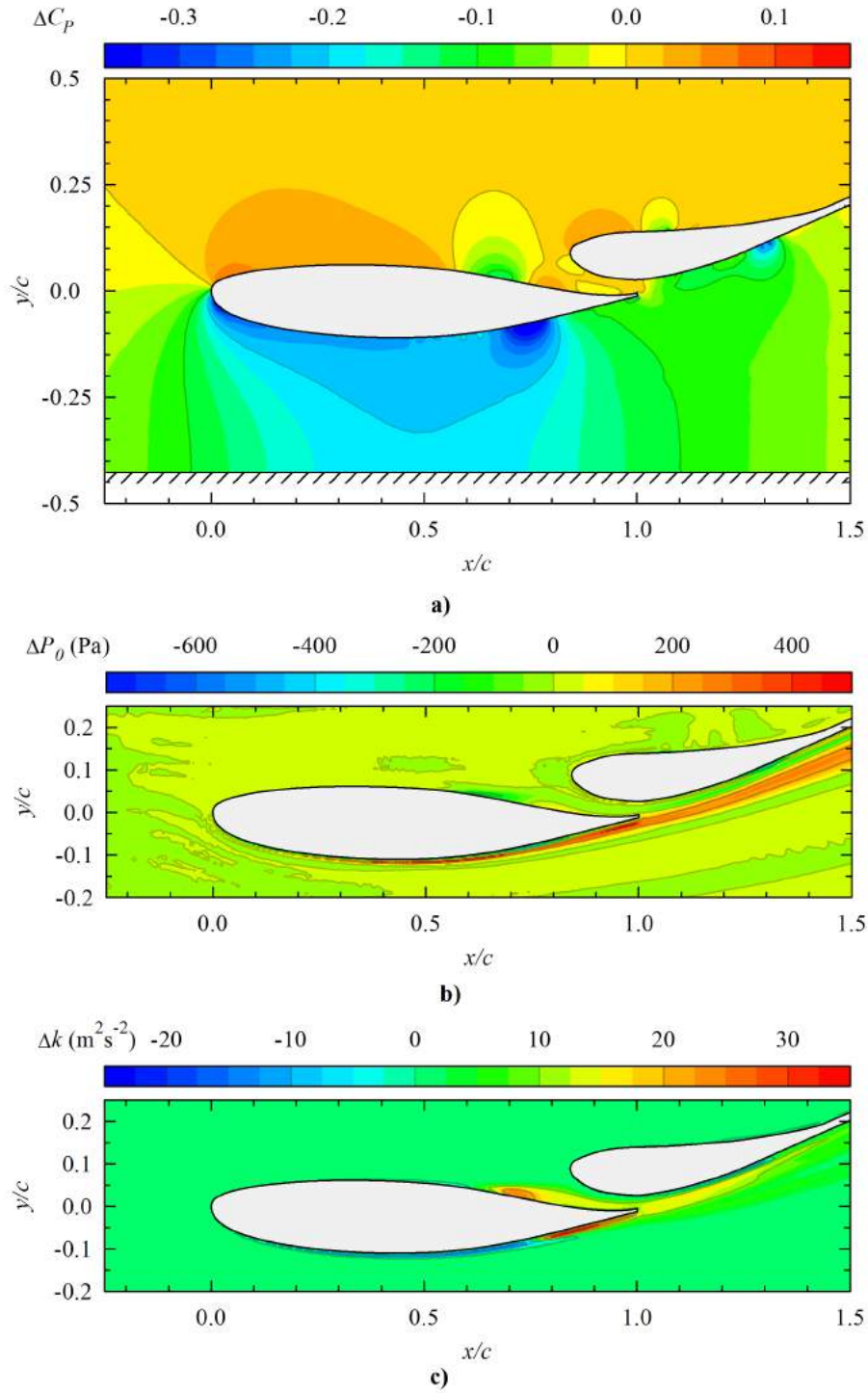


Figure 6.30: Comparison of closure models at main-plane centre line: contours of a) ΔC_p , b) ΔP_0 loss, and c) Δk (RANS $((k - k_L - \omega) - (k - \omega \text{ SST}))$, CSW-ME, $h/c = 0.3125$, $Re = 2.44 \times 10^5$)

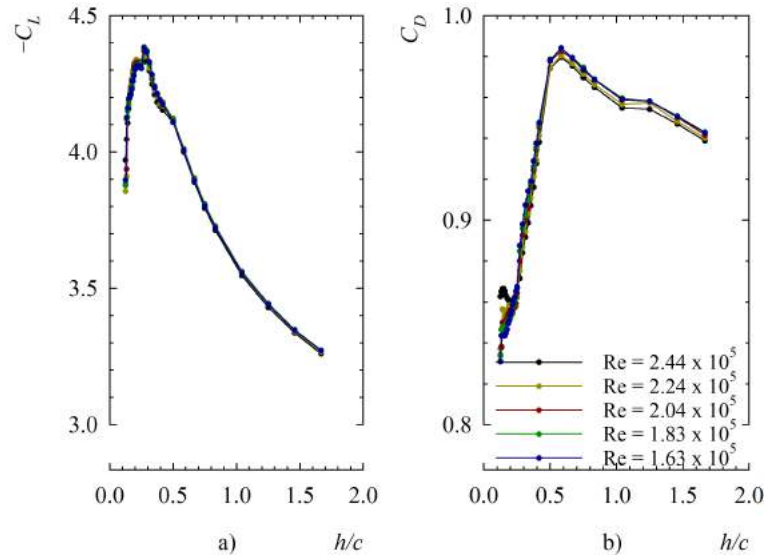


Figure 6.31: Variation of force coefficient with ground clearance for varying Reynolds number a) downforce b) drag (Experimental, F1 wing)

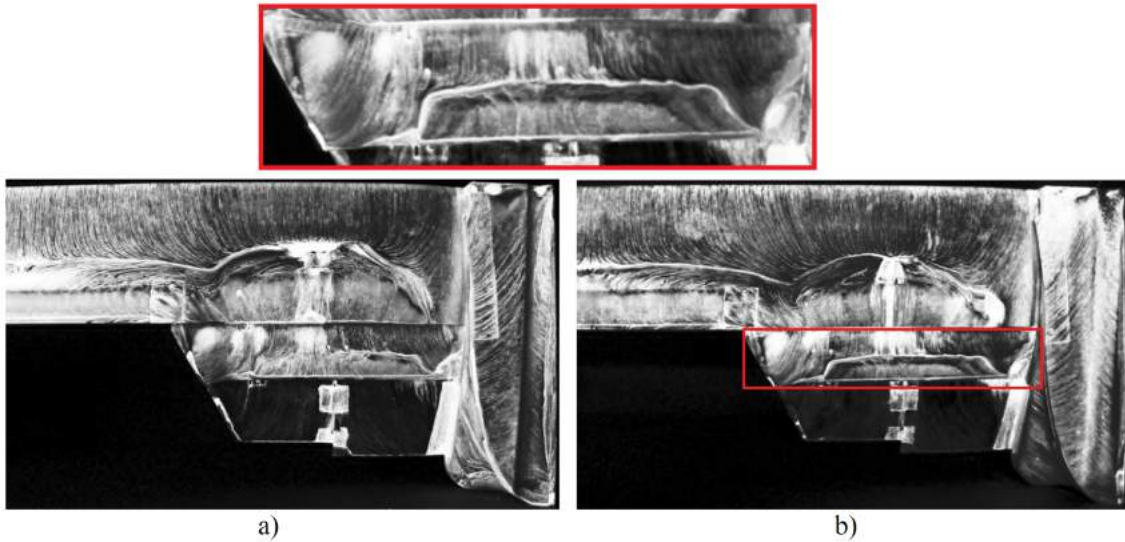


Figure 6.32: Suction-surface flow visualisation on F1 wing for varying Reynolds numbers a) $Re = 1.63 \times 10^5$ and b) $Re = 2.44 \times 10^5$ (Experimental, F1 wing, $h/c = 0.3125$, flow moving top to bottom)

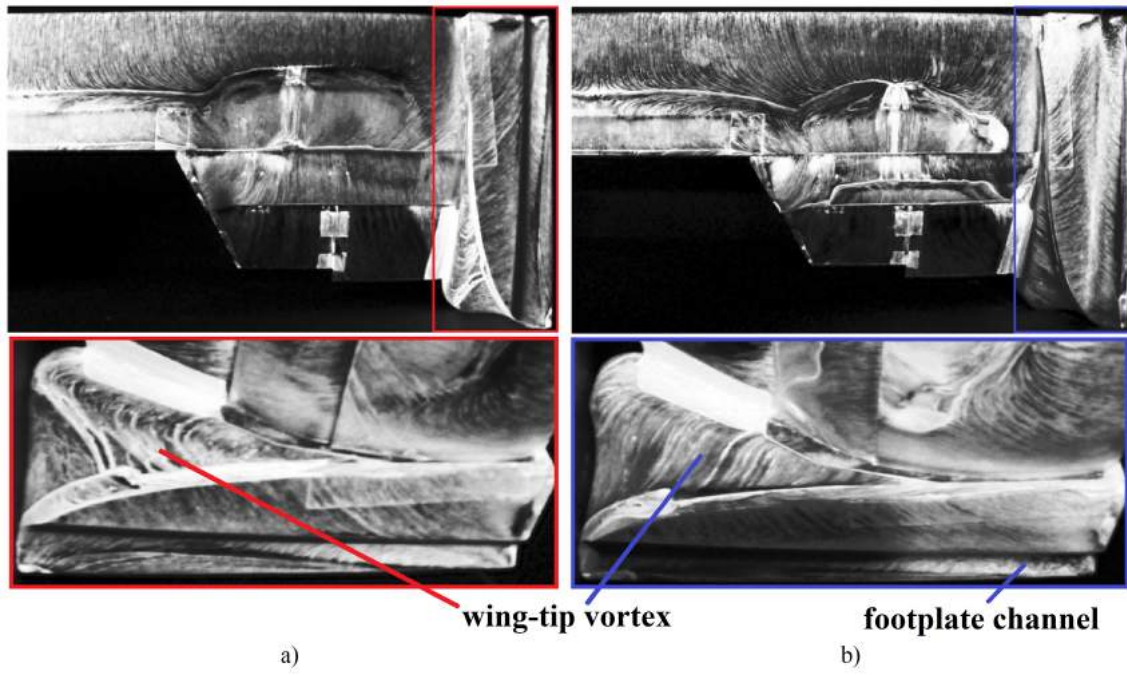


Figure 6.33: Flow visualisation of F1 wing suction-surfaces (top) and inner endplate (bottom) at a) $h/c = 0.125$ and b) $h/c = 0.3125$ (Experimental, $Re = 2.44 \times 10^5$, flow moving top to bottom)

Chapter 7

Results - Forced Transition

In the previous chapters the aerodynamic characteristics of two-dimensional and three-dimensional wings, of varying complexity and downforce capability, have been investigated under free-transition conditions. Comparisons between CFD computations using a transition-sensitive closure model, which includes both laminar, transitional and turbulent phenomena, and a fully-turbulent closure model in Sections 6.2.3 & 6.1.3 revealed markedly different results. This suggests that a wing that has a laminar boundary layer portion on its suction surface will act very differently to one that has only a turbulent boundary layer forming. As there will always be question marks over the validity of any CFD approach, since turbulence models are based on a very simplistic assumption, it is more appropriate to investigate the effect of the laminar boundary layer inclusion through wind tunnel testing. This is accomplished by placing a boundary-layer trip at specific chordwise locations in order to force the boundary layer to become turbulent. A trip is essentially any object, or method (such as vibration or noise), that destabilises the boundary layer to cause turbulence production.

The act of forcing the laminar boundary layer to prematurely transition to a turbulent state is a common technique in wind-tunnel testing to eliminate the later transition point caused by testing at reduced Reynolds numbers. Whether the higher Reynolds number flow characteristics can be replicated by a wing operating at a lower Reynolds number in forced transition will also be investigated.

In addition to allowing the importance of the laminar boundary layer on the aerodynamic characteristics of a wing in ground effect to be investigated, the forced-transition case is also representative of real-world motorsport applications. Correia et al. [108] showed that a GA(W)-1 wing operating in the wake of another car experienced a bypass transition at the wing's leading edge such that a turbulent boundary layer was formed across the entire surface of the wing. Forcing immediate transition to a turbulent boundary layer can be considered synonymous to a bypass transition, and thus is representative of a wing that is operating in high freestream turbulence. This does not necessarily have to be the wake of another car, as gusting winds and permanent trackside structures (e.g. walls, bridges, grandstands, etc.) may also be enough to provide sufficiently high freestream turbulence to cause bypass transition.

7.1 Two-Dimensional Wing

In this section, a roughness-type trip made from double-sided tape and grit 60 sand is used to cause boundary-layer transition. Both the height and roughness of the trip inside the boundary layer are intended to cause immediate transition to a turbulent state. It should be noted that the locations quoted in this section refer to the position of the trip's leading edge.

It was shown in Chapter 5 that laminar phenomena have a considerable effect on the aero-

dynamic characteristics of a wing in ground effect. In the tested Reynolds number range $2.85 \times 10^5 \leq Re \leq 1.17 \times 10^6$ the wing was shown to have a dependence on Reynolds number. Whilst this dependency did reduce as Reynolds number increased, it was still important. This dependency was observed to be due, at least in part, to laminar and transitional phenomena. As discussed in the review of relevant literature (Chapter 2), the majority of computational studies into wings in ground effect made use of RANS solvers with fully-turbulent closure models and, as such, laminar effects were not included. Despite the low Reynolds number operating conditions. Moreover, computations by Zerihan [88] and Doig et al. [93] both utilised forced-transition experimental results when making comparisons to their fully-turbulent-RANS simulations, thus showing that laminar effects were clearly manifesting in the experiments. No investigation into the importance of laminar phenomena on the performance of a wing in ground effect has been carried out hitherto. In this section the boundary layer on the suction surface of the wing is forced to transition to a turbulent state, using a roughness-type boundary-layer trip that artificially introduces perturbations into the flow to promote turbulence production, at specific chord-wise locations. This will allow not only the fully-turbulent boundary layer assumption to be evaluated but also the influence of the laminar boundary layer of varying extent and transition mechanism on the performance of the wing to be observed.

7.1.1 Pressure Measurements

Static pressure distributions of the suction surface of the wing for $0^\circ \leq \alpha \leq 5^\circ$ are given in Figure 7.1. As discussed in Chapter 5, a laminar separation bubble exists for the lower incidence free-transition cases. By forcing transition with trips whose leading edges are placed at $x/c = 0.1$, $x/c = 0.3$ and $x/c = 0.5$ the separation bubble is eliminated due to the boundary layer already being turbulent when it encounters the adverse pressure

gradient. The placement of the trip meant that some tappings were blocked by it; the readings from the affected tappings have been removed from the results in Figure 7.1. It can be observed that the trip still has an effect on the tappings it did not directly block. This is a result of the physical size of the trip, which causes a stagnation on its leading edge, and in some cases separation at the trailing edge. This slows the flow slightly such that both tappings immediately before and after the trip exhibit slightly lower suction than the general trend of the distribution would otherwise suggest. This was somewhat unavoidable, however, because it was not possible to alter the surface roughness at varying points without attaching an object on the wing's surface.

In all cases the boundary layer is tripped prior to the free-transition laminar separation point ($x/c \approx 0.6$), and thus there is no separation bubble. As the bubble is a force-enhancement mechanism, this leads to the magnitude of suction reducing for the forced-transition cases. When transition is forced at $x/c = 0.1$ the turbulent boundary layer covers the majority of the wing and the magnitude of suction is dramatically reduced across the entire chord. At $x/c = 0.45$, close to the point of maximum suction, the magnitude of suction has reduced by 10.5%. Moving the trip downstream to $x/c = 0.3$ leads to a pressure distribution that is similar to that of the $x/c = 0.1$ trip case, however the magnitude of suction is slightly increased. The same trend holds true by moving the trip to the most downstream location of $x/c = 0.5$, at which the magnitude of suction is very similar to that of the free-transition case, the only difference being the lack of the constant-pressure region due to the removal of the separation bubble. This shows that the magnitude of suction, and by extension, the downforce produced by the wing are proportional to the lengths of the laminar and turbulent boundary layer portions.

As incidence is increased from $\alpha = 0^\circ$ to $\alpha = 5^\circ$, the trend remains that the more upstream the position of the trip the lower is the suction, suggesting that downforce is reducing.

Moreover, the effect of forcing transition appears to be amplified as incidence increases, i.e. the difference between the free- and forced-transition cases becomes greater. As the increase of incidence is analogous to an increase in downforce, it could be perceived that the more downforce the wing produces the greater the effect of forced transition.

The Reynolds-number sensitivity of the free- and forced-transition wings are shown in Figure 7.2. As discussed in Chapter 5, as the Reynolds number increases the magnitude of suction also increases and the separation bubble becomes smaller. When transition is forced, however, the Reynolds-number sensitivity is significantly diminished. This shows that it must be the laminar portion and/or separation bubble that is causing the Reynolds-number sensitivity rather than the turbulent portion.

7.1.2 Surface-Flow Visualisation

Suction-surface flow visualisation for varying incidence at $Re = 6 \times 10^5$ is shown in Figure 7.3. The laminar separation bubble discussed in previous sections is again observed at $\alpha = 0^\circ$ and $\alpha = 3^\circ$ in free transition. The result of forcing the boundary layer to become prematurely turbulent, however, is that the separation bubble is eliminated. This is because the turbulent boundary layer is more resilient to deceleration and thus does not separate at the start of the adverse pressure gradient as the laminar boundary layer does. Closer to the trailing edge, however, the turbulent boundary layer separates as the momentum loss due to increased shear stress leads to it having insufficient energy to continue travelling through the adverse pressure gradient. This results in a loss of circulation and reduced downforce. It can be observed that as the trip is moved upstream, so does the trailing-edge separation point. This leads to the notion that once the turbulent boundary layer begins it can only travel so far against the adverse pressure gradient, hence the more upstream the trip, the more upstream the separation. Moreover, as the turbulent boundary

layer in the free-transition case starts much further downstream, because of the separation bubble, it has sufficient energy to stay attached until the trailing edge. The separation bubble can, therefore, be seen as a mechanism that helps the boundary layer remain attached by re-energising the boundary layer part-way through the pressure recovery. As no separation bubble was observed at $\alpha = 5^\circ$ in free-transition, but a large trailing-edge separation region occurs in forced-transition, it demonstrates that the momentum loss in the turbulent boundary layer is a more important aspect than the re-energising effect of the separation bubble.

The static pressure distribution presented in the previous section showed that the reduction in suction of the forced-transition cases becomes greater as incidence is increased. Similarly, the flow visualisation shows an increasingly earlier separation point as incidence increases. Forcing transition at $x/c = 0.5$ for $\alpha = 5^\circ$ showed an almost identical pressure distribution to the free-transition case; which is a result of the laminar and turbulent boundary-layer portions being of equal length in both the free- and forced-transition cases.

7.1.3 Hot-Film Anemometry

The quasi-shear stress and CV distributions shown in Figures 7.4 & 7.5 allow the boundary-layer separation, transition and reattachment points to be investigated. The spatial resolution of the hot-film gauges is insufficient to give precise locations of these phenomena, however it is adequate to understand the flow characteristics of the different cases. As discussed in the previous section, the region of zero-shear stress in the free-transition case shows the presence of the separation bubble. It is clear by the lack of zero-shear stress regions ahead of the trailing-edge that the separation bubble has been completely removed in the forced-transition cases, as the flow visualisation demonstrated. The shear

stress distribution essentially reinforces the characteristics identified by the flow visualisation, as the paint's movement is determined by the surface shear stress. The shear stress, however, also demonstrates the momentum loss in the turbulent boundary layer that was discussed previously, as at a given location the shear stress is lower for the upstream trip ($x/c = 0.1$). The decrease in shear stress with increasing chordwise position is indicative of the momentum loss in the turbulent flow as the boundary layer becomes thicker, whilst the increase in CV shows that turbulence is increasing. This is because the adverse pressure gradient promotes instability that leads to turbulence production, hence CV increases. The integral of the shear stress distribution represents the skin-friction drag, hence Figure 7.4 indicates that the forced-transition cases have a higher skin-friction drag.

As incidence increases, the difference between the two trip locations becomes larger, showing that the influence of forced transition becomes more prominent as the downforce level is increased. When the shear stress becomes zero near to the trailing edge, as is observed for the $x/c = 0.1$ trip at $\alpha = 3^\circ$ and $\alpha = 5^\circ$, it shows that the flow no longer has sufficient energy to overcome the adverse pressure gradient and separates. As the flow-visualisation tests showed, trailing-edge separation occurred in all forced-transition tests, however, for some cases this occurred downstream of the last gauge location.

Power spectral density (PSD) plots from the hot-film gauge signals are presented in Figure 7.6. The most important feature of Figure 7.6 is that the output of the forced-transition cases are very similar to that of the turbulent boundary layer of the free-transition case, which occurs at $x/c \geq 0.8$. Thus showing that the roughness-type trip was capable of producing a truly turbulent boundary layer.

7.1.4 LDA Velocity Profiles

LDA surveys were taken in the wake and underneath the wing, and are shown in Figures 7.7 & 7.8 respectively. Similarly to the free-transition case, as flow moves downstream turbulent mixing causes a thicker wake with a reducing velocity deficit. For the forced-transition cases the velocity deficit is increased and the height of the wake above the ground is reduced. As the integral of the velocity deficit is indicative of the wing's drag, it can be inferred that the drag has increased for the forced-transition cases, and moving the trip upstream also increases drag. Hence, the $x/c = 0.1$ tripped case exhibits the largest drag and the free-transition case the least. The thicker wake of the forced-transition cases is a result of the trailing-edge separation that was observed in previous section; thus the $x/c = 0.1$ case, which displayed the earliest separation point, has the thickest wake.

The percentage difference in area bounded by the velocity deficit of the free- and forced-transition cases (ΔW_A) is given in Table 7.1, showing that the survey at $x/c = 1.5$ has the largest difference between the two conditions. The difference between the free and forced conditions then reduces as flow moves downstream. In free transition, the location of maximum velocity deficit moved upwards from $y/c = 0.416$ at $x/c = 1.5$, to $y/c = 0.442$ at $x/c = 3$ (a movement of $y/c = 0.26$) as a result of the upwash due to the higher downforce generation that was observed through the static pressure measurements. For forced transition at $x/c = 0.1$, however, the reduced downforce generation, and thus reduced upwash component, of the forced-transition wing results in the location of maximum velocity deficit occurring at $y/c = 0.396$ at $x/c = 1.5$ and at $y/c = 0.415$ at $x/c = 3$ (a movement of $y/c = 0.19$).

The surveys taken underneath the wing at $x/c = 0.375$, which are given in Figure 7.5, show that at all tested incidences the normalised velocity is reduced in the forced-transition cases, in that the case with the trip at $x/c = 0.1$ exhibits the lowest velocity. This is

Table 7.1: Percentage difference of the total area bounded by the velocity deficit between the forced-transition cases and the free-transition case

Forced-Transition Trip Location	ΔW_A		
	$x/c = 1.5$	$x/c = 2$	$x/c = 3$
$x/c = 0.1$	68.4 %	47.1 %	42.0 %
$x/c = 0.3$	54.0 %	28.7 %	18.9 %

expected given the reduction in surface suction observed in the static pressure measurements, as a higher pressure is equivalent to a lower flow velocity according to Bernoulli's Theorem. As incidence is increased, the difference in velocity between the free- and forced-transition cases also increases. This shows, as was suggested by the hot-film results, that as incidence is increased the difference between the free- and forced-transition cases becomes greater. It can also be observed that the forced-transition cases exhibit a thicker boundary layer than the free-transition case. By nature the turbulent boundary-layer state is thicker than the laminar counterpart, hence by forcing the turbulent boundary layer to start earlier the boundary layer is thicker at the measurement location. For example, at $\alpha = 0^\circ$ the difference in thickness between the $x/c = 0.1$ forced-transition case and the free-transition case can be observed to be $y/c \approx 0.007$ (2.37%). This means that a thicker boundary layer encounters the adverse pressure gradient. As a thicker boundary layer is inherently more unstable than its thinner counterpart, the likelihood of separation is increased.

7.2 Three-Dimensional Wings

7.2.1 Roughness Trip

Single-Element Wing (CSW-SE)

Force measurements were taken at various ground clearances, Reynolds numbers and with transition forced at chordwise positions $0.1 \leq x/c \leq 0.5$. The graphs presented in Figures

7.9 & 7.10 show the downforce and drag coefficients for all forced-transition locations at various Reynolds numbers and ground clearances.

It was shown previously in Section 6.1 that the single-element wing exhibits significant Reynolds-number sensitivity both in the force-enhancement and force-reduction region. This was attributed to the location and mechanism of transition. Figure 7.9 shows that when the transition point is fixed, this Reynolds-number dependency is almost nullified at higher ground clearances. Differences still occur, however, once the ground clearance reduces to $h/c \leq 0.3$. This shows that ground effect promotes Reynolds-number dependency.

In general, the more downstream trips produce more downforce and less drag than the upstream trips; resulting in a more aerodynamically-efficient wing. Table 7.2 shows the point at which the wing enters the force-reduction region. This shows how contrasting results are given by the free- and forced-transition wing in terms of Reynolds-number dependency. In free transition, as the Reynolds number is increased the force reduction is delayed (the critical ground clearance is lower) and eventually eliminated. In all instances of forced transition, the force-reduction region is advanced such that it occurs at a higher ground clearance. In free transition at $Re = 1.63 \times 10^5$, the wing exhibits no noticeable force enhancement, but when the boundary layer is tripped the force-reduction is delayed such that substantially more downforce is generated.

Similarly to the downforce curves, the drag curves given in Figure 7.10 show reduced Reynolds-number sensitivity for the forced-transition cases in comparison to the free-transition case. The trend in drag coefficient is that the further upstream the trip is placed, the higher the drag coefficient. Additionally, the drag coefficient appears to converge towards the same value at the lowest tested ground clearance, irrespective of Reynolds

Table 7.2: Comparison of the ground clearances at which the force-reduction phenomenon begins for varying Reynolds numbers and trip location, '-' denotes that no force reduction has occurred and thus maximum downforce is at the lowest tested ground clearance

Re	Free	Trip at $x/c = 0.1$	Trip at $x/c = 0.2$	Trip at $x/c = 0.3$	Trip at $x/c = 0.4$	Trip at $x/c = 0.5$
1.63×10^5	0.750	-	0.125	0.125	-	-
1.83×10^5	0.292	0.125	0.142	0.133	-	-
2.04×10^5	0.208	0.133	0.142	0.142	0.125	-
2.24×10^5	0.150	0.142	0.15	0.15	0.142	-
2.44×10^5	-	0.150	0.158	0.158	0.150	0.125
2.65×10^5	-	0.158	0.167	0.167	0.167	0.133
2.85×10^5	-	0.158	0.167	0.175	0.167	0.212

number and trip location. The exception is when the trip is at $x/c = 0.5$, which shows more Reynolds-number sensitivity than the other (upstream) trip locations and does not fully converge at the lowest ground clearance. It should be noted, however, that the $x/c = 0.5$ trip is placed inside the adverse pressure gradient (as the wing's apex is at $x/c = 0.42$). This could be influencing these aspects as the boundary-layer stability is already reduced by the deceleration of the flow, whereas for all other trips it is in the favourable pressure gradient that has a stabilising effect.

The lower Reynolds number forced-transition cases demonstrate lower drag than the free-transition case, and thus perform similarly to higher Reynolds number cases. As Reynolds number increases, however, the drag coefficient of the forced-transition cases becomes higher than that of the free-transition case, negating its feature of acting like a higher Reynolds number flow. This is a known feature of boundary-layer trips as, although they may fix the location of transition, they introduce an additional pressure drag component due to their physical size on the wing.

Although in all cases of forced transition the Reynolds-number sensitivity of the wing reduces, the amount it is reduced by is dependent on trip location. The most upstream

trip location of $x/c = 0.1$ demonstrates the smallest Reynolds-number sensitivity of any, and $x/c = 0.5$ the most (other than the free-transition wing). Thus showing that it is the laminar boundary layer extent that introduces the Reynolds-number sensitivity rather than the presence of a laminar separation bubble.

It is unwise to read too much into these results, other than the predominant trends, due to the boundary-layer thickness being altered by Reynolds number, trip location and ground clearance at the same time. This fact also highlights the difficulties with attempting to fix transition locations for motorsport applications. Aerodynamicists will map the aerodynamic performance of the car at a range of pitch, yaw and roll angles, and ground clearances; all which affect the transition point and boundary-layer thickness, so attempting to size trips based on a procedure or guideline is difficult.

The results of flow-visualisation tests, carried out at various ground clearances and Reynolds numbers, are given in Figures 7.11 & 7.12. The roughness trip causes an immediate transition to a turbulent boundary layer, thus there is no laminar separation and consequently, no transition bubble. In some cases, there is evidence of turbulent trailing-edge separation. The line of separation is not straight, but instead shows peaks, which leads to the notion that there are vortical or unsteady motions, such as wavering of separation lines, present. A consequence of the trailing-edge separation is that it changes the effective shape of the wing such that it decreases the diffuser effect, and subsequently flow velocity underneath the wing. The loss of downforce because of this also reduces the upwash component and thus alters the wake structure.

Figure 7.11 shows the influence of ground proximity on the free- and forced-transition cases at constant Reynolds number ($Re = 2.44 \times 10^5$). Reducing ground clearance causes greater flow velocity under the wing, hence the boundary layer is thinner and skin friction

is reduced. For all trip locations the trailing-edge separation point moves upstream as the ground clearance is reduced. This occurs because the adverse pressure gradient becomes larger at lower ground clearances. As the trip is moved downstream, with Reynolds number and ground clearance remaining constant, the separation point also moves downstream. This shows why the downforce and drag coefficients incrementally increase as the trip is moved upstream. The separation bubble effectively resets the boundary layer, thus allowing it to overcome the adverse pressure gradient without separating. By forcing a turbulent boundary layer prematurely this cannot occur. In the forced-transition cases the turbulent boundary layer covers a larger portion of the wing surface, this means that it loses more momentum and makes separation more likely. As the trip location moves upstream the momentum loss that occurs in the turbulent boundary layer, that ultimately causes the separation, begins sooner. Hence the separation point moves with the trip location. No separation is observed for the $x/c = 0.5$ trip at higher ground clearances, as the turbulent boundary layer is capable of overcoming the short length of adverse pressure gradient.

The largest difference in downforce production between the free- and forced-transition cases is observed at the lowest tested ground clearance and lowest tested Reynolds number (Figure 7.9). Flow visualisation of the tripped case is given in Figure 7.12. As presented in Section 6.1, the free-transition wing at $h/c = 0.125$ and $Re = 1.63 \times 10^5$ exhibited a fully-laminar separation i.e. no turbulent reattachment; hence the small amount of downforce and large drag that it produced. For the forced-transition case, however, whilst trailing-edge separation still occurs, the boundary layer is able to stay attached longer into the adverse pressure gradient. It is because of the turbulent state that the boundary layer remains attached, and allows the wing to produce more downforce by remaining attached longer than a laminar boundary layer would have.

Figure 7.12 also shows the effect of Reynolds-number variation at constant ground clearance and trip location ($x/c = 0.1$). At $h/c = 0.3125$ (Figure 7.12a) the separation point moves downstream as Reynolds number is increased; as the flow has more energy to help it move further into the adverse pressure gradient before separating. At the lower ground clearances of $h/c = 0.225$ (Figure 7.12b) and $h/c = 0.125$ (Figure 7.12c) there is little or no change in the separation point. At these points, it must therefore be assumed that the difference in downforce production is due to a different boundary-layer thicknesses.

The small difference in downforce in the force-enhancement region of Figure 7.9 is due to the difference in trailing-edge separation point. The boundary layer of the higher Reynolds number case remains attached longer, which leads to an increase in downforce. In the force-reduction region, however, the separation point remains relatively unchanged and the boundary-layer displacement thickness is the determining factor. The thicker boundary layer of the lower Reynolds number cases cause a reduction in the effective ground clearance, leading to additional constraint to the flow. Hence, the lower Reynolds number cases produce more downforce. The ground clearance at which this crossover occurs alters with trip position, but with no distinct trend.

Multi-Element Wing (CSW-ME)

As discussed previously, in free transition the multi-element wing does not demonstrate significant Reynolds-number sensitivity. The downforce and drag coefficients at varying trip position, ground clearance and Reynolds number are given in Figures 7.13 & 7.14 respectively. The free-transition wing only displays Reynolds-number sensitivity at lower ground clearances, near to the point of maximum downforce and critical ground clearance. Thus, it is unsurprising that the forced-transition cases also exhibit negligible Reynolds-number sensitivity at higher ground clearances. The roughness trips are, however, capable

of reducing Reynolds-number sensitivity at the lower ground clearances; but this is at the expense of downforce, which is reduced across all tested ground clearances. Similarly to forced-transition tests on the single-element wing, the most upstream trips cause the wing to produce the least downforce of all tested cases; with downforce increasing as the trip is moved downstream.

The graphs of constant Reynolds number, given in Figure 7.13, show that roughness trips reduce the downforce coefficient at the majority of ground clearances. At $Re = 2.44 \times 10^5$ the trips cause reduced downforce at all ground clearances, and downforce progressively decreases as the trip is moved upstream. The trips, however, delay the force-reduction region such that it begins at a lower ground clearance, but then give a faster reduction in downforce as the ground clearance continues to decrease.

At $Re = 1.63 \times 10^5$ the upstream trip positions cause an incremental reduction in downforce for higher ground clearances. At the lower ground clearances, however, they act slightly differently. The ground clearance at which the wing enters the force-reduction region is reduced as the wing is tripped, with the more downstream trip locations delaying it most. Hence, at the minimum tested ground clearance the $x/c = 0.5$ trip gives the most downforce. It should be noted, however, that the difference in downforce and drag, prior to the reduction region, is less than 2.6% and 3.9% respectively at $Re = 1.63 \times 10^5$ and 4.5% and 1.7% respectively at $Re = 2.44 \times 10^5$.

In free transition, prior to the force-reduction region, the drag curves display incrementally more drag as the Reynolds number decreases. Once the ground clearance has sufficiently reduced, and the wing enters the force-reduction region, significant Reynolds-number sensitivity is observed. In the forced-transition cases the Reynolds-number sensitivity is essentially eliminated for all trip locations and ground clearances. The only

exception being $h/c = 0.125$ for the $x/c = 0.5$ placed trip.

Flow visualisation at varying chordwise positions for $Re = 2.44 \times 10^5$ is given in Figure 7.15. In forced transition the wing clearly behaves differently in the central region of the main plane to that of the outer region of the main plane. In the central region it acts similarly to the single-element wing and the turbulent boundary layer separates near to the trailing edge. The further upstream the trip is placed the earlier the boundary layer separates, hence the more upstream trip locations produce the least downforce. In the outer region of the main plane, no trailing-edge separation is observed as the flap element reduces the adverse pressure gradient by allowing pressure recovery to continue off surface behind the main plane. Given that the trip-location dependency for the single-element wing appeared to be due to the trailing-edge-separation point, it can be assumed that the multi-element wing has less dependency on trip location as only the central region, without the flap element, has separated flow. Although this is probably the key factor in the trip-location dependency, there are still other factors that affect the force being generated by the wing. First is the boundary-layer thickness, as the further upstream the trip is placed, and the earlier the turbulent boundary layer begins, the thicker it will be at the trailing edge. This causes the effective shape of the wing to change such that the expansion ratio between the wing's apex and the trailing edge is reduced, thus decreasing the diffuser effect and, as a result, reducing downforce. Secondly, when the trip is placed at $x/c = 0.5$ a separation bubble still forms (predominantly in the central region); this is the dark region directly behind the trip. This is because the $x/c = 0.5$ trip is located in the adverse pressure gradient, and thus close to the free-transition separation point. It, therefore, causes premature laminar separation and allows a separation bubble to form that encloses the trip. As the separation bubble is a force-enhancement mechanism, when combined with the least trailing-edge separation of any forced-transition cases, it produces

the most downforce of any trip location.

The flap elements exhibit no trip-location dependency, nor do they appear to differ from the free-transition case; presumably because it takes on a fresh boundary layer regardless of the main-plane boundary-layer characteristics. The disadvantage of surface-flow visualisation is that it does not give an indication of the flow even just a small distance from the surface, and thus features can appear more influential than they actually are. In the case of the flaps, for all cases shown in Figure 7.15, the surface-flow structure is nearly identical; that does not mean, however, that characteristics such as boundary-layer thickness and skin friction are the same.

The downforce curves showed that by fixing the transition location, the wing's Reynolds-number dependency was eliminated across the majority of the tested ground clearances. Surface-flow visualisation of the wing with the boundary layer tripped at $x/c = 0.1$ and at $h/c = 0.3125$ for varying Reynolds number is given in Figure 7.16. On the main plane there is negligible difference between the different Reynolds number cases, with trailing-edge separation occurring at the same location for all cases. It was discussed previously that moving the trip altered the downforce as the trailing-edge separation point also moved. The fact that the downforce remains essentially unchanged when the trailing-edge separation point remains static as Reynolds number is varied supports that conclusion, and thus the boundary-layer thickness has a negligible effect in this instance.

The effect of ground clearance on the forced-transition wing (trip located at $x/c = 0.1$) is shown in Figure 7.17. This shows that as the ground clearance reduces the area of trailing-edge separation increases in both length (chordwise) and width. The separation point moving upstream in the chordwise direction is not surprising, as this is the typical response to a stronger adverse pressure gradient (such as reducing ground clearance or

increasing incidence). However, the separation area moving outboard and ahead of the flap shows that the adverse pressure gradient has become so large that even off-surface pressure recovery is not enough to keep flow attached.

7.2.2 Serrated Trip

In the preceding section a roughness trip, which was capable of causing the boundary layer to immediately transition to a turbulent state, was employed. There are, however, other types of boundary-layer trips that can be used. In this section a thin, serrated trip is used to see if it is capable of manipulating the boundary layer in a different manner. Such trips are widely used in motorsport, including Formula 1 and Formula 3.

Single-Element Wing (CSW-SE)

Graphs for free- and forced-transition cases at varying Reynolds number and ground clearance are given in Figures 7.18 & 7.19 for the single-element wing. The serrated trip acts similarly to the roughness trip in that it greatly reduces Reynolds-number sensitivity and increases the downforce of the lower Reynolds-number cases. However, the wing with the serrated trip produced more downforce at all tested ground clearances than the wing with a roughness trip when the trip is positioned at $x/c \leq 0.3$, and similar downforce at trips positioned $x/c \geq 0.4$; as shown in Figure 7.20. Furthermore, the Reynolds-number sensitivity in the force-reduction region is less than it was with the roughness trip.

Although the trip at $x/c = 0.1$ still gives a large force-reduction region for $Re = 1.63 \times 10^5$, it is acting similarly to that which would be expected for $2.04 \times 10^5 \leq x/c \leq 2.24 \times 10^5$ both in terms of the shape and magnitude of the downforce curve. For the trips placed at $x/c \leq 0.3$, a small force-reduction region is observed for the higher Reynolds number cases. As the trip is moved downstream to $x/c \geq 0.4$, however, the force-reduction region

is eliminated at all tested ground clearances.

Whilst the downforce curves show significantly reduced Reynolds-number sensitivity in comparison to the free-transition case, for the drag curves forced-transition does not have a large effect on the Reynolds-number sensitivity of the drag curves. The drag curves for the forced-transition cases show a similar trend to the free-transition wing, whereby as the Reynolds number increases the drag coefficient reduces. In general, the more downstream the trip is placed the lower the drag coefficient, although for the trips placed at $x/c \geq 0.2$ there is very little difference. In comparison to the roughness trips (Figure 7.21), there is much more Reynolds-number sensitivity than the roughness trip for trips located at $x/c \leq 0.3$, and similar Reynolds-number sensitivity for trips at $x/c \geq 0.4$.

Flow visualisation in Figure 7.22 shows the influence of the serrated trip's location at $h/c = 0.3125$ and $Re = 2.44 \times 10^5$. The most important observation from Figure 7.22 is that the serrated trip does not cause the boundary layer to become turbulent immediately, as a separation bubble is still clearly visible. This is in contrast to the roughness trip, which immediately produced a turbulent boundary layer and subsequently eliminated the separation bubble. The wing with the trip placed at $x/c = 0.1$ shows almost identical surface-flow structures to the free-transition wing. When the trip is placed at $x/c = 0.3$, however, the bubble is clearly influenced by the serrations of the trip, leading to the leading edge of the bubble also appearing serrated, and a wavering of the reattachment line. This shows that the serrated trip induces a small amount of turbulence into the boundary layer due to the serrations producing vortex structures, which destabilise the boundary layer and in turn cause transition to occur prematurely compared to the free-transition case (though not immediately at the trip). Finally, as the trip is moved to $x/c = 0.5$ the separation bubble is moved upstream as the laminar boundary layer separates on the trip's leading edge. Despite the earlier separation point, the bubble's length is not too dissimilar

to the other cases, as the reattachment point is also moved upstream.

The effect of the serrated trip, placed at $x/c = 0.1$, at varying ground clearance and Reynolds number is given in Figure 7.23. At $h/c = 0.3125$, shown in Figure 7.23a, as the Reynolds number is increased the separation point moves downstream, and the bubble length reduces such that the reattachment point does not move with varying Reynolds number. The force results showed that the serrated trip positioned at $x/c = 0.1$ was capable of introducing a force-enhancement region to the wing operating at $Re = 1.63 \times 10^5$. The free-transition wing at $Re = 1.63 \times 10^5$, on the other hand, exhibited no force-enhancement at any tested ground clearance. In free transition, the lack of force enhancement was a result of the boundary layer not reattaching, hence for force-enhancement to occur the boundary layer must reattach in the forced-transition case; Figure 7.23a shows that this happens.

Although the trip does not have a clear effect on the surface-flow structures, it may still be having an effect on the stability of the boundary layer. The trip introduces perturbations into the boundary layer such that turbulence production in the separated shear layer is increased and thus promotes boundary-layer reattachment to form a bubble. Hence, it is acting similarly to the short bubble shown in Figure 5.13. As the ground clearance reduces, however, the adverse pressure gradient becomes too steep and the boundary layer does not reattach after laminar separation; as shown in Figure 7.23a & 7.23b. For $Re \geq 2.03 \times 10^5$ no force-reduction phenomenon was observed in the force measurements at any tested ground clearance, hence both feature clear laminar and turbulent regions occurring either side of the separation bubble, and are not too dissimilar from the free-transition cases.

The force measurements (Figure 7.18) showed that the serrated trip was capable of removing the force-reduction region at low Reynolds numbers, and increasing downforce such

that the wing produced similar downforce levels to higher Reynolds number cases. Figure 7.24 compares free transition and forced transition (trip at $x/c = 0.3$) at $Re = 1.63 \times 10^5$ and $h/c = 0.125$. The surface-flow structures appear very similar, with laminar separation producing a bubble of similar size for each case. Given that the force measurements show that at this Reynolds number and ground clearance the forced-transition wing is producing three times more downforce than the free-transition wing, it can only be a result of the boundary layer reattaching behind the bubble for the forced-transition case, and not for the free-transition case.

Overall, whilst the force measurements appear to show significant differences between the free- and forced-transition cases, the surface-flow structures do not seem drastically different. One notion is that the bubble size and boundary-layer thickness are different, as these are factors that affect the forces produced by the wing but cannot be seen in the shear stresses on the surface.

Multi-Element Wing (CSW-ME)

The downforce and drag curves for the multi-element wing with the serrated trip are given in Figures 7.25 & 7.26. The serrated trip is similar to the roughness trip in that it reduces the Reynolds-number sensitivity in the force-reduction region, but with the consequence of reduced downforce across all tested ground clearances. The reduction in downforce, however, is less than that of the roughness trip; hence it reduces Reynolds-number sensitivity with a smaller loss of downforce. The trip is not capable of eliminating the force-reduction region, however it makes the ground clearance at which the wing enters this region more consistent across the tested Reynolds number range. This occurs by the critical ground clearance being reduced for the lower Reynolds number cases and raised for the higher Reynolds number cases.

The drag curves (Figure 7.26) display a similar trend to the downforce curves. In all cases of forced transition with the serrated trip, the Reynolds-number sensitivity of drag is reduced but the drag force is not increased, as it was for the roughness trip. Although the serrated trip did not demonstrate a significant reduction in Reynolds-number sensitivity for the single-element wing's drag coefficient, for the multi-element wing it does. This may, however, be a result of the Reynolds-number sensitivity already being low for the multi-element wing.

Again, the act of forcing transition has the largest effect at the lowest tested ground clearance and Reynolds number, for which the wing is in the force-reduction region. The serrated trip does not fully eliminate the force-reduction region, but it is capable of allowing the wing to produce more downforce. Figure 7.27 shows the effect of the trip at this ground clearance and Reynolds number. As previously discussed, the force-reduction is a result of trailing-edge separation primarily in the central region of the main-plane. Both the $x/c = 0.1$ and $x/c = 0.3$ located trips eliminate the majority of separated flow. This is a result of the laminar separation point occurring further downstream, and thus the bubble being located further into the adverse pressure gradient. The laminar boundary layer is capable of remaining attached longer due to the perturbations introduced by the serrated trip. It has been discussed previously how the bubble helps the boundary layer overcome the adverse pressure gradient by effectively reattaching a fresh boundary layer. Hence, when the bubble occurs further downstream, the remaining adverse pressure gradient that the reattached boundary layer must overcome is shorter. Finally, with the trip placed at $x/c = 0.5$ the laminar boundary layer separates at the trip, which aids in reducing the Reynolds-number sensitivity compared to trips at $x/c \leq 0.4$.

It is clear that the serrated trip may be used as a method to manipulate the boundary layer, rather than tripping it to a fully-turbulent state, and that it is capable of reducing

Reynolds-number sensitivity whilst maintaining a similar surface-flow structure to the free-transition case. It will not always be possible for the wing to act as if it were operating at a higher Reynolds number because it is mainly the elimination of trailing-edge separation for the low Reynolds number cases that gives this characteristic, and there is no trailing-edge separation at sufficiently high Reynolds numbers. Furthermore, the height of the trip relative to the boundary layer thickness has an effect, and lower Reynolds numbers will always yield thicker boundary layers, and so a thinner trip at a higher Reynolds number may influence the wing in a different manner.

7.3 Summary & Discussion

Forced-transition tests were conducted by placing roughness-type trips on the wing's suction surface. At low Reynolds numbers the use of the trips is capable of making the wing act as if it is at a higher Reynolds number in terms of force production; however this aspect is lost as the Reynolds number increases. This could be due to the size of the trip compared to the boundary-layer thickness, as when the Reynolds number increases the boundary layer becomes thinner, making the trip appear larger in the boundary layer and potentially changing the transition characteristics. Whilst the difference in boundary-layer thickness means that Reynolds-number dependency can never be fully negated, the primary goal of a racing team when wind-tunnel testing is that improvements found in the wind tunnel translate to the track. Therefore, surface-flow features are not of particular importance as long as the force measurements correlate correctly. The trips are a step towards achieving this, unfortunately, the Reynolds numbers which they replicate are still quite low compared to track, and the higher Reynolds number forced-transition cases did not appear to replicate that which would be expected at even higher Reynolds numbers.

The act of forcing boundary-layer transition is commonly cited as a cure for the lower Reynolds number of sub-scale wind tunnel testing, as it moves the transition location upstream. Whilst this may hold true in aerospace applications, where the full-scale Reynolds number is so high that turbulent boundary layers will occur almost immediately; for motorsport applications, where the difference between full- and test-scale is quite small this may not necessarily hold true. Although it may be possible to match the force coefficient of a lower Reynolds number wing to that of a higher Reynolds number, the flow characteristics will never be identical due to the difference in boundary-layer thickness and energy content of the flow. The fully-turbulent CFD simulations shown in Chapter 6 acted in an almost identical manner to the wing with forced transition at $x/c = 0.1$, as both had a turbulent boundary layer form from close to the leading edge of the wing. This showed that the free- and forced-transition wings behave in completely different manners in terms of flow characteristics. A racing team, therefore, must presumably choose between a better estimate of full-scale force coefficient, or better representation of full-scale flow structures from their sub-scale wind tunnel models.

A serrated trip was also used, and had a similar effect of reducing the Reynolds-number sensitivity of the downforce coefficient. Unlike the roughness trip, however, it did not have a significant influence on the drag coefficient. Surface-flow visualisation showed that unlike the roughness trip, the serrated trip did not cause the boundary layer to immediately transition to a turbulent state. Instead, it only introduced perturbations into the boundary layer such that a separation bubble still formed, but it was shorter in length. It was capable, however, of eliminating the force-reduction phenomena of low Reynolds number, low ground clearance cases. This allowed the wing to produce a similar downforce to a higher Reynolds number case, and with similar surface-flow structures to the free-transition wing.

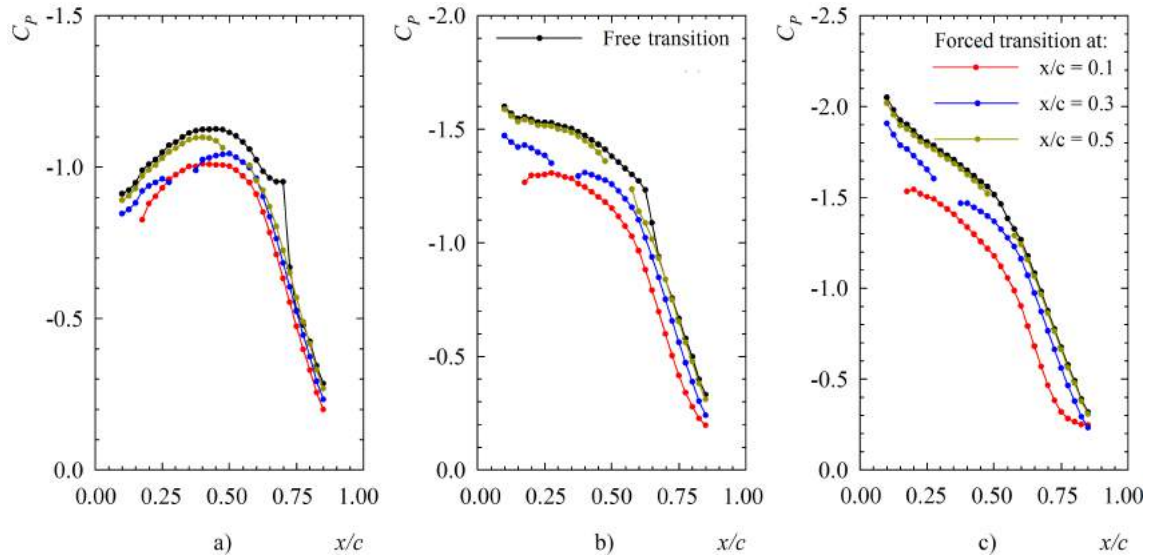


Figure 7.1: Suction-surface pressure distributions for free- and forced-transition at varying wing incidence a) $\alpha = 0^\circ$, b) $\alpha = 3^\circ$, and c) $\alpha = 5^\circ$ (Experimental, 2D wing, points for tappings covered by trip are removed, $Re = 5.94 \times 10^5$, $h/c = 0.3$)

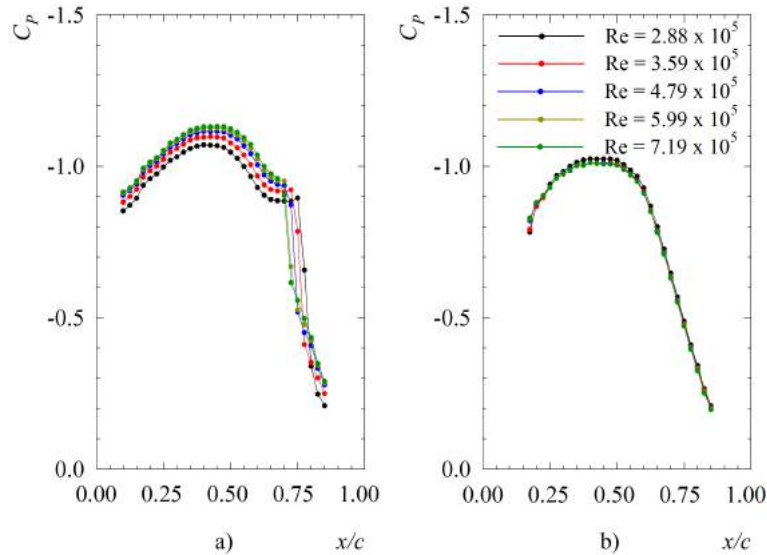


Figure 7.2: Suction-surface pressure distribution for varying Reynolds number with a) free-transition and b) transition forced at $x/c = 0.1$ (Experimental, 2D wing, points for tappings covered by trip are removed, $h/c = 0.3$, $Re = 5.94 \times 10^5$)

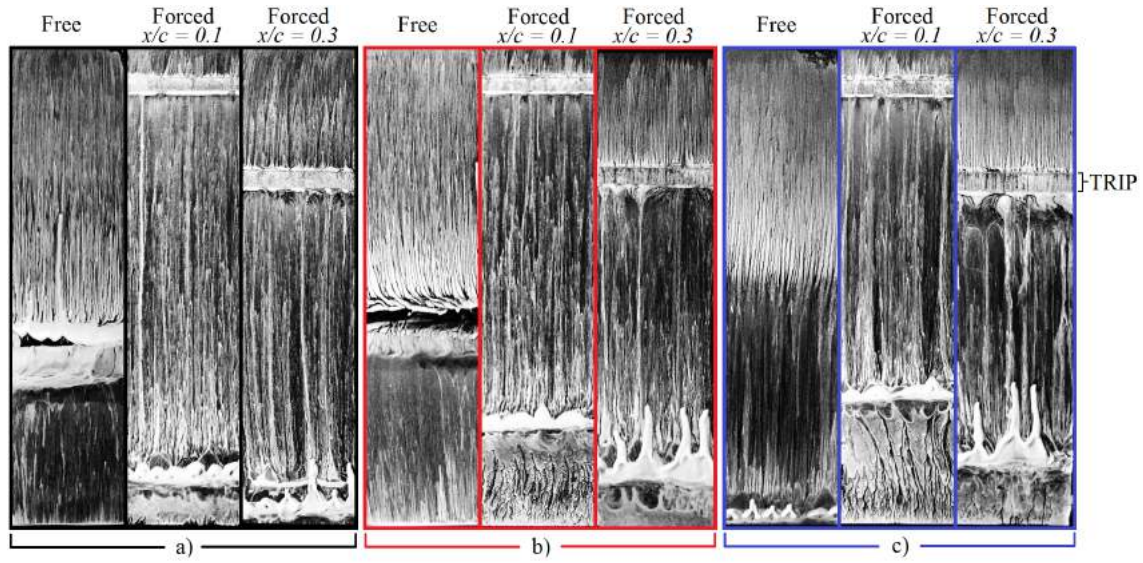


Figure 7.3: Suction-surface flow visualisation for free- and forced-transition cases at incidence a) $\alpha = 0^\circ$, b) $\alpha = 3^\circ$, and c) $\alpha = 5^\circ$ (Experimental, 2D wing, flow moving from top to bottom, $h/c = 0.3$, $Re = 6 \times 10^5$)

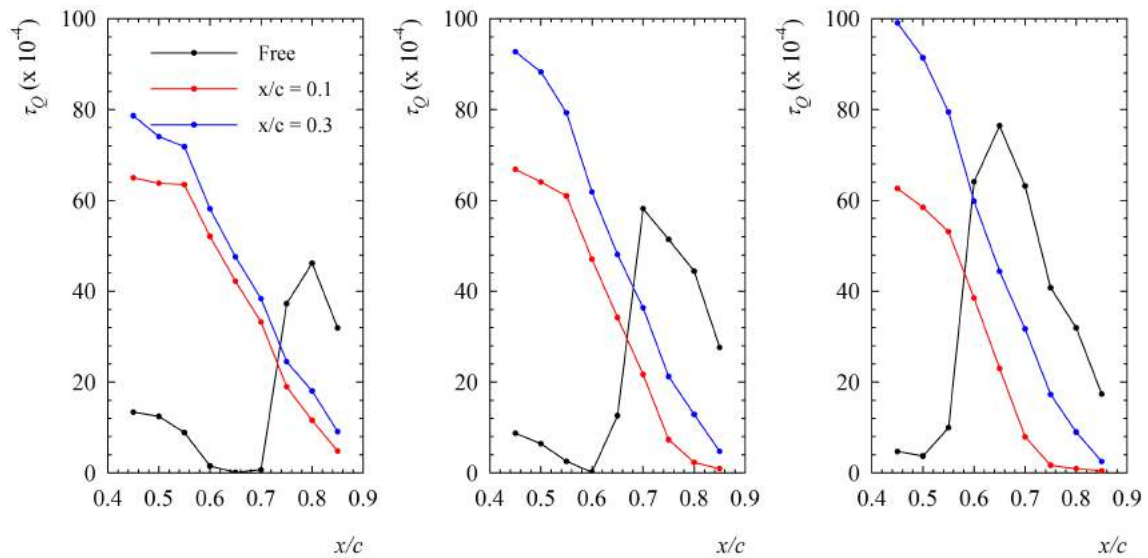


Figure 7.4: Quasi-wall-shear stress for free- and forced-transition cases at incidence a) $\alpha = 0^\circ$, b) $\alpha = 3^\circ$, and c) $\alpha = 5^\circ$ (Experimental, 2D wing, $h/c = 0.3$, $Re = 6 \times 10^5$)

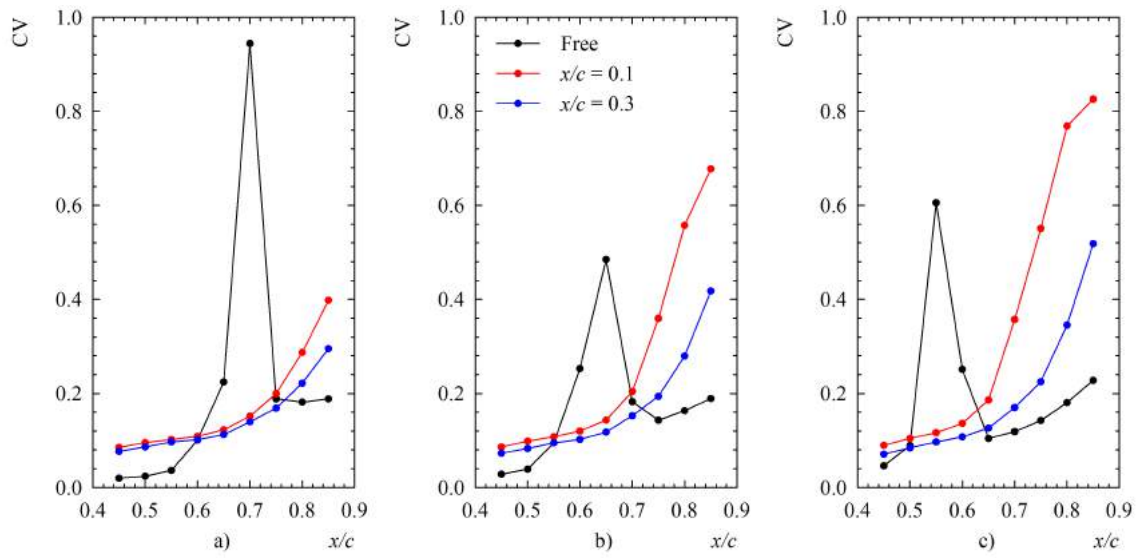


Figure 7.5: CV for free- and forced-transition cases at incidence a) $\alpha = 0^\circ$, b) $\alpha = 3^\circ$, and c) $\alpha = 5^\circ$ (Experimental, 2D wing, $h/c = 0.3$, $Re = 6 \times 10^5$)

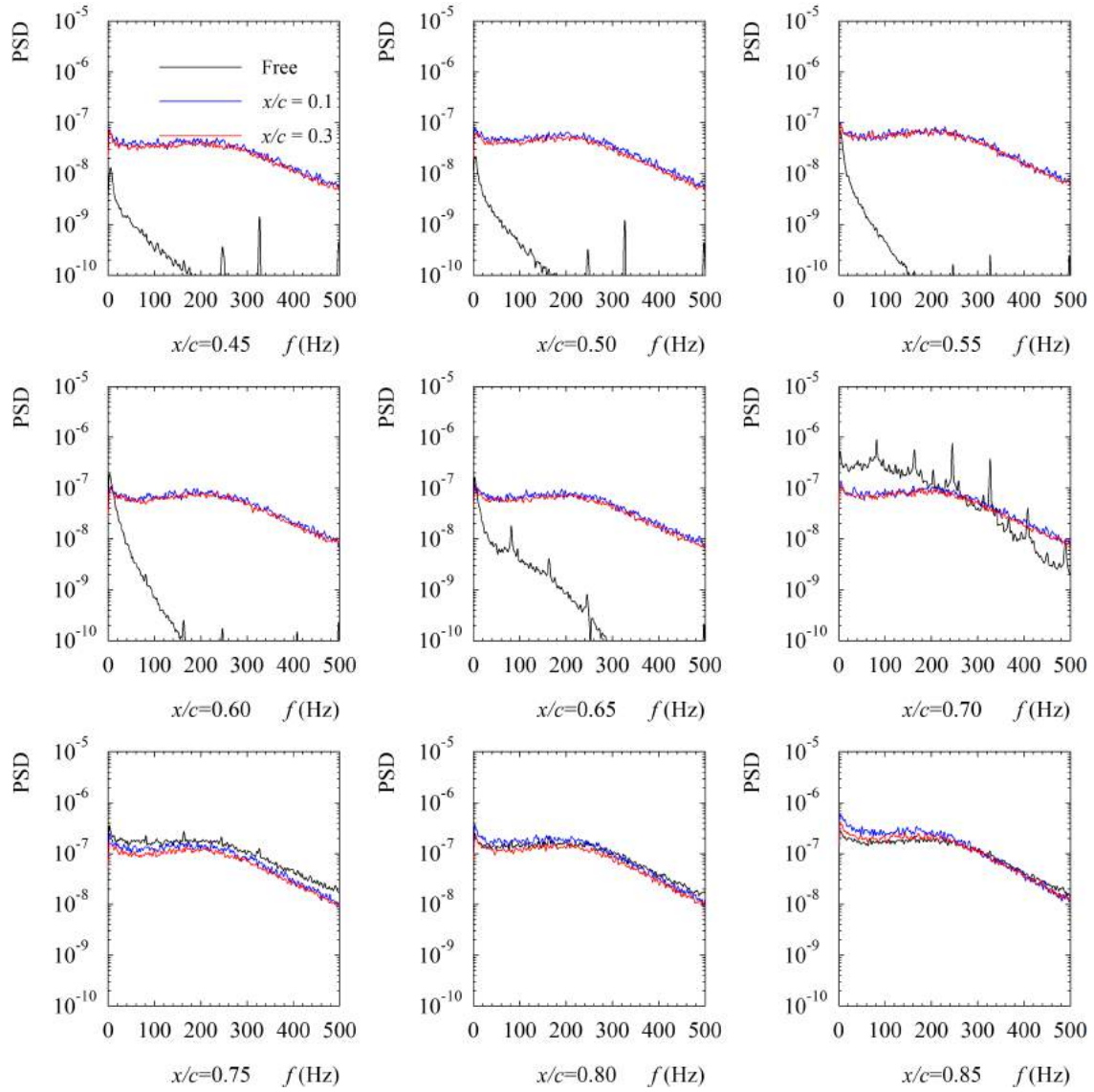


Figure 7.6: PSD of hot-film gauge outputs for free- and forced-transition cases at $\alpha = 0^\circ$ (Experimental, 2D wing, $h/c = 0.3$, $Re = 6 \times 10^5$)

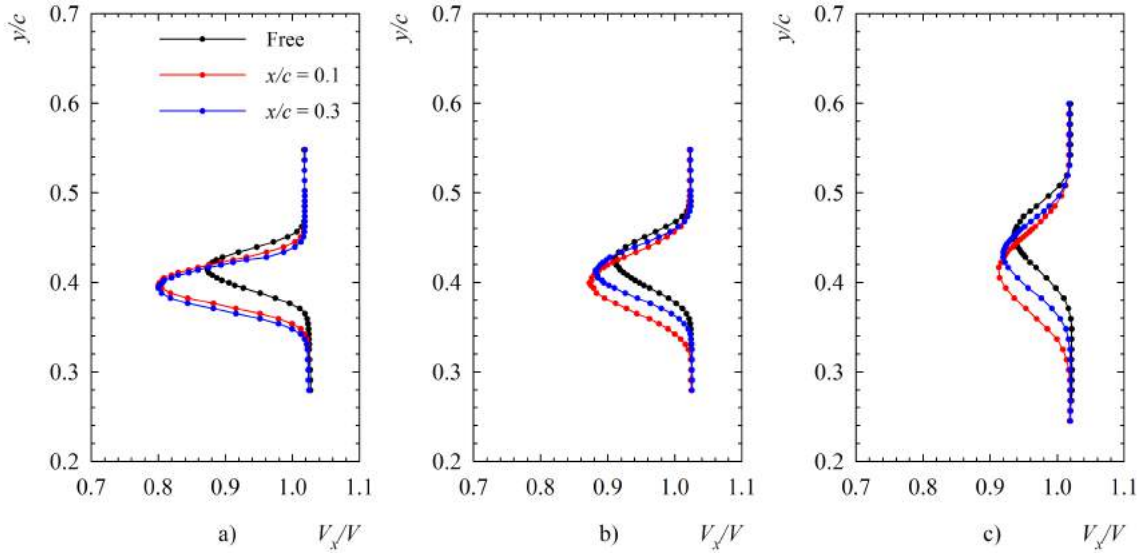


Figure 7.7: Wake surveys for free- and forced-transition cases at $\alpha = 0^\circ$ for a) $x/c = 1.5$, b) $x/c = 2$, and c) $x/c = 3$ (Experimental, 2D wing, $h/c = 0.3$, $Re = 6 \times 10^5$)

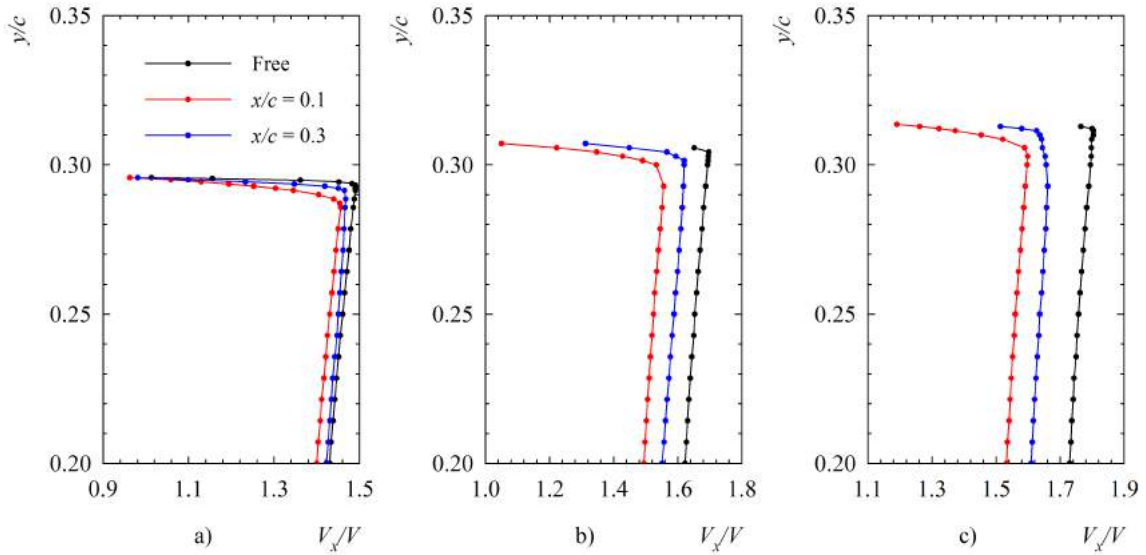


Figure 7.8: Surveys for free- and forced-transition cases at $x/c = 0.375$ for a) $\alpha = 0^\circ$, b) $\alpha = 3^\circ$, and c) $\alpha = 5^\circ$ (Experimental, 2D wing, $h/c = 0.3$, $Re = 6 \times 10^5$)

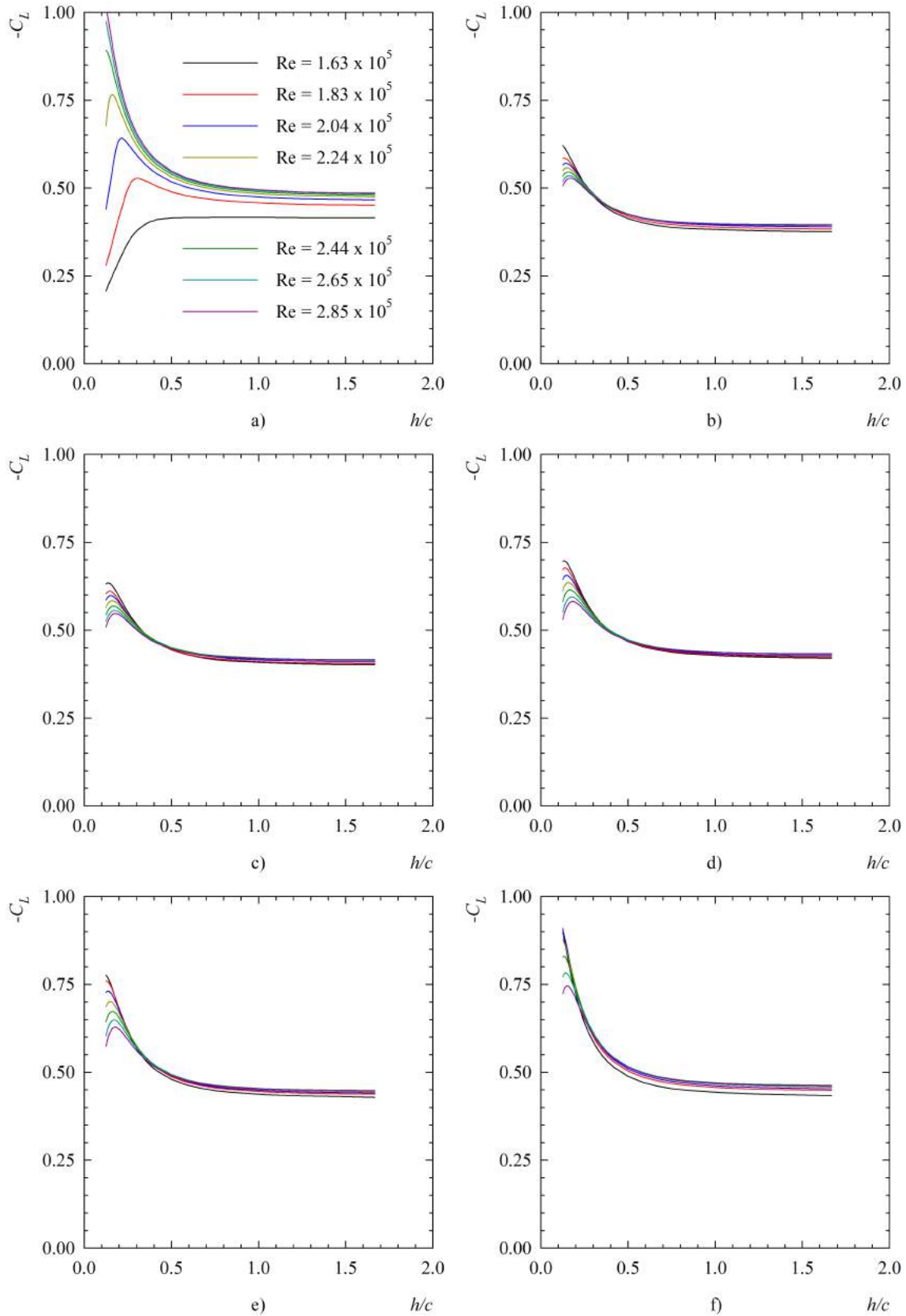


Figure 7.9: Downforce measurements of CSW-SE wing at varying Reynolds number, ground clearance and trip location a) clean wing, b) tripped at $x/c = 0.1$, c) tripped at $x/c = 0.2$, d) tripped at $x/c = 0.3$, e) tripped at $x/c = 0.4$ and f) tripped at $x/c = 0.5$ (Experimental, CSW-SE, roughness trip)

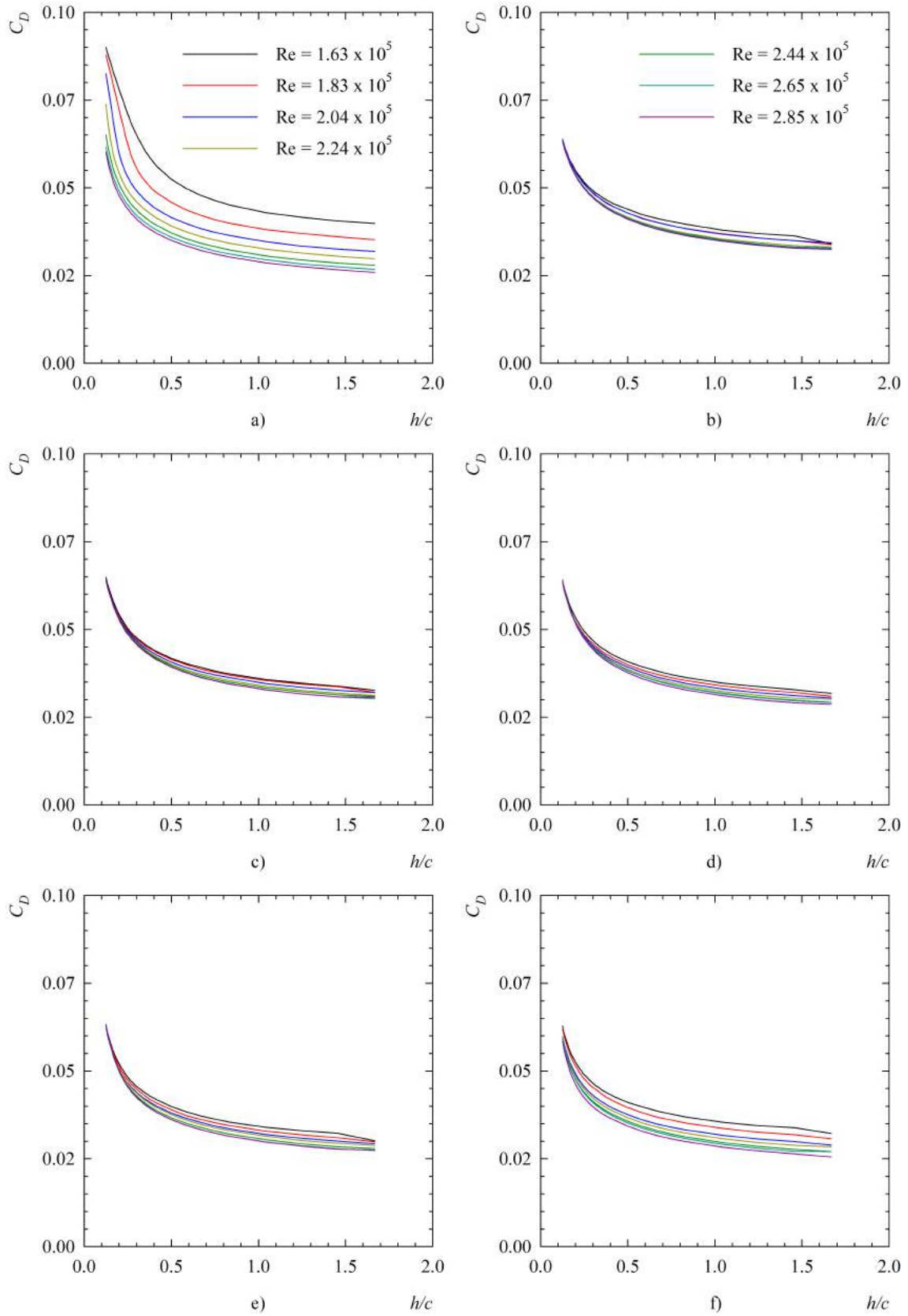


Figure 7.10: Drag measurements of CSW-SE wing at varying Reynolds number, ground clearance and trip location a) clean wing, b) tripped at $x/c = 0.1$, c) tripped at $x/c = 0.2$, d) tripped at $x/c = 0.3$, e) tripped at $x/c = 0.4$ and f) tripped at $x/c = 0.5$ (Experimental, CSW-SE, roughness trip)

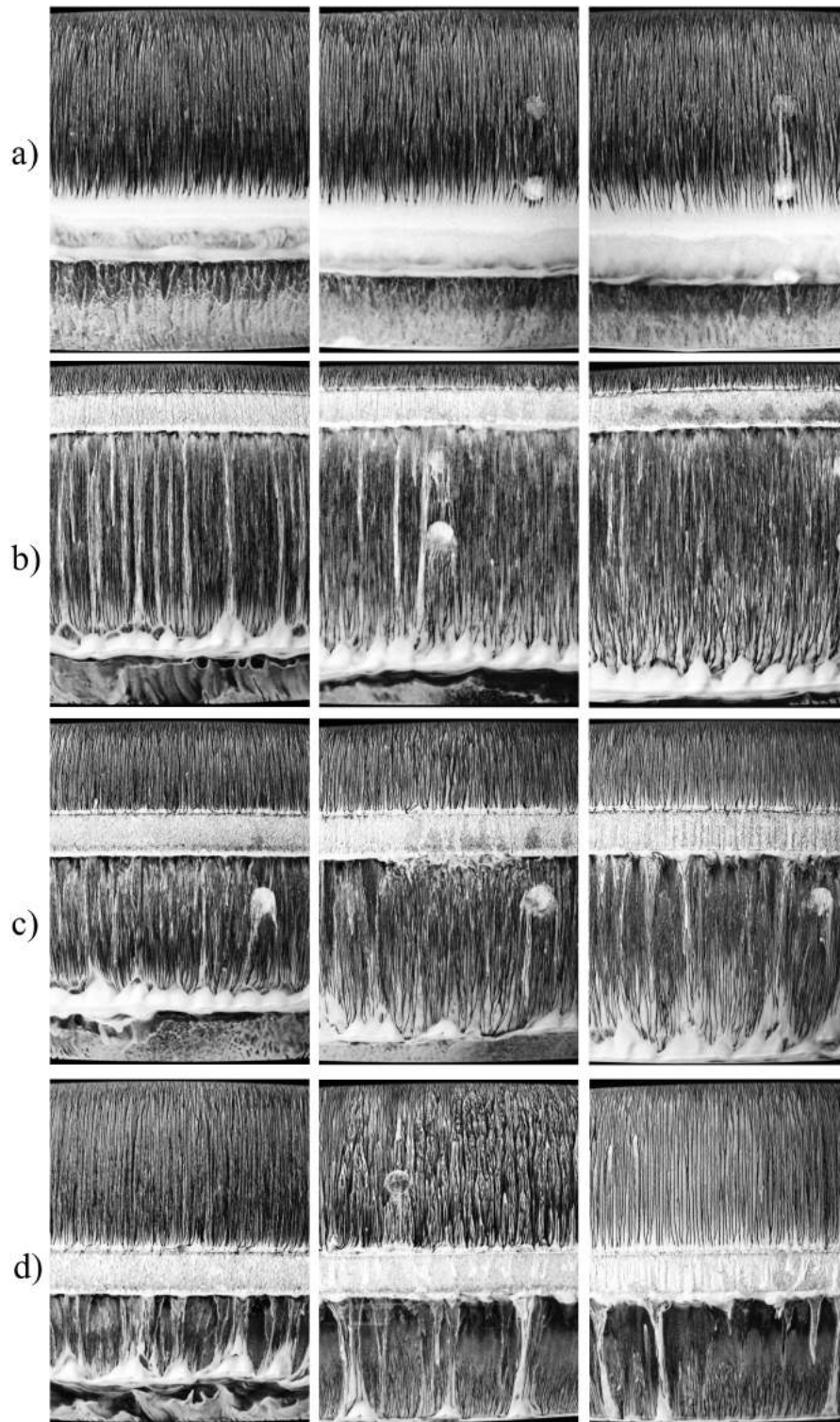


Figure 7.11: Suction-surface flow visualisation for varying trip location at $h/c = 0.125$ (left), $h/c = 0.225$ (middle), and $h/c = 0.3125$ (right) for a) clean wing, b) tripped at $x/c = 0.1$, c) tripped at $x/c = 0.3$ and d) tripped at $x/c = 0.5$ (Experimental, CSW-SE, roughness trip, $Re = 2.44 \times 10^5$, flow moving from top to bottom)

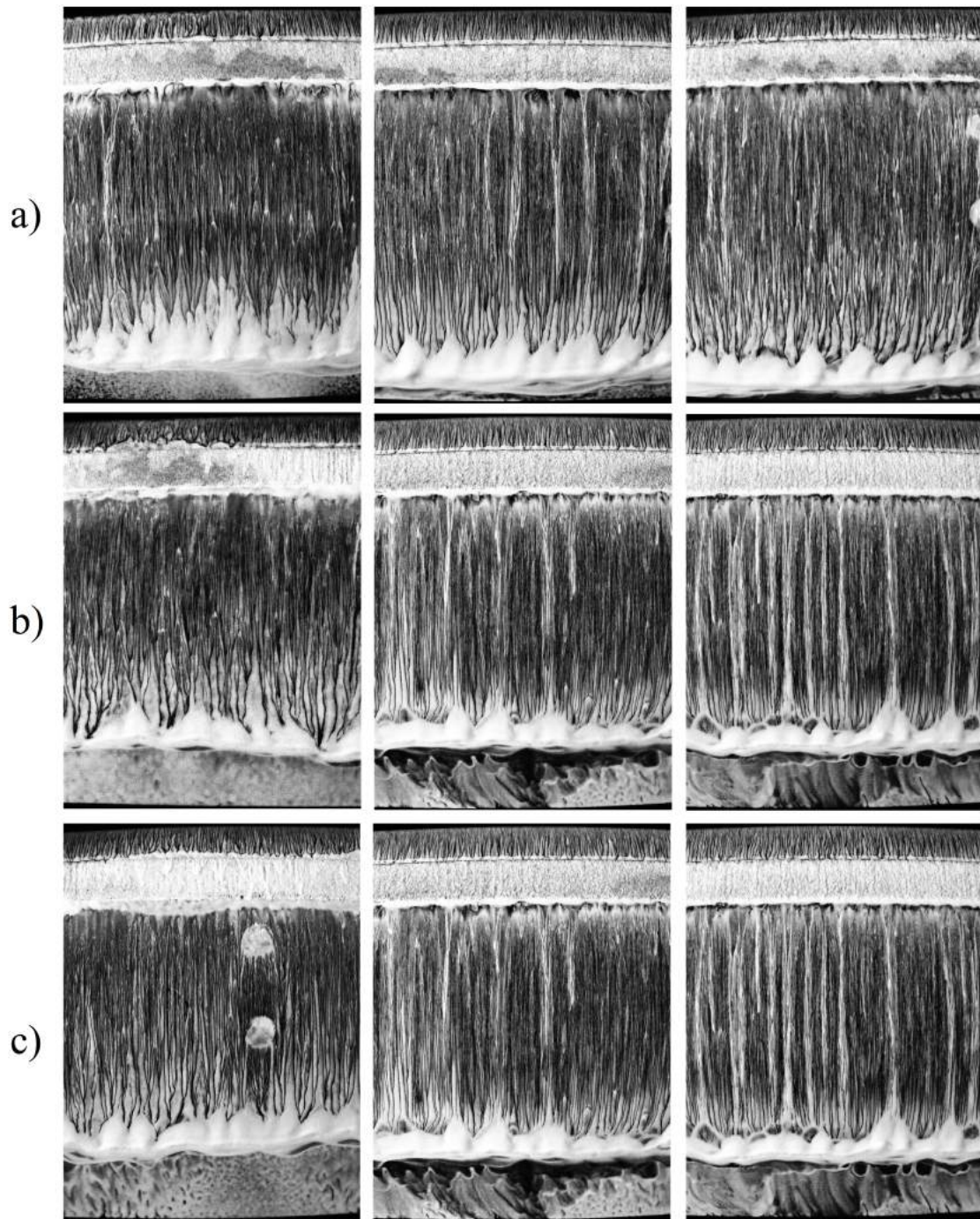


Figure 7.12: Suction-surface flow visualisation with boundary-layer trips at $x/c = 0.1$ for varying Reynolds numbers $Re = 1.63 \times 10^5$ (left), $Re = 2.03 \times 10^5$ (middle), and $Re = 2.44 \times 10^5$ (right) at a) $h/c = 0.3125$, b) $h/c = 0.225$, and c) $h/c = 0.125$ (Experimental, CSW-SE, roughness trip, $Re = 2.44 \times 10^5$, flow moving from top to bottom)

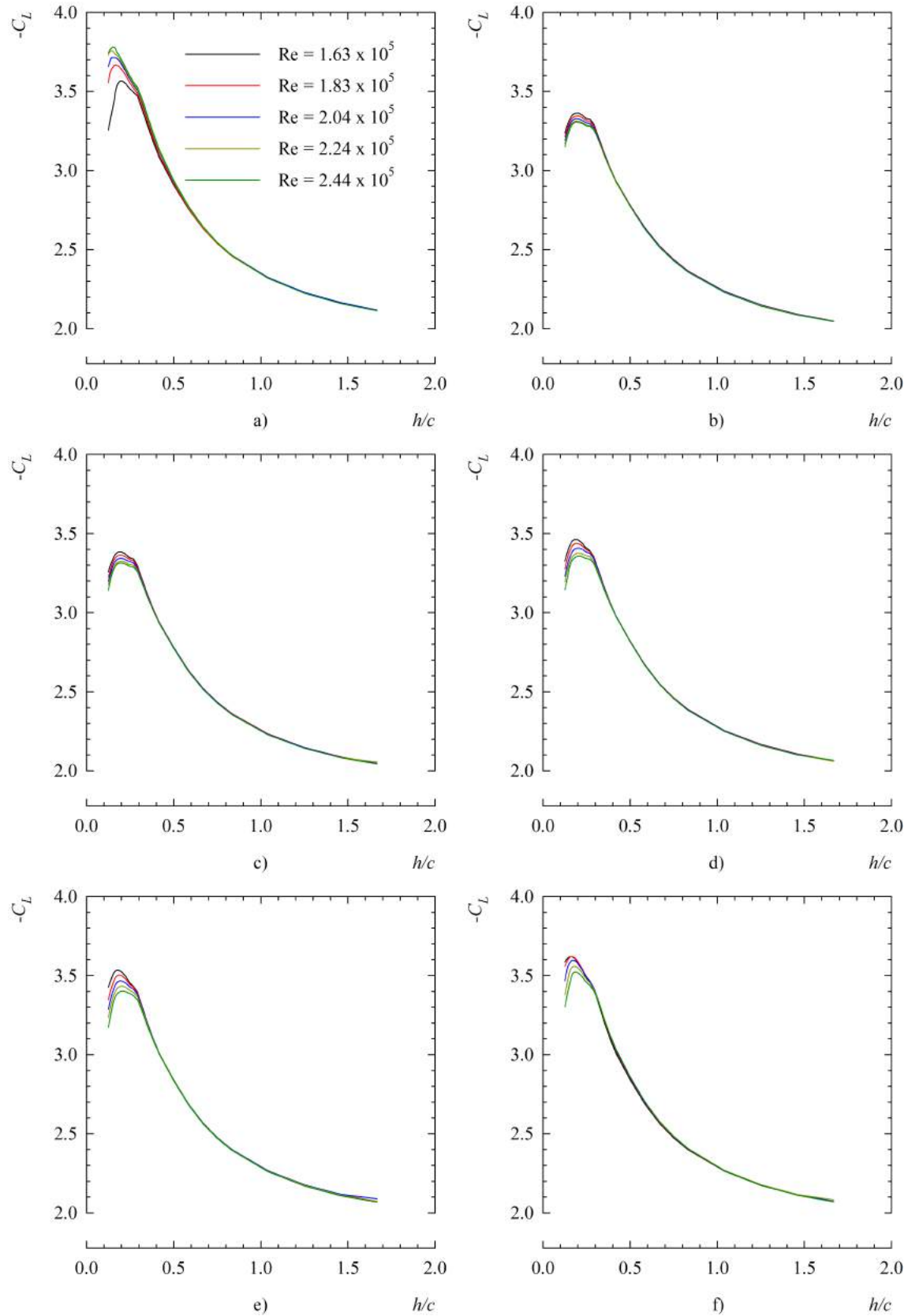


Figure 7.13: Experimental downforce measurements of CSW-ME wing at varying Reynolds number, ground clearance and trip location a) clean wing, b) tripped at $x/c = 0.1$, c) tripped at $x/c = 0.2$, d) tripped at $x/c = 0.3$, e) tripped at $x/c = 0.4$ and f) tripped at $x/c = 0.5$ (Experimental, CSW-ME, roughness trip)

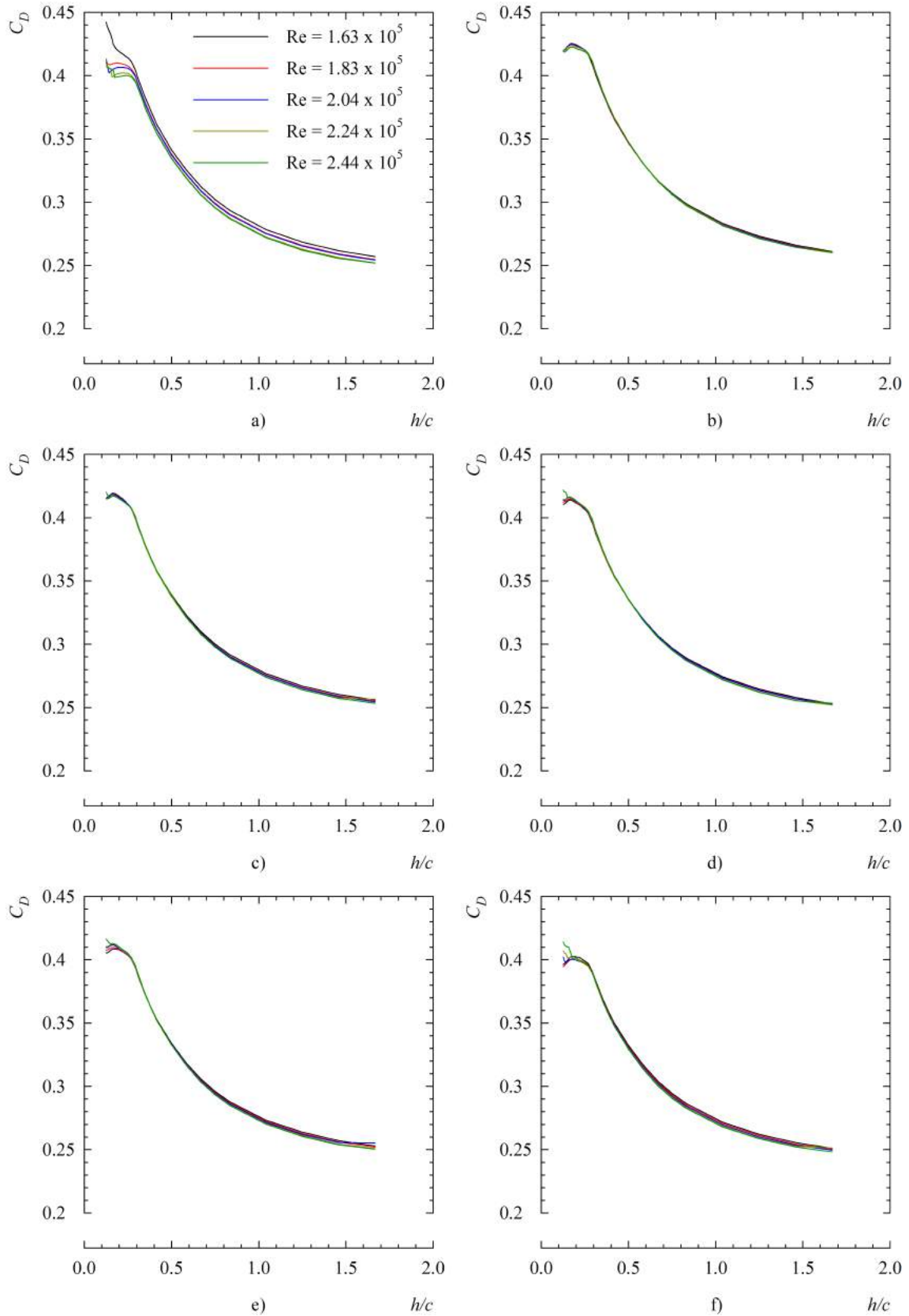


Figure 7.14: Experimental drag measurements of CSW-ME wing at varying Reynolds number, ground clearance and trip location a) clean wing, b) tripped at $x/c = 0.1$, c) tripped at $x/c = 0.2$, d) tripped at $x/c = 0.3$, e) tripped at $x/c = 0.4$ and f) tripped at $x/c = 0.5$ (Experimental, CSW-ME, roughness trip)

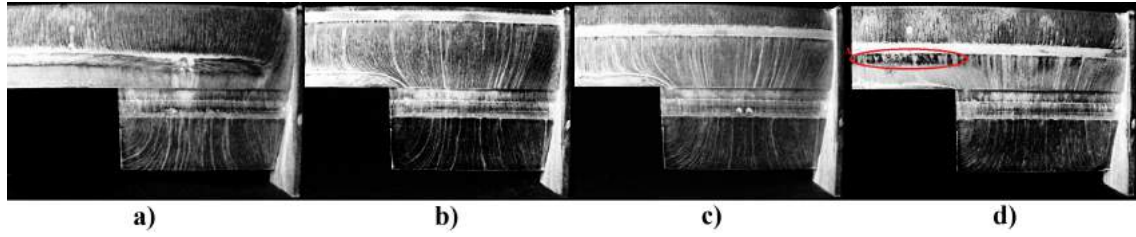


Figure 7.15: Suction-surface flow visualisation on multi-element wing for a) clean wing, b) tripped at $x/c = 0.1$, c) tripped at $x/c = 0.2$, d) tripped at $x/c = 0.3$ (Experimental, CSW-ME, roughness trip, $Re = 2.44 \times 10^5$, $h/c = 0.3125$, flow moving from top to bottom)

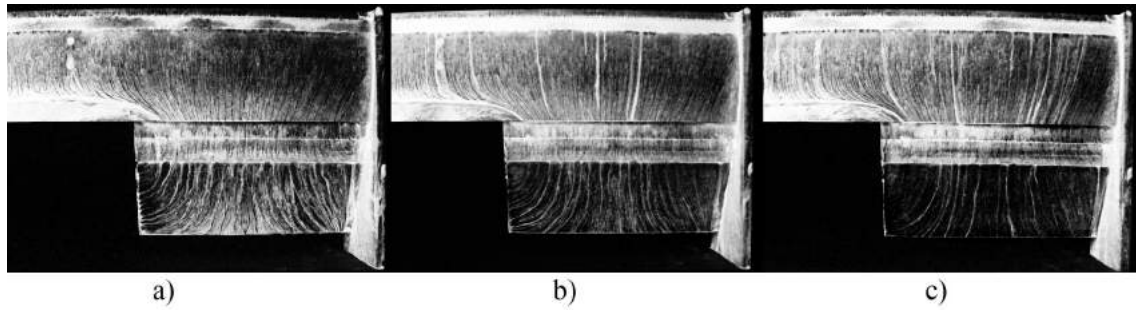


Figure 7.16: Suction-surface flow visualisation on multi-element wing with trip at $x/c = 0.1$ for a) $Re = 1.63 \times 10^5$, b) $Re = 2.03 \times 10^5$, and c) $Re = 2.44 \times 10^5$ (Experimental, CSW-ME, roughness trip, $h/c = 0.3125$, flow moving from top to bottom)

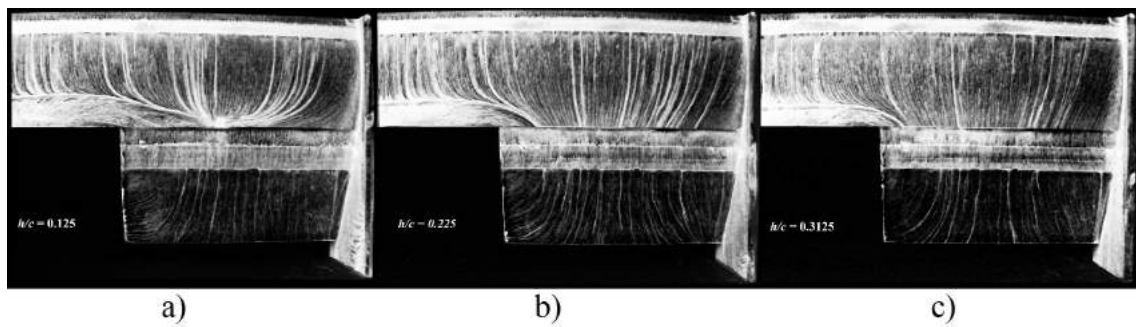


Figure 7.17: Suction-surface flow visualisation on multi-element wing with trip at $x/c = 0.1$ for a) $h/c = 0.125$, b) $h/c = 0.225$, and c) $h/c = 0.3125$ (Experimental, CSW-ME, roughness trip, $Re = 2.44 \times 10^5$, flow moving from top to bottom)

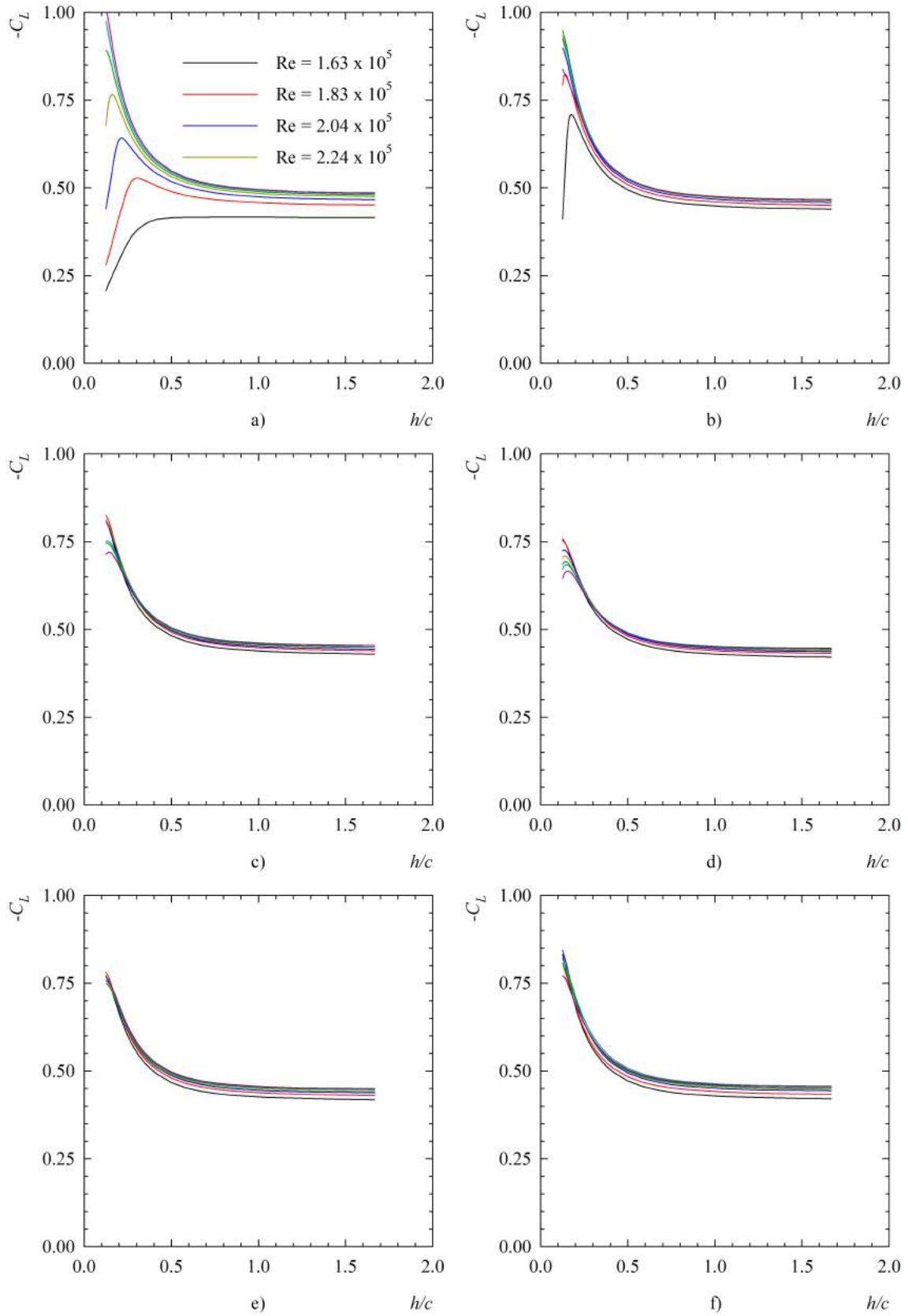


Figure 7.18: Downforce measurements of CSW-SE wing at varying Reynolds number, ground clearance and trip location a) clean wing, b) tripped at $x/c = 0.1$, c) tripped at $x/c = 0.2$, d) tripped at $x/c = 0.3$, e) tripped at $x/c = 0.4$ and f) tripped at $x/c = 0.5$ (Experimental, CSW-SE, serrated trip)

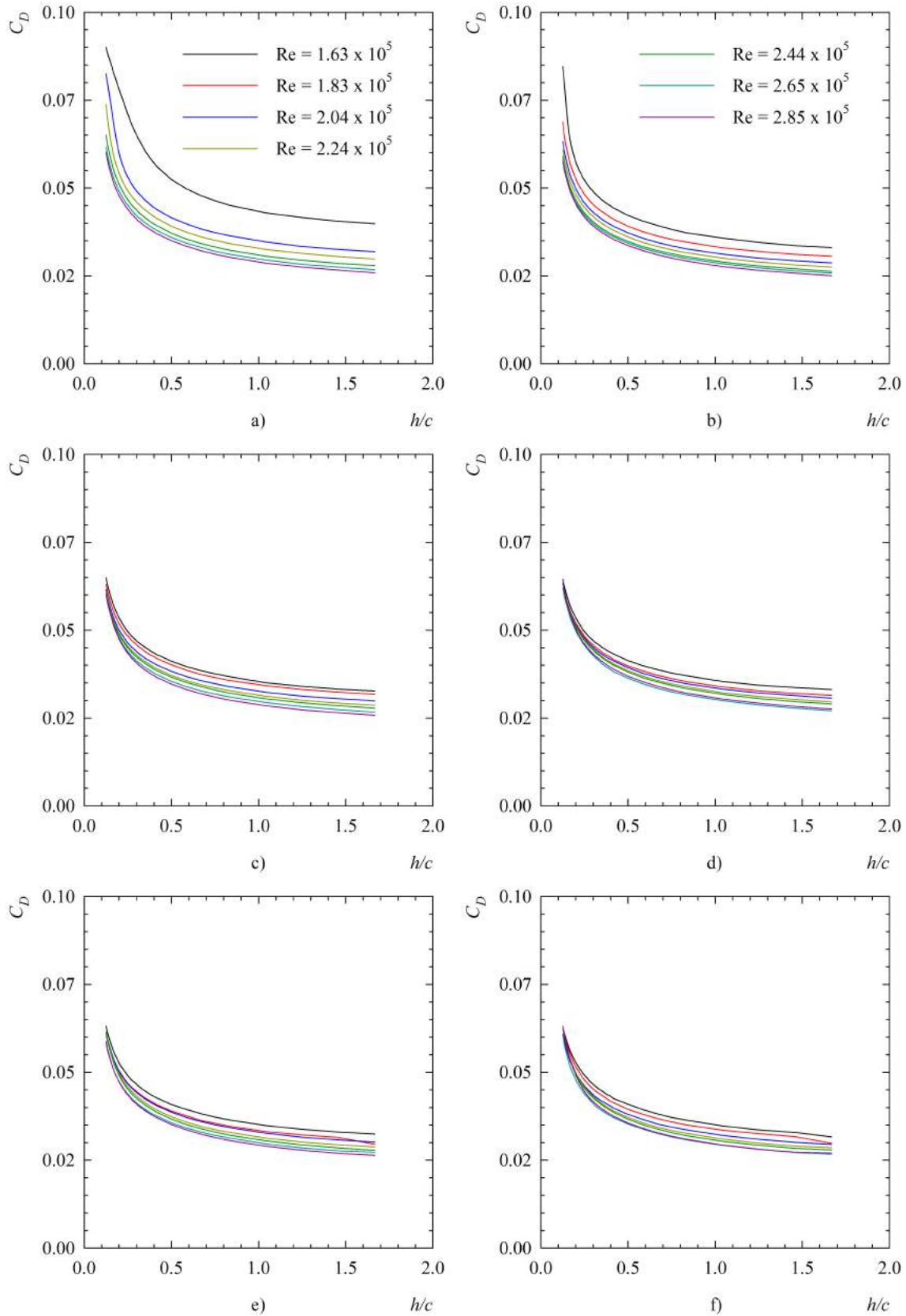


Figure 7.19: Drag measurements of CSW-SE wing at varying Reynolds number, ground clearance and trip location a) clean wing, b) tripped at $x/c = 0.1$, c) tripped at $x/c = 0.2$, d) tripped at $x/c = 0.3$, e) tripped at $x/c = 0.4$ and f) tripped at $x/c = 0.5$ (Experimental, CSW-SE, roughness trip)

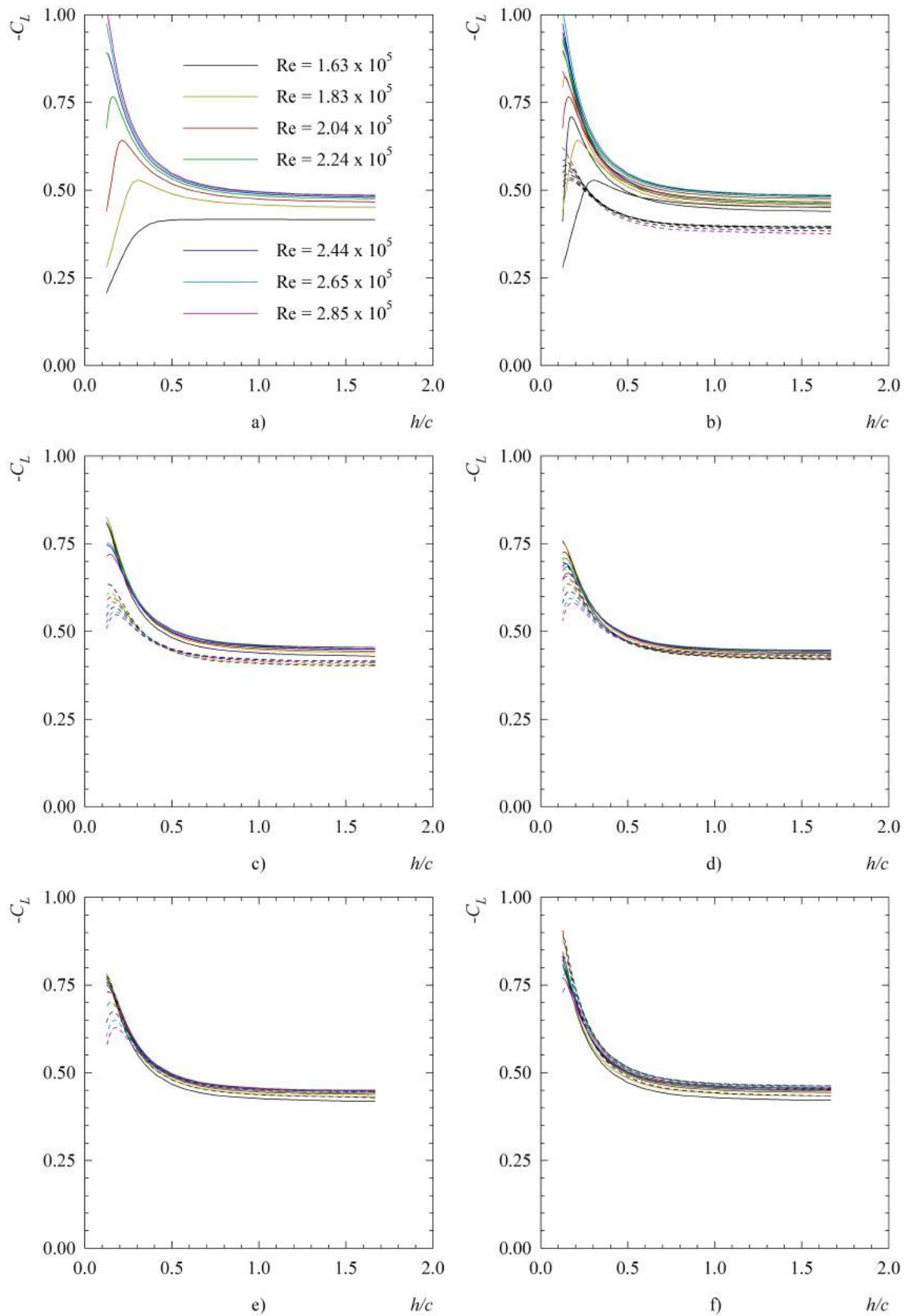


Figure 7.20: Downforce measurements of CSW-SE wing at varying Reynolds number and ground clearance for a) clean wing, b) tripped at $x/c = 0.1$, c) tripped at $x/c = 0.2$, d) tripped at $x/c = 0.3$, e) tripped at $x/c = 0.4$ and f) tripped at $x/c = 0.5$ with serrated trip (solid line) and roughness trip (dashed line)

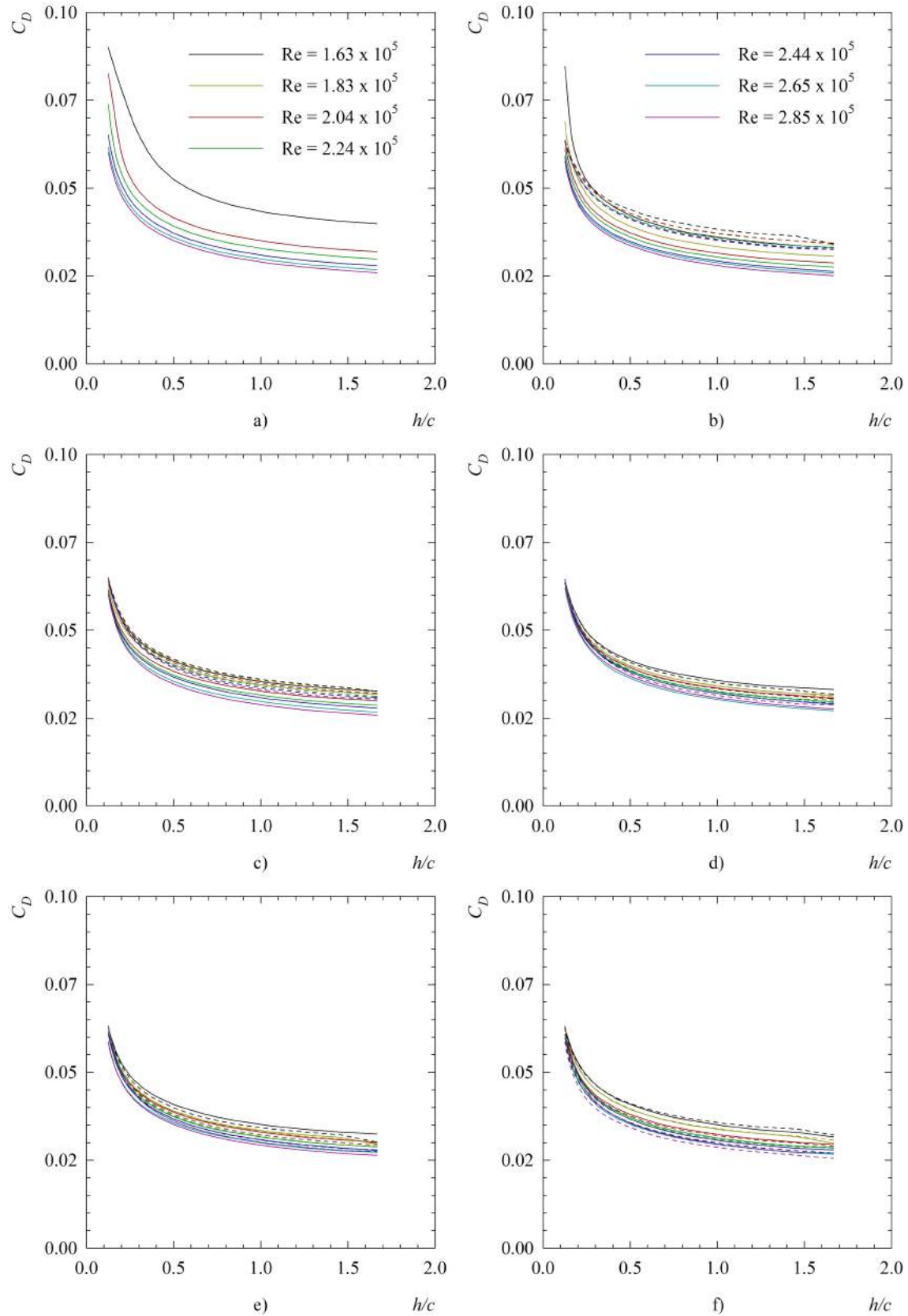


Figure 7.21: Drag measurements of CSW-SE wing at varying Reynolds number and ground clearance for a) clean wing, b) tripped at $x/c = 0.1$, c) tripped at $x/c = 0.2$, d) tripped at $x/c = 0.3$, e) tripped at $x/c = 0.4$ and f) tripped at $x/c = 0.5$ with serrated trip (solid line) and roughness trip (dashed line)

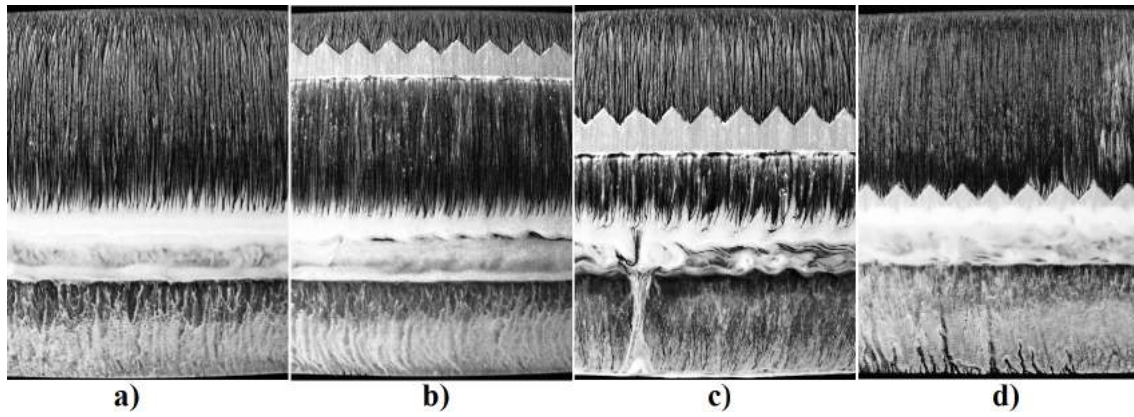


Figure 7.22: Suction-surface flow visualisation for a) free-transition and serrated trip placed at b) $x/c = 0.1$, c) $x/c = 0.3$, and d) $x/c = 0.5$ (Experimental, CSW-SE, $h/c = 0.3125$, $Re = 2.44 \times 10^5$, serrated trip, flow moving top to bottom)

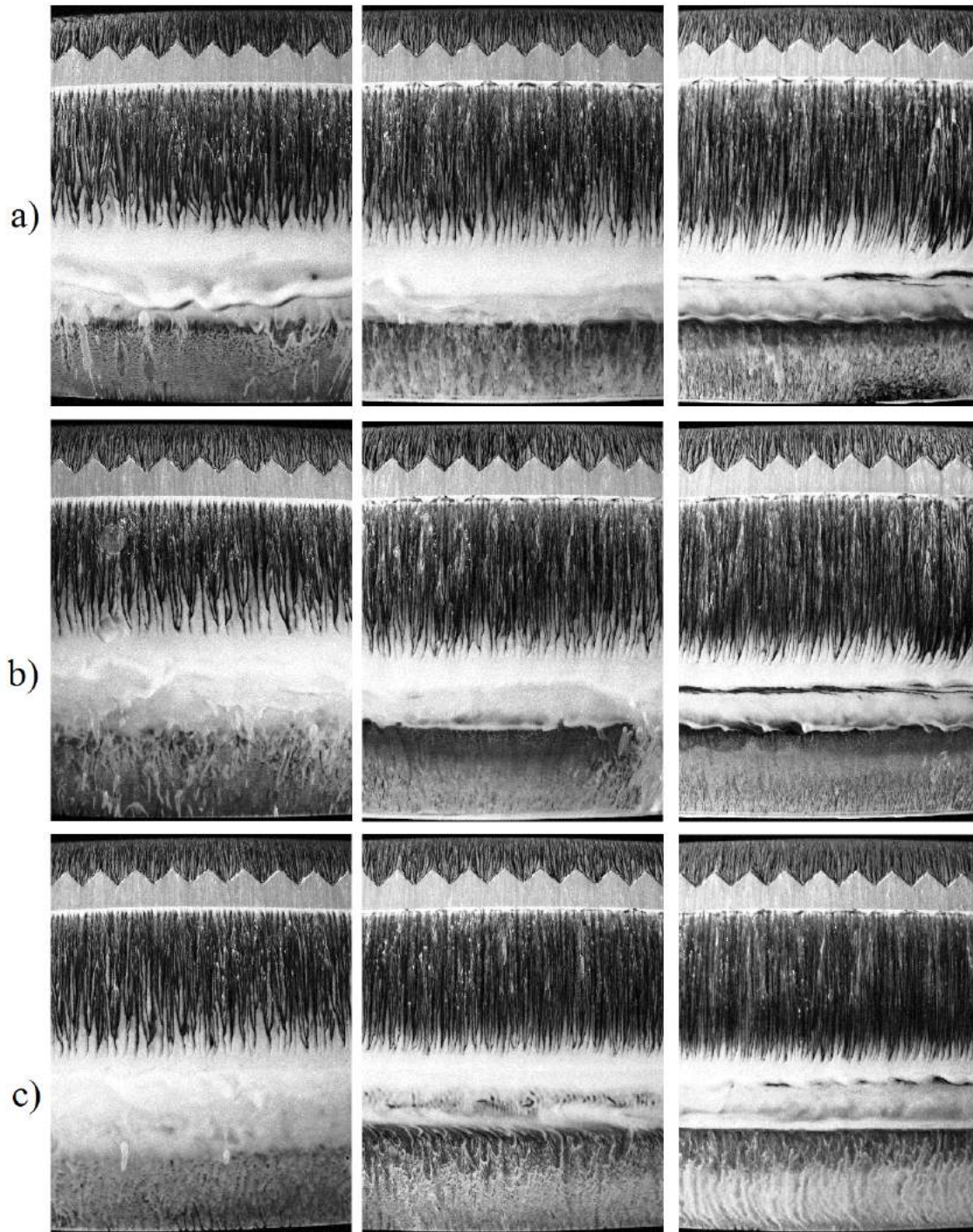


Figure 7.23: Suction-surface flow visualisation for serrated trip placed at $x/c = 0.1$ for a) $h/c = 0.3125$, b) $h/c = 0.225$, and c) $h/c = 0.125$ at $Re = 1.63 \times 10^5$ (left), $Re = 2.03 \times 10^5$ (middle), and $Re = 2.44 \times 10^5$ (right) (Experimental, CSW-SE, serrated trip, flow moving from top to bottom)

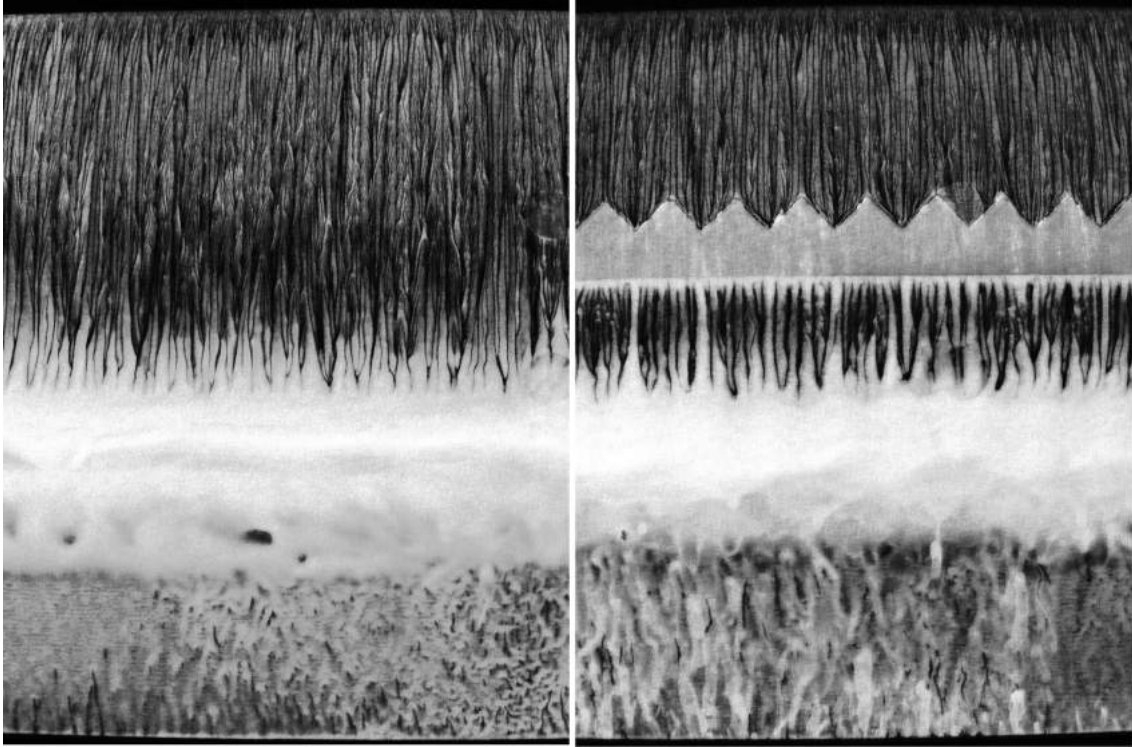


Figure 7.24: Comparison of free transition (left) and forced transition at $x/c = 0.3$ (right) for $h/c = 0.125$ at $Re = 1.63 \times 10^5$ (Experimental, CSW-SE, serrated trip, flow moving from top to bottom)

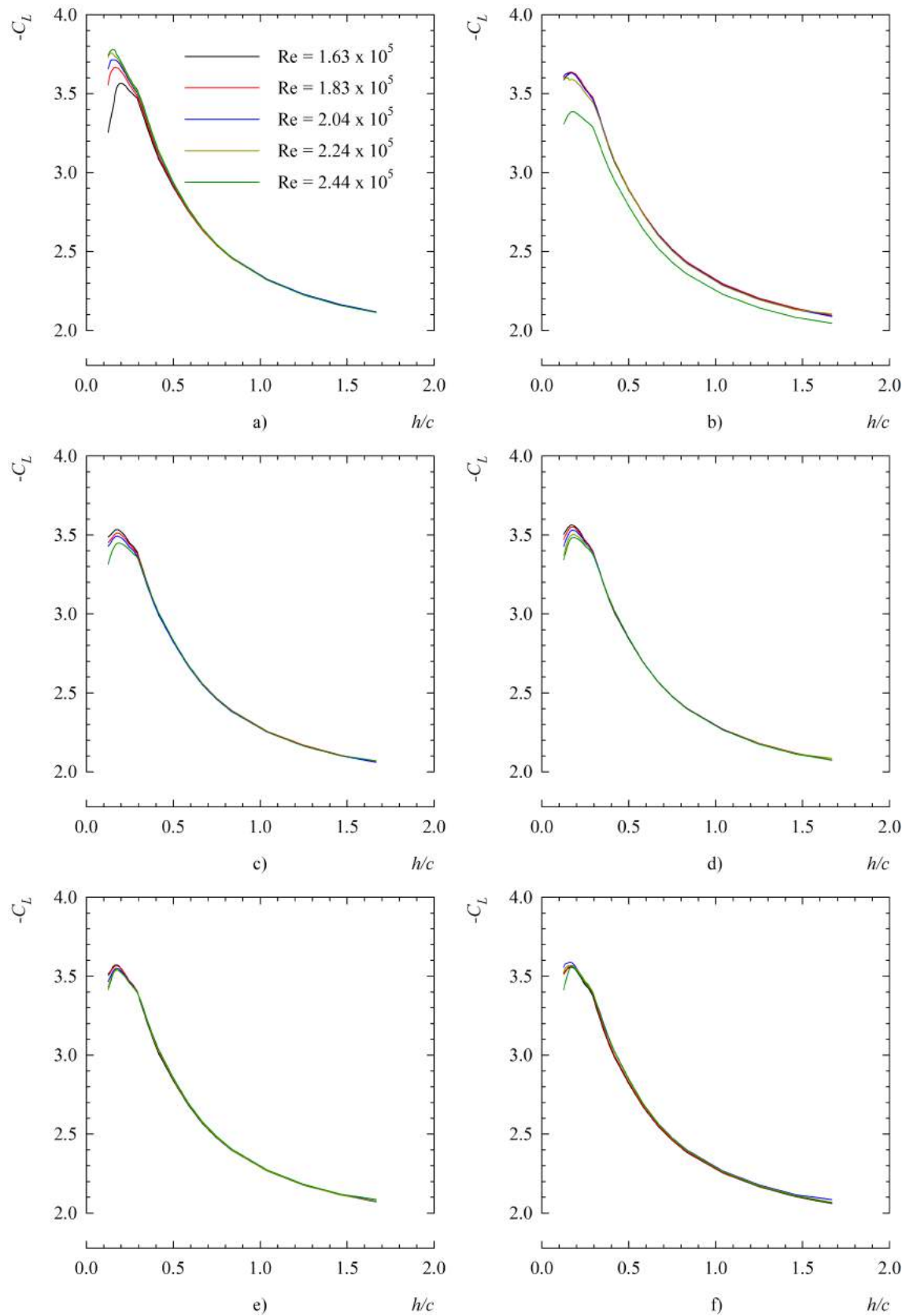


Figure 7.25: Downforce measurements of CSW-ME wing at varying Reynolds number, ground clearance and trip location a) clean wing, b) tripped at $x/c = 0.1$, c) tripped at $x/c = 0.2$, d) tripped at $x/c = 0.3$, e) tripped at $x/c = 0.4$ and f) tripped at $x/c = 0.5$ (Experimental, CSW-ME, serrated trip)

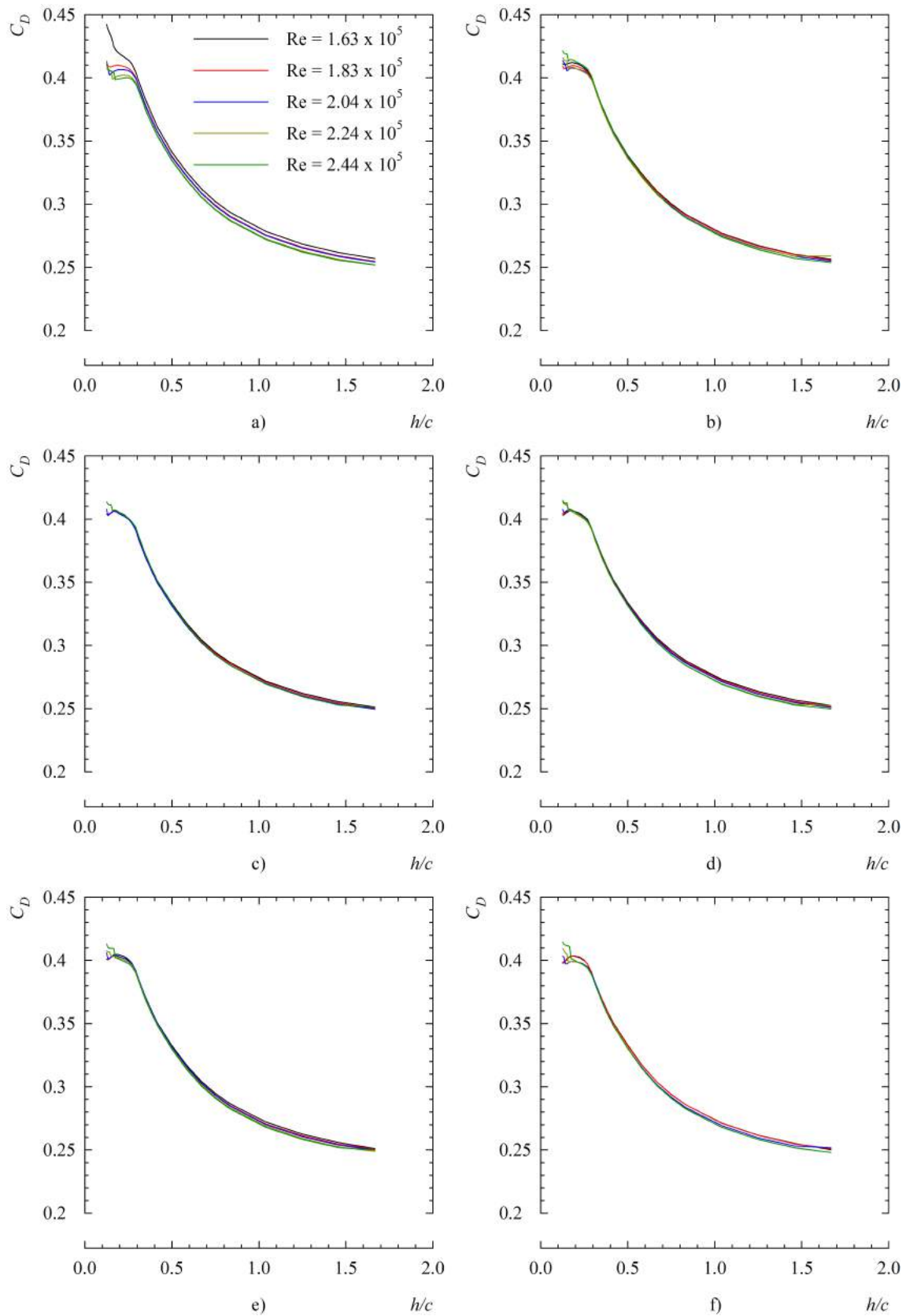


Figure 7.26: Drag measurements of CSW-ME wing at varying Reynolds number, ground clearance and trip location a) clean wing, b) tripped at $x/c = 0.1$, c) tripped at $x/c = 0.2$, d) tripped at $x/c = 0.3$, e) tripped at $x/c = 0.4$ and f) tripped at $x/c = 0.5$ (Experimental, CSW-ME, serrated trip)

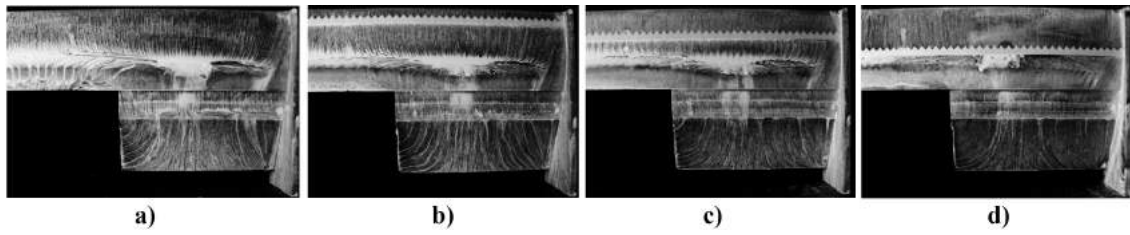


Figure 7.27: Suction-surface flow visualisation for a) free-transition and forced-transition wing with serrated trip at b) $x/c = 0.1$, c) $x/c = 0.3$, and d) $x/c = 0.5$ (Experimental, CSW-ME, serrated trip, $h/c = 0.3125$, $Re = 1.63 \times 10^5$, flow moving top to bottom)

Chapter 8

Conclusions

The present work has investigated the influence of boundary-layer transition and the laminar boundary-layer state on wings operating in ground effect in both two- and three-dimensional flow fields. This has been done empirically in a wind tunnel and numerically using Euler simulations and RANS CFD. Experimental techniques included force measurement, surface-flow visualisation, static pressure measurements, hot-film anemometry and LDA velocity measurements.

It was established in the literature that despite motorsport applications of wings in ground effect being at low Reynolds numbers, no consideration of laminar or transitional phenomena was reported. Whilst some authors reported the presence of a laminar separation bubble in their wind tunnel tests, its existence was usually omitted from any computational study. The purpose of this thesis was to answer the question "can the flow characteristics of a full-scale racing-car wing be replicated at model scale, and represented adequately by simulation?". The following objectives were set.

- Observe the transition mechanism and the influence of Reynolds number, ground

clearance and incidence for down-lifting wings operating in ground effect.

- Use the results to validate a numerical code such that other wing profiles can be investigated.
- Ascertain whether two-dimensional effects are relevant to a three-dimensional, single-element wing.
- Investigate how Reynolds number and transition affect wings of varying complexity, from single-element to highly-complex multi-element configurations.
- Evaluate the use of a transitional eddy-viscosity model and compare it to a fully-turbulent model.
- Determine the significance of laminar boundary-layer extent by forcing transition at various chordwise locations.
- Clarify the importance of boundary-layer transition and Reynolds-number scaling for wings in ground effect.

Each of these objectives has been met, and the key findings will be presented in the subsequent sections.

8.1 Characteristics of Transition at Varying Reynolds Number, Ground Clearance and Incidence

For all three-dimensional wing configurations and the two-dimensional wing at $\alpha \leq 3$ transition occurred through a laminar separation bubble. Hence, it can be concluded that separation-induced transition is the primary transition mechanism for wings in ground effect unless the freestream turbulence is high enough to cause a bypass transition, or incidence is sufficiently high that the boundary layer is destabilised to the point that transition occurs naturally in the attached shear layer. The typical structure of a separation bubble is

cited as the laminar boundary layer separating, transition occurring in the separated shear layer due to Kelvin-Helmholtz instabilities, then the boundary layer reattaching due to the wall-normal momentum transfer of the turbulent state. In the present work, however, hot-film results on the two-dimensional wing suggested that the boundary layer can still in the transitional state when it reattaches to the surface, but is instead still in the transitional state. Similarly, it may not be fully laminar when it detaches from the surface, due to perturbations beginning to generate turbulence. There is, however, insufficient wall-normal momentum transfer (and thus near-wall energy) to keep the boundary layer attached in the adverse pressure gradient. The state of the boundary layer prior to separation was found to have an effect on the length of the bubble, because when perturbations already exist in the 'laminar' boundary layer, amplification of these in the separated shear layer causes turbulence production to be higher.

The experiments and simulations revealed the principle effects of Reynolds number, ground clearance and incidence. The trends observed for each of these variables were applicable to the three-dimensional wings, other than incidence as that was not a variable for the three-dimensional wings. The effect of increasing Reynolds number was that downforce incrementally increased, drag incrementally reduced, and the separation bubble moved downstream and became shorter in length. This is because at higher Reynolds numbers the higher energy in the flow causes the laminar boundary layer to separate later, and the viscous effects reduce turbulence production such that the bubble becomes shorter.

As the ground clearance reduced, the bubble moved upstream and became shorter in length. This was attributed to the larger adverse pressure gradient causing earlier laminar separation, but with the higher flow velocity under the wing leading to an increase in Kelvin-Helmholtz instabilities that led to earlier turbulent reattachment. For the two-dimensional wing, the increase in turbulence production was evident by the lack of tran-

sitional frequencies in the hot-film PSD. Moreover, as the ground clearance reduced, both the influence of the bubble on the static pressure distribution and the Reynolds-number sensitivity of the static pressure distribution and hot-film PSD, dramatically increased. This showed that ground effect amplifies the effects of transition.

Reduced ground clearance causes the flow velocity under the wing to increase, and thus local Reynolds number to increase. Despite this, calculations in MSES showed that whilst the boundary layer is initially thinner because of these characteristics, the much larger adverse pressure gradient causes it to become thicker at the trailing edge than at higher tested ground clearances.

Overall, changes to the separation bubble and boundary-layer characteristics were observed as either Reynolds number or ground clearance was altered. This showed that ground proximity is of equal importance to Reynolds number when investigating transitional phenomena.

8.2 Two- and Three-Dimensional Effects

The effects observed in the two-dimensional tests were relevant to the three-dimensional wing, but the bubble's characteristics varied between spanwise locations on the 3D wings due to the spanwise pressure distribution being affected by either the endplates, nose, or inclusion of flaps.

Experiments conducted with the three-dimensional wing configurations showed that the more complex the wing, and equally the higher the downforce coefficient that it produced (for a given ground clearance and incidence), the less sensitive to Reynolds number it was. The single-element wing (CSW-SE) showed considerable Reynolds-number depen-

dency, both in the force-enhancement and force-reduction regions. The multi-element (CSW-ME) and F1 wings, however, only showed Reynolds-number sensitivity in the force-reduction region. At the lowest tested Reynolds number, the single-element wing displayed no force-enhancement at any ground clearance; a result of a fully-laminar separation occurring.

The multi-element and F1 wings generate significantly more downforce than the single-element wing, and thus much higher suction underneath the wing. This causes the influence of the bubble on the static pressure distribution to be smaller in percentage terms. Moreover, the higher flow velocity under the wing leads to a reduction in bubble height, reducing the force-enhancement mechanism of the bubble.

As the off-surface flow structures of the three wing configurations were considerably different, due to the multiple vortex structures generated by the increasingly complex wing configurations, it is proposed that these vortices aid the reduction in Reynolds-number dependency. This is because the multi-element and F1 wings are more reliant than the single-element wing on downforce produced by vortex-induced suction, rather than flow acceleration. Thus, force-enhancement due to the bubble altering the effective shape of the wing is not as influential on the multi-element and F1 wing's total downforce.

The two-dimensional wing also showed dependency on downforce capability in that when incidence was increased to $\alpha = 5^\circ$ the separation bubble was eliminated, and Reynolds-number sensitivity of the wing was considerably reduced. It must be concluded, therefore, that there is a clear relationship between the downforce capability of the wing configuration and its Reynolds-number dependency, in which the higher the downforce capability the lower is the Reynolds-number sensitivity.

8.3 Numerical Modelling of Transition

The CFD solution produced by the transition-sensitive RANS closure model $k - k_L - \omega$ showed that an eddy-viscosity model is capable of accurately predicting the aerodynamic forces and surface-flow structures observed in the wind-tunnel experiments. Although some tuning of the inlet conditions was required in order to give an accurate representation of wind-tunnel results (Appendix A.3), the model as implemented in Ansys FLUENT was capable of providing correct surface-flow features and force coefficients within 10% of wind tunnel results. This allowed the static pressure distribution to be observed for the single- and multi-element three-dimensional wings, for which experimental measurements were not available. This demonstrated how the bubble increased the integral of the static pressure distribution more for the single-element wing than for the multi-element wing in percentage terms. Thus, the downforce contribution of the bubble was proportionately larger on the single-element wing than the multi-element wing.

A clear Reynolds-number sensitivity was observed for the $k - k_L - \omega$ turbulence model when evaluating its use with the single-element wing, in which the lower the Reynolds number the poorer the prediction of both aerodynamic forces and surface-flow structures. The issue stemmed from the model's failure to represent correctly the turbulent reattachment phase of the separation bubble. This was, however, also due to the laminar separation point occurring further downstream than in the wind tunnel tests.

The model gave a more accurate prediction of the flow around the multi-element wing than the single-element wing, though the accuracy of the result also improved as the Reynolds number increased. The force prediction was within 4.1% of experimental results for all tested Reynolds numbers. Hence, although the prediction improved with increasing Reynolds number, even the poorest prediction was still acceptably accurate. The relative

lack of Reynolds-number sensitivity for the force prediction for the multi-element wing in CFD may, in part, be due to the lack of sensitivity of the geometry in general. The model performed better at outboard regions (ahead of the flaps) than in the central section, which showed that the model becomes worse in areas of greater adverse pressure gradients and turbulent flow separations.

The transition-sensitive $k - k_L - \omega$ model was compared to the fully-turbulent $k - \omega$ SST model, which has been employed regularly in previous wing-in-ground-effect studies. This was to evaluate the importance of including laminar and transitional flows in such simulations. Although there are still some deficiencies with the transitional model, when compared to the results of the fully-turbulent model it gave considerably better results. The key difference is that the fully-turbulent formulation of $k - \omega$ SST predicts no separation bubble and the turbulent boundary layer originates from the leading edge. This leads to dramatically incorrect surface structures for the $k - \omega$ SST result, which is characterised by trailing-edge separation. This occurs because a turbulent boundary layer has higher momentum loss. The adverse pressure gradient is too large and the boundary layer does not have enough energy (momentum) to overcome it. Therefore, whilst the prediction of drag was very good, almost comparable to that given by $k - k_L - \omega$, there was a large under-estimation of downforce, due to the lack of force-enhancement from the separation bubble combined with a trailing-edge separation that caused a reduction in circulation. Moreover, as total drag is implicitly defined by the boundary layer and the separation structure, neither of which $k - \omega$ SST models accurately, the accurate drag estimation is purely coincidental. In comparison to $k - k_L - \omega$, a thicker wake was predicted by $k - \omega$ SST, due to the trailing-edge separation, and significantly less turbulent kinetic energy. Moreover, differences in the pressure field between the two models were evident.

The front wing of a racing car is arguably the most important aerodynamic feature, as it

defines the operating conditions for features placed downstream of it. It is, therefore, important to represent the off-surface flow characteristics correctly, as well as accurately predicting aerodynamic forces. The significant differences between the results given by the two turbulence models highlights the consequence of correct turbulence model choice, particularly whether a fully-turbulent formulation is a realistic approach. For low Reynolds number conditions where laminar flow is a large component, the $k - k_L - \omega$ model offers a distinct advantage in accuracy. There are situations, however, such as high Reynolds number or bluff body flows in which laminar flow is not important and the additional equation of the $k - k_L - \omega$ model only increases the computational time. The present work highlights the importance of including laminar and transitional phenomena in simulations of wings operating in ground effect, and more importantly, shows that this can be accomplished using an eddy-viscosity model implemented in a commercial solver. Overall, the $k - k_L - \omega$ model has been shown to be appropriate for the simulation of separation-induced transition on a wing operating in ground effect.

8.4 The Influence of Laminar and Turbulent Boundary-Layer Extent

The fully-turbulent nature of the $k - \omega$ SST model results in a turbulent boundary layer encompassing the entire wing's surface, and led to a poor representation of wind tunnel results due to the lack of transition. Thus, the importance of the laminar boundary layer's presence was investigated by tripping the boundary layer to a turbulent state, by use of a roughness trip, at specific chordwise locations on all wing configurations in the wind tunnel. Forced transition was also simulated in MSES by specification of the transition point. The main effect of forcing transition was that the laminar separation bubble was

eliminated, and trailing-edge separation occurred; similarly to the $k - \omega$ SST CFD results. This led to a reduction in downforce and increase in drag, the magnitude of which was dependent on trip location. The more upstream the trip was placed, the greater the downforce loss and drag increase. The downforce reduction was shown by LDA velocity measurements and surface-pressure measurements on the two-dimensional wing, and force measurements on the three-dimensional wing. The loss of downforce was a result of lower velocity under the wing and thus less suction being applied to the surface. The increase in drag was indicated by LDA measurements to be due to a thicker wake with high velocity deficit occurring behind the two-dimensional wing; and by force measurements for the three-dimensional wings. It was found that each of these aspects was amplified as the trip location, and thus start of the turbulent boundary layer, moved upstream. This was again attributed to the momentum loss in the turbulent boundary layer causing the boundary layer to separate earlier. When the boundary-layer characteristics were investigated using MSES, it was clear that the Reynolds-number sensitivity stemmed from the laminar boundary layer portion, hence when the boundary layer was tripped the Reynolds-number sensitivity reduced.

Hot-film measurements on the two-dimensional wing showed that the roughness trips, formed from double-sided tape and Grit-60 sand, were capable of producing a turbulent boundary layer almost identical, in terms of where in the frequency range energy was contained, to that which formed post-transition in the free-transition case. The downside of the trips, however, was that they were observed to alter the surface pressure immediately before and after the trip. Simulations in MSES did not have this issue though, as the start of the turbulent boundary layer is simply specified. MSES tended to over-estimate surface suction values, due to it assuming all flow outside the boundary layer as inviscid, and thus momentum loss due to freestream turbulence being excluded. However, the trends

observed in both the wind tunnel, for both two-dimensional and three-dimensional wings configurations, and MSES as the trip was moved, incidence increased or Reynolds number varied were the same. The results, therefore, highlight the significant influence that the laminar boundary layer has on the aerodynamic characteristics of a wing in ground effect, and operating at Reynolds numbers relevant to motorsport applications. It has been shown that the presence of laminar boundary layers is not only significant, but also their influence on the wing is considerable in almost every aspect. Moreover, it is clear that the downforce and drag produced by the wing are proportional to the chordwise extent of the laminar and turbulent boundary layers, with downforce reducing and drag increasing the closer to the leading edge the turbulent boundary layer begins.

As the incidence of the two-dimensional wing was increased, the influence of the separation bubble on the pressure distribution reduced. This suggests that transitional effects become less apparent at higher downforce. This does not appear true, however, because the difference between the free- and forced-transition cases becomes larger as incidence increases. It must be concluded, therefore, that it is not the presence of the separation bubble, but rather the laminar boundary layer that is the important aspect. Turbulent boundary layers are often cited as helping to suppress trailing-edge separation due to wall-normal momentum transfer, and in many cases, such as golf balls, cyclist and speed-skater helmets and suits, and sailplanes (in the form of turbulators) this is true. For the present work, however, the turbulent boundary layer is clearly contributing to the occurrence of trailing-edge separation. This shows that the inclusion of laminar boundary layers in any computational model is important, especially since the Reynolds numbers at which these laminar boundary layers occur are relevant to full-scale applications.

Although the act of forcing transition is usually conducted in order to replicate the earlier transition point of higher Reynolds number flow, the difficulty is that other flow as-

pects, such as energy in the flow, boundary-layer thickness, Reynolds stresses, and shear stresses, cannot also be replicated. Thus, thicker boundary layers will always be produced, turbulence production reduced, and separation more likely at lower Reynolds numbers. For the lowest tested Reynolds number cases of the three-dimensional wings, the trips were capable of increasing the downforce and lowering the drag at all ground clearances, particularly at low ground clearances and for the single-element configuration. This was, however, due to the turbulent boundary layer being capable of remaining attached longer than the laminar boundary layer, which separated near to the start of the adverse pressure gradient and never reattached. Whilst this led to the force measurement being similar to a higher Reynolds number case, the surface-flow structures were completely different. Hence, it can be concluded that while it may be possible to match the forces by tripping the boundary layer, the manner in which they are generated may not match the higher Reynolds number case.

Tests with a serrated trip showed that, unlike the case with the roughness trip, it is possible to manipulate the boundary layer by de-stabilising it such that the wing acts similarly to a higher Reynolds number in terms of both force and flow structures. Although the serrated trip was capable of making the lowest tested Reynolds number case act like a higher Reynolds number case, it was again due to the boundary being manipulated into not undergoing a fully-laminar separation. Thus, it appears to be a very specific circumstance rather than a general principle that could be applied to all Reynolds numbers.

The serrated trips enhanced the performance of the single-element wing at low Reynolds numbers, and particularly at low ground clearances, by reducing the critical ground clearance. Whilst it offered no improvement at higher Reynolds number, it had no appreciable performance reduction. This may have been a result of the trip being too thick for the thinner boundary layer that forms at Reynolds numbers, thus future tests with a similar

shape, but thinner, trip would be beneficial.

8.5 Importance of Transition for Wings in Ground Effect

The present work has shown that the presence of the laminar boundary layer and separation bubble are important aspects for wings operating in ground effect because the effects of transition, whether free or forced, were found to be amplified as ground clearance reduced. It has been shown that laminar and transitional phenomena can be simulated numerically through use of eddy viscosity-based closure models inside a commercial RANS solver. Such an approach estimates aerodynamic forces and surface-flow structures much more accurately than fully-turbulent models. Given that a laminar separation bubble was shown to be present for Reynolds numbers relevant to full-scale racing cars, can be modelled accurately using RANS computations, and exists on both fore-loaded and aft-loaded aerofoil profiles, it is concluded that it is an important feature of wings operating in ground effect. Laminar separation bubbles, therefore, need to be included in any design or analysis of wings operating in ground effect at Reynolds numbers typical of both full-scale and model-scale monoposto racing-car front wings.

Whilst some previous studies have observed the presence of a laminar separation bubble on the suction surface of a wing operating in ground effect, no investigation into the importance or influence of this phenomenon was discussed. Moreover, some computational studies had to make use of forced-transition experimental data in order to get their CFD simulations to correlate accurately, thus indicating that laminar and transitional regimes were having a considerable influence on the flow.

Whilst the GA(W)-1 profile, which has formed the basis of most models used in this thesis, is an aft-loaded aerofoil, and thus tends to promote laminar flow, other results have

shown separation-induced transition to be important for other geometries too. Firstly, the fore-loaded Tyrrell-026 wing demonstrated a separation bubble in the lower tested Reynolds number range in MSES. Additionally, the F1 wing features an aft-loaded aerofoil in the central region of the main-plane, as this is the regulation-defined symmetrical 'neutral' aerofoil section, but a fore-loaded profile in the outer regions. Hence, flow visualisation of the F1 wing showed the separation bubble moving significantly upstream in the outboard region. The fact that a separation bubble was observed for all three-dimensional wing configurations, including the fore-loaded aerofoil sections, shows that it is an important aspect of wings in ground effect.

The aim of this thesis was to establish the significance of laminar and transitional flow phenomena for wings operating in ground effect. To do this it was important first to demonstrate that laminar and transitional flow phenomena were applicable to both model-scale and full-scale Reynolds numbers. This was accomplished through use of a super-scale model that, for the first time, considered a wing in ground effect at full-scale Reynolds number (rather than at model scale). Not only was it shown that a laminar separation bubble still formed at full-scale Reynolds numbers, but also that laminar flow was more influential as ground clearance reduced. As a result, laminar and transitional flow phenomena are more important for wings in ground effect than for their out-of-ground-effect equivalents at the same Reynolds number.

In addition to demonstrating the significance of laminar and transitional flow phenomena at full-scale Reynolds numbers for a two-dimensional wing, this study has provided the first published evidence that laminar separation bubbles will form on model-scale racing-car wings of both simplified and F1-specification three-dimensional geometries. Hence, the importance of transition is not limited to the two-dimensional case, but is also significant for highly-complex and high-downforce capability wings operating in ground effect.

Chapter 9

Recommendations for Future Work

9.1 Freestream turbulence intensity

Simulations in MSES (Section 5.2.4) showed that the freestream turbulence intensity altered the length of the laminar separation bubble, and thus altered the downforce and drag coefficients. It would be interesting to use turbulence-inducing grids to experimentally observe the effect of freestream turbulence. Moreover, observing the transition characteristics in an unsteady flowfield, such as downstream of a sinusoidal gust generator [121] would be of interest.

9.2 Further Measurements on Three-Dimensional Wings

Although the measurement techniques for the three-dimensional wings were limited to force measurement and surface-flow visualisation, the results still revealed interesting characteristics in terms of Reynolds-number sensitivity. It was postulated that the higher downforce capability wings (CSW-ME and F1 wings) had reduced Reynolds-number sen-

sitivity due to their increased reliance on vortex-induced suction. Taking static pressure distributions at various spanwise locations and measuring vorticity in the wake with a PIV or LDA system would allow this to be investigated. Moreover, using hot-film sensors to provide further information on the boundary-layer characteristics, in a similar manner to the 2D wing, would be beneficial.

9.3 High Resolution Hot-Film Measurements

The hot-film gauges were capable of indicating the presence of a laminar separation bubble and the boundary-layer state. If a high spatial resolution hot-film array were to be used, then accurate locations of separation and reattachment could be determined, and thus accurate size and location of the laminar separation bubble be found. It would be advantageous to apply this technique to both the two-dimensional and three-dimensional wings.

9.4 LES Simulation

The transition-sensitive RANS model, $k - k_L - \omega$, was shown to be more accurate at predicting experimental force measurements and surface-flow visualisation. Whilst detached eddy simulation (DES) still relies on RANS in the near-wall region, large eddy simulation (LES) solves all but the smallest eddies, and as such is capable of simulating laminar and transitional flow. The main issue with $k - k_L - \omega$ was that the inlet conditions needed to be tu

List of Published Works

The contents of this thesis have been used to produce articles for journals and conferences.

Journal articles:

- L. S. Roberts, M. V. Finnis & K. Knowles, *Characteristics of Boundary-Layer Transition and Reynolds-Number Sensitivity of Three-Dimensional Wings of Varying Complexity Operating in Ground Effect*. ASME J. Fluids Eng., Vol 138, pp. 091106 1-10, 2016.
- L. S. Roberts, M. V. Finnis & K. Knowles, *Forcing Boundary-Layer Transition on a Single-Element Wing in Ground Effect*. ASME J. Fluids Eng., Vol 139(10), pp. 101205 1-12, 2017.
- L. S. Roberts, M. V. Finnis, K. Knowles & N. J. Lawson, *Forcing Boundary-Layer Transition on an Inverted Airfoil in Ground Effect*. J. Aircraft, doi:10.2514/1.C034304 (In Press), 2017.
- L. S. Roberts, M. V. Finnis & K. Knowles, *Modelling Boundary-Layer Transition on Wings Operating in Ground Effect at Low Reynolds Numbers*. Proc. IMechE Pt. D, J. Automobile Eng., (Under Review), 2017.

Conferences papers:

- L. S. Roberts, M. V. Finnis & K. Knowles, *Investigation of Forcing Boundary Layer Transition on a Single-Element Inverted Wing in Ground Effect*. In IMechE International Vehicle Aerodynamics Conference. Paper C1385 015. Loughborough, UK, 2014.
- L. S. Roberts, M. V. Finnis, K. Knowles & N. J. Lawson, *Forcing Boundary-Layer Transition on an Inverted Airfoil in Ground Effect at Varying Incidence*. In 34th AIAA Applied Aerodynamics Conference. Paper AIAA 2016-3724. Washington D.C, USA, 2016.

Other related work not included in this thesis:

- J. Correia, L. S. Roberts, M. V. Finnis & K. Knowles, *Scale Effects on a Single-Element Inverted Wing in Ground Effect*. The Aeronautical Journal, Vol. 118(1205), pp. 797-809, 2014.
- J. Correia, L. S. Roberts, M. V. Finnis & K. Knowles, *Aerodynamic Characteristics of a Monoposto Racing Car Front Wing Operating in a Turbulent Wake*. In IMechE International Vehicle Aerodynamics Conference, Loughborough, UK, 2014.
- L. S. Roberts, J. Correia, M. V. Finnis & K. Knowles, *Aerodynamic Characteristics of a Wing & Flap Configuration in Ground Effect & Yaw*. Proc. IMechE Pt. D, J. Automobile Eng., Vol. 230(6), pp 841-854, 2015.
- L. S. Roberts, J. Correia, M. V. Finnis & K. Knowles, *Aerodynamic Characteristics of a Monoposto Racing-Car Front Wing in Yaw*. Aerovehicles 1, Bordeaux, France, 2014 (Oral presentation only).

Appendix A

Appendices

A.1 Force Coefficient Uncertainty

As the same wind tunnel, model and equipment was used by Correia [108], the same process for the calculation of uncertainty has been used.

The AeroTech force balance error given by the manufacturer is given in Table A.1.

Table A.1: AeroTech force balance uncertainty in measured coefficient

Drag	Lift	Pitch	Roll	Side	Yaw
0.078%	0.016%	0.037%	0.055%	0.067%	0.093%

The total uncertainty of force coefficients, which is given in Table A.2, is made up of uncertainties relating to the accuracy of the model position, wind tunnel dynamic pressure, and force balance error. The total uncertainty value for a 95% confidence level for each component was calculated using the root-mean-square method.

The uncertainty of each setup parameter was calculated by measuring the forces during two runs between which a single variable was altered. The difference in measured forces

Table A.2: Contribution to total uncertainty of force coefficients

	Drag	Downforce	Yaw	Pitch	Roll	Side
Balance	0.00018	0.00034	0.00567	0.00012	0.00017	0.00004
h/c	0.00097	0.00823	0.06139	0.01563	0.01486	0.00196
Mainplane incidence	5.43E-06	8.19E-05	1.60E-05	4.05E-06	3.27E-06	6.39E-07
Flap angle	0.00189	0.00750	0.05415	0.00123	0.00127	0.00038
Freestream velocity	0.00093	0.00840	-0.03377	-0.00022	-0.00015	-0.00005
Roll	0.00002	0.00019	0.02054	0.00028	0.00048	0.00007
Yaw	0.00003	0.00033	0.00246	0.00564	0.01482	0.00202
Total	0.00233	0.01396	0.09112	0.01667	0.02103	0.00284

between the two runs was assumed to be a linear relationship and combined with the setup accuracy of that variable to generate an uncertainty. For example, the yaw angle was varied between 0° and 5° over two runs; this gave a difference in force coefficient per degree of yaw angle. This was then multiplied by the uncertainty in the yaw angle measurement (which is based on the accuracy of the inclinometer) to give the uncertainty due to yaw angle. The variation of the tested settings is given in Table A.3 along with the measurement uncertainty and source of the measurement uncertainty.

Table A.3: Variation of each variable between two tests and uncertainty of variable measurement

Variable	Variation	Measurement uncertainty	Source
h/c	+0.001984	$\pm 8.35\text{E-}06$	Height gauge accuracy
Mainplane incidence	+0.2127°	$\pm 1.7\text{E-}04$	Calculated
Flap angle	+10.2°	$\pm 0.1^\circ$	Inclinometer accuracy
Freestream velocity	+0.2%	$\pm 0.2\%$	Sensor accuracy
Roll	+3°	$\pm 4\text{E-}3^\circ$	Calculated
Yaw	+5°	$\pm 2.1\text{E-}2^\circ$	Calculated

A.2 Evaluation of Appropriate Inlet Conditions for $k - k_L - \omega$

During initial testing, it was discovered that the $k - k_L - \omega$ model is sensitive to the domain inlet values, in particular the turbulence quantities. These quantities are usually adapted in order to recreate real conditions, as their values are generally estimates. For external flows, the values of turbulent kinetic energy (k) and specific dissipation rate (ω) are generally estimated using turbulence intensity, which is usually the freestream turbulence level of the wind tunnel used in related experiments, and a turbulence length scale or turbulent viscosity ratio.

Simulations in the present work initially made use of turbulence intensity of 0.5% ($k = 0.03375m^2s^{-2}$) at the velocity inlet. The turbulence intensity would then dissipate such that it would be 0.3% at the leading edge of the model; the same as the current wind tunnel experiments. It was found, however, that this led to insufficient turbulence being generating in the separated shear layer of the laminar separation bubble, which led to poor representation of the turbulent reattachment process. Different turbulent length scales were tested in an attempt to reduce the dissipation rate and allow more turbulence to be generated, but this was unsuccessful. The prescription of turbulence intensity (Tu), however, had a much more profound effect on the result, as this sections describes. The inlet turbulence intensity was increased to 3% ($k = 1.215m^2s^{-2}$) and 5% ($k = 3.375m^2s^{-2}$) in order to boost turbulence production. The large increase was made because it was hoped that a large change would occur and the effect of the parameter would be easily identifiable.

Figure A.1 shows the suction-surface flow structures produced by the three inlet condi-

tions for the single- and multi-element wings in comparison to wind tunnel flow visualisation. In general, only the $Tu = 3\%$ case produces a correct separation bubble that is representative of the wind tunnel results. The $k - k_L - \omega$ model uses a function known as "shear sheltering". Walters & Cokljat [46] explain that this function damps turbulence in thin regions of high vorticity, such as non-linear turbulence breakdown mechanisms that occur in the pre-transitional boundary layer. In the turbulent boundary layer, the function is restricted to the viscous sub-layer. Turner [117] suggested, however, that shear sheltering is inappropriate for separation-induced transition as it inhibits turbulence production in the separated shear layer. Observation of the results given in the present study, where higher turbulence intensities than those found in the wind tunnel were required in order for the model to produce enough turbulent kinetic energy to give full turbulent-boundary-layer reattachment, support this.

The downforce and drag coefficients generated by each inlet condition for the single- and multi-element wing configurations are given in Table A.4. For the single-element wing, there is little difference between the $Tu = 0.5\%$ and $Tu = 3\%$ cases, with both giving similarly accurate predictions of experimental downforce values. The $Tu = 5\%$ case, however, has a much poorer prediction of drag coefficient than $Tu = 3\%$ but better than $Tu = 0.5\%$. The $Tu = 3\%$ case gives the best prediction of downforce and drag. It must be noted, however, that the drag force is so small that these differences are all mostly negligible. For the multi-element wing, all three cases produce similarly accurate results. Based on comparison of the surface streamlines it would be assumed that the $Tu = 3\%$ case would give the best prediction of aerodynamic forces, as the surface-flow structures it produced were most similar to the flow-visualisation experiments. It is, however, the $Tu = 5\%$ case that contains the closest prediction to the wind tunnel force values. This is likely to be a result of the lack of force-enhancement from the bubble compared to $Tu =$

3% causing reduced downforce and drag to be produced.

The shear stress distribution at two spanwise locations given in Figure A.2 shows the separation and reattachment locations in more detail. In general, as the turbulence intensity is increased the separation point moves downstream. For the single-element wing with $Tu = 0.5\%$ the boundary layer does not reattach, and what can be considered as a long bubble is produced. The $Tu = 3\%$ case, produces sufficient turbulence in the separated shear layer that turbulent reattachment occurs. This produces a bubble of similar size to the experiments based on the surface flow visualisation. The $Tu = 5\%$ case gives the most downstream laminar separation point and shortest length bubble. Moreover, it generates significantly more shear stress upon reattachment. In comparison to the wind tunnel results, if the earlier separation point of $Tu = 0.5\%$ were combined with the earlier fully-turbulent reattachment of $Tu = 5\%$, it would give a result very close to the experiments in terms of bubble size and location. This highlights why $Tu = 3\%$ was chosen for this study, as it gives the best compromise between bubble position and reattachment characteristics.

For the multi-element wing, $Tu = 0.5\%$ shows poor reattachment characteristics at the centre of the wing ($z/s = 0$). At the outer-span of the wing ahead of the flap, however, the

Table A.4: Comparison of downforce and drag generated by varying inlet turbulence intensity ($k - K_L - \omega$ RANS, $Re = 2.44 \times 10^5$, $h/c = 0.3125$)

Configuration	Case	C_L	C_L error (%)	C_D	C_D error (%)
Single-Element (CSW-SE)	Experiment	0.627	-	0.0431	-
	$Tu = 0.5\%$	0.681	7.93	0.0395	-9.20
	$Tu = 3\%$	0.676	7.22	0.0399	-8.01
	$Tu = 5\%$	0.744	15.73	0.0396	-8.92
Multi-Element (CSW-ME)	Experiment	3.471	-	0.389	-
	$Tu = 0.5\%$	3.523	1.47	0.403	3.54
	$Tu = 3\%$	3.536	1.85	0.403	3.50
	$Tu = 5\%$	3.527	1.59	0.402	3.31

prediction is better because full reattachment occurs. For $Tu = 3\%$ at $z/s = 0$, turbulent reattachment occurs, but a second separation occurs at $x/c = 0.97$, as shown in Figure A.1. At the outboard plane ($z/s = 2/3$) the turbulent boundary layer remains fully attached for all cases. For $Tu = 5\%$ no separation bubble is formed. Instead the instabilities grow sufficiently in the laminar boundary layer such that transition occurs in the attached boundary layer prior to the laminar separation point.

Overall, it can be seen that as the freestream turbulence intensity is increased the laminar separation point moves downstream, the bubble becomes shorter in length, and turbulence production is increased. The model was shown to be sensitive to inlet conditions, specifically the value of turbulence intensity. $Tu = 3\%$ gives the most accurate representation of the flow structures observed in experiments, and thus was used throughout this study. The increased turbulence intensity requirement stems from the model producing insufficient turbulent kinetic energy in the separated shear layer, and as a result, reattachment of the turbulent boundary layer being inhibited.

Although it has been possible to produce an accurate representation of the wind tunnel results with the $k - k_L - \omega$ model, the main concern is that because the model produces insufficient turbulent kinetic energy to give reattachment at wind tunnel freestream turbulence intensity, it has to be artificially increased. Therefore, for cases where wind tunnel results are not available before CFD is undertaken, it may be difficult to obtain accurate predictions as there is no way of validating, and in some way calibrating, the model.

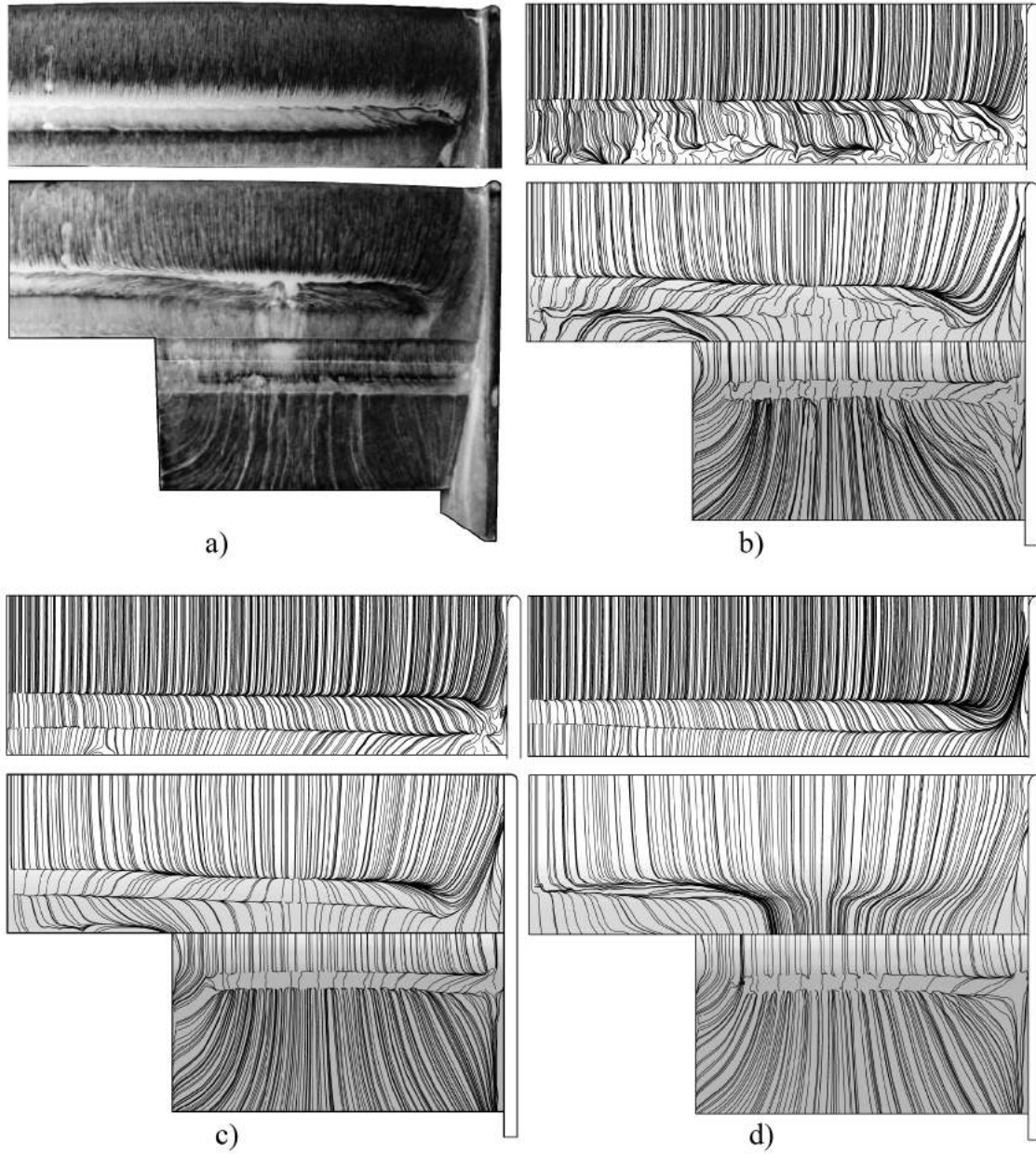


Figure A.1: Comparison of a) experimental flow visualisation to CFD surface streamlines, generated by shear stress, with inlet turbulence intensity of b) $Tu = 0.5\%$, c) $Tu = 3\%$, and d) $Tu = 5\%$ ($Re = 2.44 \times 10^5$, $h/c = 0.3125$, flow from top to bottom, CSW-SE (top) and CSW-ME (bottom) in each case)

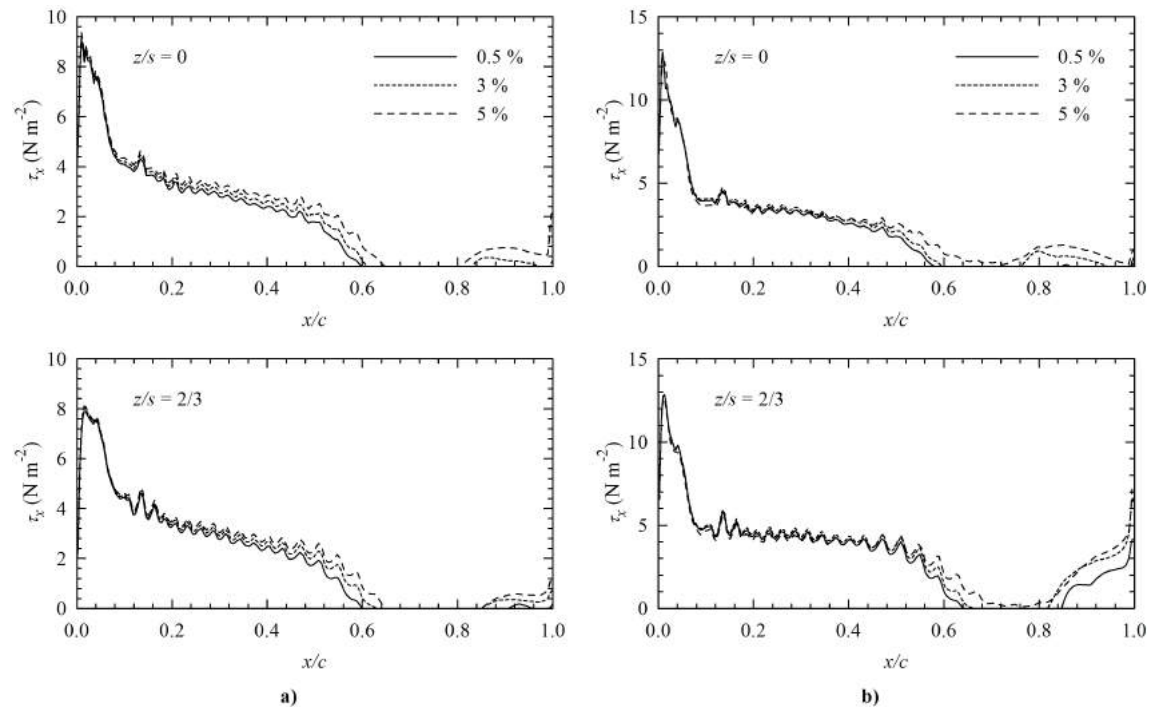


Figure A.2: x-component of shear stress on suction surface of a) single-element wing, and b) multi-element wing at $z/s = 0$ and $z/s = 2/3$ for varying inlet turbulence intensity ($Re = 2.44 \times 10^5$, $h/c = 0.3125$, RANS $k - k_L - \omega$)

A.3 CFD Parameters

Table A.5: IcemCFD mesh settings

Global mesh parameters	
Scale factor	1
Max element	100mm
Min element size	0.1mm
Shell meshing parameters	
Mesh type	All tri
Mesh method	Patch independent
Prism meshing parameters	
Growth law	Exponential
Initial height	0.006
Height ratio	1.22
Number of layers	21

Table A.6: IcemCFD part settings

	Max size	Prism Layers
Domain/Floor-Inner	3	10
Domain/Floor-Outer	50	0
Domain-Inlet	100	0
Domain/Left	100	0
Domain/Outlet	100	0
Domain/Symmetry-Outer	5	0
Domain/Symmetry-Inner	50	0
Domain/Top	100	0
Fluid		
Wing/Mainplane	1	21
Wing/Flap	1	21
Wing/Endplate	1	21
Wing/Pylon	1	21
Body	3	10

Table A.7: Fluent solution methods

Pressure-velocity coupling	
Scheme	SIMPLE
Spatial discretisation	
Gradient	Green-Gauss node based
Pressure	Second order upwind
Momentum	Second order upwind
Turbulent kinetic energy	Second order upwind
Laminar kinetic energy	Second order upwind
Specific dissipation rate	Second order upwind

Table A.8: Fluent under-relaxation factors for $k - k_L - \omega$ simulations

Pressure	0.3
Density	1
Body forces	1
Momentum	0.5
Turbulent kinetic energy	0.4
Laminar kinetic energy	0.4
Specific dissipation rate	0.4
Turbulent viscosity	0.4

References

- [1] C. Smith, *Tune to Win: : The Art and Science of Race Car Development and Tuning*. Fallbrook, CA, Aero Publisher, 1978.
- [2] B. Agathangelou & M. Gascoyne, *Aerodynamic Considerations of a Formula 1 Racing Car*. Technical Report SAE 980399, Society of Automotive Engineers, Warrendale, PA, USA, February, 1998.
- [3] L. F. Richardson, *Weather Prediction by Numerical Process*. Cambridge University Press, 1922.
- [4] H. Schlichting & K. Gersten, *Boundary-Layer Theory*. 8th Edition, Springer-Verlag Berlin Heidelberg, 2000.
- [5] G. B. Schubauer & H. K. Skramstrad, *Laminar-Boundary-Layer Oscillations and Transition on a Flat-Plate*. NACA Report 909, 1948.
- [6] M. Nishioka, S. Iida & Y. Ichikawa, *An Experimental Investigation of the Stability of Plane Poiseuille Flow*. Journal of Fluid Mechanics, Vol. 72(4), pp. 731-751, 1975.
- [7] H. W. Emmons, *The Laminar-Turbulent Transition in a Boundary Layer Part I*. Journal of Aeronautical Sciences, Vol. 18(7), pp. 400-498, 1951.

- [8] H. W. Emmons & A. E. Bryson, *The Laminar-Turbulent Transition in a Boundary Layer Part II*. Proceedings of First US Congress of Applied Mechanics (June 1951, Chicago, IL), ASME, pp. 859-868, 1952.
- [9] G. B. Schubauer & P. S. Klebanoff, *Contributions on the Mechanics of Boundary-Layer Transition*. NACA Report 1289, 1956.
- [10] H. W. Bennett, *An Experimental Study of Boundary Layer Transition*. Office of Naval Research, Neenah, Wisconsin, 1953.
- [11] A. C. N. Tan & J. D. Auld, *Study of Laminar Separation Bubbles at Low Reynolds Number Under Various Conditions*. 11th Australasian Fluids Mechanics Conference, University of Tasmania, Hobart, Australia, 1992.
- [12] P. R. Spalart & M. K. Strelets, *Mechanisms of Transition and Heat Transfer in a Separation Bubble*. Journal of Fluid Mechanics, Vol. 403, pp. 239-349, 2000.
- [13] Z. Yang & P. R. Voke, *Large-Eddy Simulation of Boundary-Layer Separation and Transition at a Change of Surface Curvature*. Journal of Fluid Mechanics, Vol. 439, pp. 305-333, 2001.
- [14] R. Hain, C. J. Kahler & R. Radespiel, *Dynamics of Laminar Separation Bubbles at Low-Reynolds-Number Aerofoils*. Journal of Fluid Mechanics, Vol. 630, pp. 129-153, 2009.
- [15] H. P. Horton, *Laminar Separation Bubbles in Two and Three Dimensional Incompressible Flow*. PhD Thesis, University of London, 1968.
- [16] X. F. Zhang, A. Mahallati & S. A. Sjolander, *Hot Film Measurements of Boundary Layer Transition, Separation and Reattachment on a Low-Pressure*

- Turbine Airfoil at Low Reynolds Numbers*. AIAA Paper 2002-3643, 38th AIAA/ASME/SAE/ASEE Joint Propulsion Conference & Exhibit, Indianapolis, Indiana, 2002.
- [17] I. Tani, *Low-speed Flows involving Bubble Separations*. Progress in Aerospace Sciences, Vol. 5, pp. 70-103, 1974.
- [18] R. E. Mayle, *The Role of Laminar-Turbulent Transition in Gas Turbine Engines*. Journal of Turbomachinery, Vol. 113 pp. 509-537, 1991.
- [19] L. Brandt, P. Schlatter & D. S. Henningson *Transition in Boundary Layers Subject to Free-Stream Turbulence*. Journal of Fluid Mechanics, Vol. 517, pp. 167-198, 2004.
- [20] P. S. Klebanoff, *Effect of Free-Stream Turbulence on a Laminar Boundary Layer*. Bulletin Am. Phys. Soc, Vol. 10(11), pp. 1323 1971.
- [21] D. Arnal & J. Juillen, *Contribution Experimentale a l'Etude Receptivite d'une Couche Limite Laminaire, a la Turbulence de l'Ecoulement General*. ONERA Rapport Technique No 1/5018 AYD, 1978.
- [22] J. M. Kendall, *Experiments on Boundary-Layer Receptivity to Freestream Turbulence*. AIAA Paper 98-0530, 1998.
- [23] X. Wu & M. Choudhari, *Effects of Long-Wavelength Klebanoff Modes on Boundary-Layer Instability*. Center for Turbulence Research Annual Research Briefs, NASA, Stanford Vol. 19, pp. 305-316, 2001.
- [24] T. A. Zaki, *From Streaks to Spots and on to Turbulence: Exploring the Dynamics of Boundary Layer Transition*. Flow, Turbulence and Combustion, Vol. 91(3), pp. 451-473, 2013.

- [25] R. Narasimha & K. R. Sreenivasan, *Relaminarization of Fluid Flows*. Advances in Applied Mechanics, Vol. 19, pp. 221-309, 1979.
- [26] B. J. Abu-Ghannam and R. Shaw, *Natural Transition of Boundary Layers - The Effects of Turbulence, Pressure Gradient and Flow History*. J. Mechanical Engineering Science, Vol. 22(5), pp. 213-228, 1980.
- [27] J. H. Watmuff, *Evolution of a Wave Packet into Vortex Loops in a Laminar Separation Bubble*. Journal of Fluid Mechanics, Vol. 397, pp. 119-169, 1999.
- [28] Y. Lian and W. Shyy, *Laminar-Turbulent Transition of a Low Reynolds Number Rigid or Flexible Airfoil*. AIAA Journal, Vol. 45(7), pp. 1501-1513, 2007.
- [29] D. J. Hall and J. C. Gibbings, *Influence of Stream Turbulence and Pressure Gradient upon Boundary Layer Transition*. J. Mechanical Engineering Science, Vol. 14(2), pp. 134-146, 1972.
- [30] S. Yoshioka, S. Kikuchi, F. Ohta, T. Kato, J. Song and Y. Kohama, *Measurement of Ground Effect and Boundary-Layer Transition by Towing Wind Tunnel*. Fluid Dynamics Research, Vol. 41(2), 021408 (12pp), 2009.
- [31] D. G. Mabey, *A Review of Scale Effects in Unsteady Aerodynamics*. Prog. Aerospace Sci, Vol. 1991, pp. 273-321, 1993.
- [32] S. S. Diwan and O. N. Ramesh, *Laminar Separation Bubbles: Dynamics and Control*. Sadhana, Vol. 32, pp. 103-109, 2007.
- [33] P. H. Hoffman, K. C. Muck and P. Bradshaw, *The Effect of Concave Surface Curvature on Turbulent Boundary Layers*. Journal of Fluid Mechanics, Vol. 161, pp. 371-403, 1985.

- [34] K. C. Muck, P. H. Hoffman and P. Bradshaw, *The Effect of Convex Surface Curvature on Turbulent Boundary Layers*. Journal of Fluid Mechanics, Vol. 161, pp. 347-369, 1985.
- [35] R. S. Barlow and J. P. Johnston, *Structure of a Turbulent Boundary Layer on a Concave Surface*. Journal of Fluid Mechanics, Vol. 191, pp. 137-176, 1988.
- [36] P. S. Klebanoff and K. D. Tidstrom, *Mechanism by Which a Two-Dimensional Roughness Element Induces Boundary Layer Transition*. Physics of Fluids, Vol. 15(7), pp. 1173-1188, 1972.
- [37] F. R. Marzabadi and M. R. Soltani, *Effect of Leading-Edge Roughness on Boundary Layer Transition of an Oscillating Airfoil*. Scientia Iranica Transactions B: Mechanical Engineering, Vol. 20(3), pp. 508-515, 2013.
- [38] M. Gaster, *Understanding the Effects of Surface Roughness on the Growth of Disturbances*. 46th AIAA Fluid Dynamics Conference, AIAA Aviation, (Paper no. AIAA 2016-4384), Washington D.C., USA, 2016.
- [39] W. Linke, *Über den Strömungswiderstand einer beheizten ebenen Platte*. Luftfahrtforschung, Vol. 19, pp. 157-160, 1942.
- [40] H. W. Liepmann and G. H. Fila, *Investigations of Effects of Surface Temperature and Single Roughness Elements on Boundary-Layer Transition*. NACA Technical Note No. 1196, Washington, 1947.
- [41] H. K. Versteeg and W. Malalasekera, *An Introduction to Computational Fluid Dynamics*. Second Edition, Pearson Education Limited, Harlow, UK 2007.
- [42] A. M. Savill, *Further Progress in the Turbulence Modelling of Bypass Transition*. Applied Scientific Research, Vol. 51, pp. 555-562, 1993.

- [43] A. M. Savill, *Further Progress in the Turbulence Modelling of Bypass Transition*. Engineering Turbulence Modelling and Experiments 2, Elsevier, Vol. 2, pp. 583-592, 1975.
- [44] A. M. Savill, *Some Recent Progress in the Turbulence Modelling of Bypass Transition*. Proceedings of the International Conference on Near-Wall Turbulent Flows, Elsevier, Tempe, AZ, USA, pp. 829-848, 1993.
- [45] F. R. Menter, R. B. Langtry, S. R. Likki and Y. B. Suzen, *A Correlation-based Transition Model Using Local Variables Part I: Model Formulations*. Journal of Turbomachinery, Vol. 128, pp. 413-422, 2006.
- [46] D. K. Walters and D. Cokljat, *A Three-Equation Eddy-Viscosity Model for Reynolds-Averaged Navier-Stokes Simulations of Transitional Flow*. Journal of Fluids Engineering, Vol. 130, 121401 2008.
- [47] D. K. Walters and J. H. Leylek, *A New Model for Boundary Layer Transition Using a Single-Point RANS Approach*. Journal of Turbomachinery, Vol. 126, pp. 193-202, 2004.
- [48] W. J. Devenport, M. C. Rife, S. Liapis and G. J. Follin, *The Structure and Development of a Wing-tip Vortex*. Journal of Fluid Mechanics, Vol. 312, pp.67-106 1996.
- [49] J. Russell, *Length and Bursting of Separation Bubbles: A Physical Interpretation*. NASA. Langley Res. Center; The Sci. and Technol. of Low Speed and Motorless Flight, Pt. 1 Cambridge, MA, USA, pp. 177-202 1979.
- [50] S. Burgmann, C. Brucker and W. Schroder, *Scanning PIV Measurements of a Laminar Separation Bubble*. Experiments in Fluids, Vol. 41, pp. 319-326, 1986.

- [51] C. Lyon, M. Selig and A. Broeren, *Boundary Layer Trips on Airfoils at Low Reynolds Numbers*. AIAA Paper 97-0511 1997.
- [52] H. Hu and Z. Yang, *An Experimental Study of the Laminar Flow Separation on a Low-Reynolds-Number Airfoil*. Journal of Fluids Engineering, Vol. 130(5), 051101 (11pp), 2008.
- [53] M. Drela and B. Giles, *Viscous-Inviscid Analysis of Transonic and Low Reynolds Number Airfoils*. AIAA Journal, Vol. 10(10), pp. 1347-1355, 1987.
- [54] M. S. Genc, I. Karasu and H. H. Ackel, *An Experimental Study on Aerodynamics of NACA2415 Aerofoil at Low Re Numbers*. Experimental Thermal and Fluid Science, Vol. 39, pp. 252-264, 2012.
- [55] P. Ghadimi, A. B. Rostami and F. Jafarkazemi, *Aerodynamic Analysis of the Boundary Layer Region of Symmetric Airfoils at Ground Proximity*. Aerospace Science and Technology, Vol. 17, pp. 7-20, 2012.
- [56] E. W. Warren and H. A. Hassan, *Alternative to the Method for Determining Onset of Transition*. AIAA Journal, Vol. 36(1), pp. 111-113, 1998.
- [57] M. S. Genc, *Numerical Simulation of Flow Over a Thin Aerofoil at a High Reynolds Number Using a Transition Model*. Proc. IMechE Part C: J. Mechanical Engineering Science, Vol. 224, pp. 2155-2164, 2015.
- [58] V. Chitta, T. Jamal and D. K. Walters, *Numerical Investigation of Low-Reynolds Number Airfoil Flows Using Transition-Sensitive and Fully Turbulent RANS Models*. in ASME 2014 4th Joint US-European Fluids Engineering Division Summer Meeting, Chicago, Illinois, USA, 2014.

- [59] T. P. Dhakal, T. Jamal and D. K. Walters, *Numerical Simulation of a PAK-B Airfoil Using Fully Turbulent and Transition-Sensitive RANS Models*. in ASME 2011 International Mechanical Engineering Congress and Exposition, Denver, Colorado, USA, 2011.
- [60] O. Eisele and G. Pechlivanoglou, *Single and Multi-element Airfoil Performance Simulation Study and Wind Tunnel Validation*. Wind Energy - Impact of Turbulence, Springer Vol. 2, pp. 17-22, 2014.
- [61] R. G. Dominy, *Aerodynamics of Grand Prix Cars*. Proc. IMechE Pt. D, J. Automobile Engineering, Vol. 206, pp. 267-274, 1992.
- [62] K. Knowles, D. T. Donogue and M. V. Finnis, *A Study of Wings in Ground Effect*. RAeS Vehicle Aerodynamics Conference, Loughborough University, 1994.
- [63] R. Ranzenbach and J. B. Barlow, *Two-Dimensional Airfoil in Ground Effect, An Experimental and Computational Study*. SAE Paper 94-2509, 1994.
- [64] R. Ranzenbach and J. B. Barlow, *Cambered Airfoil in Ground Effect - Wind Tunnel and Road Conditions*. AIAA paper no. 95-1909, 1995.
- [65] R. Ranzenbach and J. B. Barlow, *Cambered Airfoil in Ground Effect: An Experimental and Computational Study*. SAE Paper 96-0909, 1996.
- [66] R. Ranzenbach and J. B. Barlow, *Multi-Element Airfoil in Ground Effect - An Experimental and Computational Study*. AIAA paper no. 97-2238, 1997.
- [67] D. W. Marshall, S. F. Newman and C. B. Williams, *Boundary Layer Effects on a Wing in Ground-Effect*. Aircraft Engineering and Aerospace Technology: An International Journal, Vol. 82(2), pp. 99-106, 2010.

- [68] W. J. Jasinski and M. S. Selig, *Experimental Study of Open-Wheel Race Car Front Wings*. SAE Paper 98-3042, 1998.
- [69] J. Zerihan and X. Zhang *Aerodynamics of a Single-Element Wing in Ground Effect*. Journal of Aircraft, Vol. 37(6), pp. 1058-1064, 2000.
- [70] X. Zhang and J. D. C. Zerihan, *Turbulent Wake Behind a Single Element Wing in Ground Effect*. 10th International Symposium: Applications of Laser Techniques to Fluid Mechanics, Lisbon, 2000.
- [71] X. Zhang, J. Zerihan, A. Ruhrmann and M. Deviese, *Tip Vortices Generated by a Wing in Ground Effect*. 11th International Symposium: Applications of Laser Techniques to Fluid Mechanics, Lisbon, 2002.
- [72] X. Zhang and J. Zerihan, *Aerodynamics of a Double Element Wing in Ground Effect*. AIAA Journal, Vol. 45(6), pp. 1007-1016, 2003.
- [73] X. Zhang and J. Zerihan, *Edge Vortices of a Double-Element Wing in Ground Effect*. Journal of Aircraft, Vol. 41(5), pp. 1127-1137, 2004.
- [74] A. M. O. Smith, *High-Lift Aerodynamics*. Journal of Aircraft, Vol. 12(6), pp. 501-530, 1975.
- [75] R. P. Moseley, *The Application of Particle Image Velocimetry to Wing Vortex Flows*. PhD Thesis, Imperial College London, 1999.
- [76] B. Garrood, *Aerodynamic Interaction Mechanisms Relevant to the Under-Bodies of Formula 1 Cars*. PhD Thesis, Imperial College London, 2004.
- [77] J. M. Pegrum *Experimental Study of the Vortex System Generated by a Formula 1 Front Wing*. PhD Thesis, Imperial College London, 2006.

- [78] V. Galoul and T. J. Barber, *A Study of an Inverted Wing with Endplates in Ground Effect*. 16th Australasian Fluid Mechanics Conference, Gold Coast, Australia, 2007.
- [79] J. W. Vogt and T. J. Barber, *Variation of Ground Effect Phenomena About Downforce Generating Tyrrell and NACA4412 Aerofoils*. International Journal of Aerodynamics, Vol. 1(1), pp. 82-86, 2010.
- [80] J. W. Vogt and T. J. Barber, *Variation of Ground Effect Phenomena About Downforce Generating Aerofoils Caused by Changes in Aerofoil Shape*. International Journal of Aerodynamics, Vol. 2(1), pp. 36-49, 2012.
- [81] J. Correia, L. S. Roberts, M. V. Finnis and K. Knowles, *Scale Effects on a Single-Element Inverted Wing in Ground Effect*. The Aeronautical Journal, Vol. 118(1205), pp. 797-809, 2014.
- [82] J. Correia, L. S. Roberts, M. V. Finnis and K. Knowles, *Aerodynamic Characteristics of a Monoposto Racing Car Front Wing Operating in a Turbulent Wake*. International Vehicle Aerodynamics Conference, Loughborough, UK, 2014.
- [83] J. Keogh, G. Doig and S. Diasinos, *Flow Compressibility Effects Around an Open-Wheel Racing Car*. The Aeronautical Journal, Vol. 118(1210) pp. 1409-1430, 2014.
- [84] J. Keogh, G. Doig, S. Diasinos and T. Barber, *The Influence of Cornering on the Vortical Wake Structures of an Inverted Wing*. Proc. IMechE Part D: Journal of Automobile Engineering, Vol. 229(13), pp. 1817-1829, 2015.
- [85] L. S. Roberts, J. Correia, M. V. Finnis and K. Knowles, *Aerodynamic Characteristics of a Wing & Flap Configuration in Ground Effect & Yaw*. Proc. IMechE Part D: Journal of Automobile Engineering, Vol. 230(6), pp. 841-854, 2015.

- [86] J. Molina and X. Zhang, *Aerodynamics of a Heaving Airfoil in Ground Effect*. AIAA Journal, Vol. 49(6), pp. 1168-1179, 2011.
- [87] J. Katz, *Calculation of the Aerodynamic Forces on Automotive Lifting Surfaces*. Journal of Fluids Engineering, Vol. 107, pp. 438-443, 1985.
- [88] J. Zerihan and X. Zhang, *A Single Element Wing in Ground Effect; Comparisons of Experiments and Computation*. 39th AIAA Aerospace Sciences Meeting and Exhibit (Paper no. 2001-0423), Reno, NV, 2001.
- [89] J. Zerihan, *An Investigation into the Aerodynamics of Wings in Ground Effect*. PhD Thesis, University of Southampton, 2001.
- [90] N. Lawson, K. Knowles, R. J. E. Hart, J. N. Wray and J. M. Eyles, *An Experimental Investigation Using PIV of the Underflow of a GA(W)-1 Aerofoil Section in Ground Effect*. 4th MIRA International Vehicle Aerodynamics Conference, MIRA, Warwick, UK, 2002.
- [91] S. Mahon and X. Zhang, *Computational Analysis of Pressure and Wake Characteristics of an Aerofoil in Ground Effect*. Journal of Fluids Engineering, Vol. 127, pp. 290-298, 2005.
- [92] S. Mahon and X. Zhang, *Computational Analysis of a Inverted Double-Element Airfoil in Ground Effect*. Journal of Fluids Engineering, Vol. 128, pp. 1172-1180, 2006.
- [93] G. Doig, T. J. Barber and A. J. Neely, *The Influence of Compressibility on the Aerodynamics of an Inverted Wing in Ground Effect*. Journal of Fluids Engineering, Vol. 133, pp. 1-12, 2011.

- [94] J. Heyder-Bruckner and X. Zhang, *Detached-Eddy Simulation of a Double-Element Wing in Ground Effect*. 48th AIAA Aerospace Meeting (Paper no. 2010-59), Orlando, FL, 2010.
- [95] M. A. van den Berg and X. Zhang, *The Aerodynamic Interaction Between an Inverted Wing and a Rotating Wheel*. Journal of Fluids Engineering, Vol. 131, 101104, 2009.
- [96] M. Soso and P. Wilson, *Investigating Changes to the Downforce Curve of a Double Element Airfoil in Ground Effect*. SAE Technical Paper 2004-01-3558, 2004.
- [97] Y. Moryossef and Y. Levey, *Effect of Oscillations on Airfoils in Close Proximity to the Ground*. AIAA Journal, Vol. 42(9), pp. 1755-1764, 2004.
- [98] W. A. Mokhtar, *Aerodynamics of High-Lift Wings with Ground Effect for Racecars*. SAE Technical Paper 2008-01-0656, 2008.
- [99] Y. Kuya, K. Takeda, X. Zhang, S. Beeton and T. Pandaleon, *Flow Separation Control on a Race Car Wing with Vortex Generators in Ground Effect*. Journal of Fluids Engineering, Vol. 131, 121103, 2009.
- [100] J. Molina, X. Zhang and D. Angland, *On the Unsteady Motion and Stability of a Heaving Airfoil in Ground Effect*. Acta Mechanica Sinica, Vol. 27(2), pp. 164-178, 2011.
- [101] F. M. White, *Viscous Fluid Flow*. McGraw-Hill, New York, NY, 1974.
- [102] R. Knowles, *Monoposto Racecar Wheel Aerodynamics: Investigation of Near-Wake Structure and Support-Sting Interference*. PhD thesis, Cranfield University, 2005.

- [103] K. Knowles and M. V. Finnis, *Development of a New Open-Jet Wind Tunnel and Rolling Road Facility*. In 2nd MIRA International Conference on Vehicle Aerodynamics, Nuneaton, UK 1998.
- [104] E. C. Maskell, *A Theory of the Blockage Effect on Bluff Bodies and Stalled Wings in a Closed Wind Tunnel*. ARC R&M 340, 1963.
- [105] P. Kohut, M. Sulitka and Z. Randa, *Determination of Blockage Correction in Open-Jet Wind Tunnel*. Proc. of 16th International Symposium on Transport Phenomena, Prague, 2005.
- [106] E. Mercker and K. Cooper, *A Two-Measurement Correction for the Effects of a Pressure Gradient on Automotive, Open-Jet, Wind Tunnel Measurements*. SAE Technical Paper 2006-01-0568, 2006.
- [107] T. Lounsberry and J. Walter, *Practical Implementation of the Two-Measurement Correction Method in Automotive Wind Tunnels*. SAE Int. J. Passeng. Cars - Mech. Syst., Vol. 8(2), pp. 676-686, 2015.
- [108] J. Correia. *The Influence of External Disturbances on the Aerodynamic Performance of a Wing in Ground Effect*. PhD Thesis, Cranfield University, 2016.
- [109] A. E. von Doenhoff and E. A. Horton *A Low-speed Investigation of the Effect of a Sandpaper Type of Roughness on Boundary Layer Transition*. NACA Report 1349, Langley Aeronautical Laboratory, 1958.
- [110] M. Drela, *Design and Optimisation Method for Multi-Element Airfoils*. AIAA Paper 93-0960, Aerospace Design Conference, Irvine, CA, 1993.

- [111] J. L. van Ingen *A Suggested Semi-empirical Method for the Calculation of Boundary Layer Transition Region*. Report VTH-74, Delft University of Technology, Netherlands, 1956.
- [112] A. M. O. Smith and N. Gamberoni, *Transition, Pressure Gradient and Stability Theory*. Technical Report ES-26388, Douglas Aircraft Company, 1956.
- [113] D-A Kim, J-H Yang, J-W Chang and J Chung, *Boundary Layer and Near-wake Measurements of NACA 0012 Airfoil at Low Reynolds Numbers*. AIAA Paper 2009-1472, 47th AIAA Aerospace Sciences Meeting Including The New Horizons Forum and Aerospace Exposition, Orlando, FL, 2009.
- [114] A. L. Braslow and E. C. Knox, *Simplified Method for Determination of Critical Height of Distributed Roughness Particles for Boundary-Layer Transition at Mach Numbers from 0 to 5*. NACA Technical Note 4363, Langley Aeronautical Laboratory, VA, 1958.
- [115] M. Lanfrit, *Best Practice Guidelines for Handling Automotive External Aerodynamics with Fluent*. Version 1.2, guide. Darmstadt: Fluent Deutschland GmbH, 2005.
- [116] E. N. Jacobs and A. Sherman, *Airfoil Section Characteristics as Affected by Variations of the Reynolds Number*. NACA Report No. 586, 1937.
- [117] C. Turner, *Laminar Kinetic Energy Modelling for Improved Laminar-Turbulent Transition Prediction*. PhD Thesis, University of Manchester, 2012.
- [118] Photograph taken from <http://www.adac-motorsport.de>
- [119] Photograph taken from <http://www.thecheckeredflag.co.uk>

- [120] A. V. Dovgal, V. V. Kozlov and A. Michalke, *Laminar Boundary Layer Separation: Instability and Associated Phenomena*. Prog. Aerospace Sci., Vol. 30, pp. 61-94, 1994. .
- [121] A. J. Saddington, M. V. Finnis and K. Knowles, *The Characterisation of a Gust Generator for Aerodynamic Testing*. Proc. IMechE Part G: Journal of Aerospace Engineering, Vol. 229(7), pp. 1214-1225, 2014.

