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**ON THE DESIGN AND AERODYNAMIC PERFORMANCE
OF JOINED-WING AIRCRAFTS**

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**SCHOOL OF MECHANICAL AND AEROSPACE ENGINEERING
NANYANG TECHNOLOGICAL UNIVERSITY**

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SUBMITTED

BY

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SCHOOL OF MECHANICAL AND AEROSPACE ENGINEERING

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Abstract

The increase in air traffic in the last decade has given further incentive to explore unconventional configurations which may have added aerodynamic efficiency. The configuration of choice in the study is the joined-wing configuration. An experimental and numerical study was carried out to investigate the aerodynamic characteristics, as well as the flow behaviour, of two joined-wing configurations, one with negative stagger and one with positive stagger, as well as a typical commercial aircraft configuration for reference, at various orientations by permuting a combination of angles of attack, sideslip angle as well as roll angle.

Force balance measurements and surface oil flow visualization were carried out in closed-loop subsonic wind tunnel to examine the lift/drag behaviour as well as the longitudinal, directional and lateral stability. Through the wind tunnel experiments, it was found that both joined-wing configurations do not show sharp dips in the lift coefficient throughout the range of angles of attack, unlike the reference aircraft at approximately 15° . However, in general, it was found that the lift coefficients of the joined-wing aircraft were lower than that of the reference across the range of angles tested, and do not show any conclusive evidence of increased aerodynamic efficiency over the reference configuration at the tested Reynolds numbers. In addition, there were some minor problems in terms of the lateral stability for the positively staggered joined-wing configuration.

Numerical simulations were carried out at similar conditions with a commercially available code for corroboration and to provide additional data not collected in the wind

tunnel experiments for further analysis. The trends in the numerical simulations matched the wind tunnel experiments quite well, especially for the positively staggered joined-wing configuration. The streamlines generated from the numerical simulation results also greatly aided in the analysis and interpretation of the surface oil flow visualization results. However, the numerical simulations were also unable to accurately model the flow physics of the laminar separation bubble at the lower AOAs. In addition, the discrepancies with the experimental results were higher when combinations of roll and yaw were implemented.

In the course of designing the wind tunnel test model, difficulties were faced when attempting to fabricate with conventional machining methods. Hence, modern additive manufacturing techniques were used. However, due to the method of fabricating the wings and the material used, wing deflections were also observed. These deflections were photographed and digitized, then reproduced as a smooth curve through regression. It was found that the deflection magnitudes are tightly associated with the lift curve behaviour.

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Table of Contents

Abstract	i
Acknowledgments	iii
Table of Contents	iv
List of Figures	vi
List of Tables	xii
List of Symbols	xiii
Chapter 1 Introduction	1
1.1 Background.....	1
1.2 Literature Review	2
1.2.1 Overview of Joined Wings.....	2
1.2.2 Parameters of Joined Wings.....	5
1.2.3 Works done so far	11
1.3 Research Aims and Scope	13
1.4 Organization of Thesis	14
Chapter 2 Experimental Setup and Procedures	15
2.1 Overview	15
2.2 Experimental Apparatus	15
2.2.1 Close-loop Wind Tunnel.....	15
2.2.2 Sting Balance	16
2.2.3 Wind Tunnel Model.....	18
2.2.3.1 Wind Tunnel Model Configurations	18
2.2.3.2 Wind Tunnel Model Sizing	23
2.2.3.3 Wind Tunnel Model Material Selection.....	24
2.3 Capturing Deflections.....	26
2.4 Surface Oil Flow Visualizations.....	27
Chapter 3 Numerical Setup and Procedures.....	29
3.1 Overview	29
3.2 Choice of Turbulence Models	30
3.2.1 High Speed Cases.....	31
3.2.1.1 2D Airfoil	31
3.2.1.2 3D Wing	32
3.2.1.3 3D Wing Body.....	33
3.2.2 Low Speed Case.....	37
3.2.2.1 2D Airfoil	37
3.3 Numerical Setup and Grid Independence	37
Chapter 4 Aerodynamic Characterization of PP250	40
4.1 Overview	40
4.2 Variations in AOA.....	40
4.2.1 Variations in AOA: Wind Tunnel Results	40
4.2.2 Variations in AOA: Numerical Simulation Results.....	42
4.2.3 Variations in AOA: Surface Oil Flow Visualization Results	46
4.3 Variations in AOA and Sideslip Angle	52
4.3.1 Variations in AOA and Sideslip Angle: Wind Tunnel Results.....	53
4.3.2 Variations in AOA and Sideslip Angle: Numerical Simulation Results ..	55
4.3.3 Variations in AOA and Sideslip Angle: Surface Oil Flow Visualization Results	60

4.4	Variations in AOA and Roll Angle	67
4.4.1	Variations in AOA and Roll angle: Wind Tunnel Results.....	67
4.4.2	Variations in AOA and Roll Angle: Numerical Simulation Results	67
4.4.3	Variations in AOA and Roll Angle: Surface Oil Flow Visualization Results	71
Chapter 5	Aerodynamic Characterization of D150.....	76
5.1	Overview	76
5.2	Variations in AOA.....	76
5.2.1	Variations in AOA: Wind Tunnel Results	76
5.2.2	Variations in AOA: Numerical Simulation Results.....	78
5.2.3	Variations in AOA: Surface Oil Flow Visualization Results	82
5.3	Variations in AOA and Sideslip Angle	85
5.3.1	Variations in AOA and Sideslip Angle: Wind Tunnel Results.....	86
5.3.2	Variations in AOA and Sideslip Angle: Numerical Simulation Results ..	87
5.3.3	Variations in AOA and Sideslip Angle: Surface Oil Flow Visualization Results	92
5.4	Variations in AOA and Roll Angle	97
5.4.1	Variations in AOA and Roll Angle: Wind Tunnel Results.....	97
5.4.2	Variations in AOA and Roll Angle: Numerical Simulation Results	98
5.4.3	Variations in AOA and Roll Angle: Surface Oil Flow Visualization Results	102
Chapter 6	Aerodynamic Characterization of B150.....	106
6.1	Overview	106
6.2	Variations in AOA.....	106
6.2.1	Variations in AOA: Wind Tunnel Results	106
6.2.2	Variations in AOA: Numerical Simulation Results.....	109
6.2.3	Variations in AOA: Surface Oil Flow Visualization Results	114
6.3	Variations in AOA and Sideslip Angle	117
6.3.1	Variations in AOA and Sideslip Angle: Wind Tunnel Results.....	118
6.3.2	Variations in AOA and Sideslip Angle: Numerical Simulation Results	118
6.3.3	Variations in AOA and Sideslip Angle: Surface Oil Flow Visualization Results	123
6.4	Variations in AOA and Roll Angle	128
6.4.1	Variations in AOA and Roll Angle: Wind Tunnel Results.....	128
6.4.2	Variations in AOA and roll angle: Numerical Simulation Results	128
6.4.3	Variations in AOA and Roll Angle: Surface Oil Flow Visualization Results	132
Chapter 7	Characterization of Wing Deflections	135
7.1	Overview	135
7.2	Wing Deflections of PP250	135
7.3	Wing Deflections of D150.....	140
7.4	Wing Deflections of B150.....	142
Chapter 8	Conclusions	146
8.1	Summary.....	146
8.2	Recommendations for Future Work	148
References	.....	150
List of Publications	.....	166
1.	Journal Articles.....	166
2.	Conference Articles	166

List of Figures

Fig. 1-1 (a) General joined-wing configuration and (b) control surface deflections for direct lift and lateral force (Wolkovitch, 1986).	3
Fig. 1-2 (a) Span efficiencies, e , of different wing configurations and (b) reduction of induced drag against gap for different configurations (Kroo, 2001).	4
Fig. 1-3 (a) Stagger (Knight and Noyes, 1929a), (b) gap (Bagwill and Selburg, 1996), (c) decalage, and (d) overhang (Knight and Noyes, 1929b).	7
Fig. 1-4 (a) Diagram of Prandtl's Best Wing System (Prandtl, 1929) and (b) lift distribution for minimum induced drag (Frediani, 2005).	8
Fig. 1-5 (a) Twist angle viewed from the top view and (b) cant angle (Bagwill and Selburg, 1997).	10
Fig. 2-1 Coordinate system used in this thesis.	17
Fig. 2-2 Shift in moments.	17
Fig. 2-3 (a) Data organization in CPACS (Rizzi et al., 2012) and (b) TiGL view of joined-wing design	19
Fig. 2-4 CAD representations of (a) PP250, (b) D150 and (c) B150.	19
Fig. 2-5 Planform view of PP250.	20
Fig. 2-6 Planform view of D150.	21
Fig. 2-7 Planform view of B150.	22
Fig. 2-8 DSLR camera and its position circled in red for capturing (a) deflections in the z-axis from the side and (b) span-wise curvatures.	26
Fig. 3-1 Comparison of C_p distribution for RAE 2822.	32
Fig. 3-2 Location of comparison for C_p for ONERA M6.	33
Fig. 3-3 Comparison of C_p distribution for M6 wing at (a) section 1, (b) section 2, (c) section 3, (d) section 4, (e) section 5, (f) section 6 and (g) section 7.	34
Fig. 3-4 Location of comparison for C_p for DLR-F6 (Godard, 2003).	35
Fig. 3-5 Comparison of C_p distribution for F6 at (a) section 1, (b) section 2, (c) section 3, (d) section 4, (e) section 5, (f) section 6, (g) section 7 and (h) section 8.	36
Fig. 3-6 Comparison of (a) C_l and (b) C_d of SD7003 airfoil.	38

Fig. 3-7 (a) Domain boundary size in terms of fuselage length and (b) surface mesh of the different configurations.	39
Fig. 4-1 Plot of (a) C_L and (b) C_D for PP250 for varying AOA.	41
Fig. 4-2 Plot of pitching moment coefficient, C_M , against effective angle of attack, α_{eff} for PP250.	42
Fig. 4-3 Comparison of (a) coefficient of lift, C_L , and (b) coefficient of drag, C_D , against AOA between experimental and numerical simulation results.	43
Fig. 4-4 C_P plots at (a) 1/4 semi-span of front wing, (b) 1/4 semi-span of rear wing, (c) 1/2 semi-span of front wing, (d) 1/2 semi-span of rear wing, (e) 3/4 semi-span of front wing, (f) 3/4 semi-span of rear wing.	45
Fig. 4-5 Plot of streamlines and C_P at junction between the front wing and fuselage at (a) 4° , (b) 8° , (c) 10° and (d) 12° AOA.	46
Fig. 4-6 Plot of streamlines and C_P at along the rear wing at (a) 4° , (b) 8° , (c) 10° and (d) 12° AOA.	46
Fig. 4-7 Surface oil flow visualizations for (a) 0° AOA, (b) 4° AOA, (c) 6° AOA, (d) 8° AOA, (e) 10° AOA, (f) 18° AOA, (g) 20° AOA and (h) 24° AOA.	48
Fig. 4-8 Flow pattern observed on wing of sailplane (Boermans et al., 1998).	49
Fig. 4-9 (a) Mesh of airfoil, (b) wall shear stress across the upper surface of the airfoil, (c) streamlines across the airfoil and (d) plot of intermittency, γ	50
Fig. 4-10 Surface streamlines based on wall shear for (a) 0° AOA, (b) 4° AOA, (c) 6° AOA, (d) 8° AOA, (e) 10° AOA, (f) 18° AOA, (g) 20° AOA and (h) 24° AOA.	52
Fig. 4-11 Plot of (a) C_L and (b) C_D for PP250 for varying β	53
Fig. 4-12 Plot of (a) yawing moment coefficient, C_N and (b) rolling moment coefficient, C_l , against sideslip angle, β	54
Fig. 4-13 Comparison of coefficients between wind tunnel (WT) and CFD results for (a) $\beta=15^\circ$ and (b) $\beta=30^\circ$	55
Fig. 4-14 Flow fields generated for $\beta=15^\circ$, (a) AOA 0° , (b) AOA 10° and (c) AOA 20°	57
Fig. 4-15 Flow fields generated for $\beta=30^\circ$, (a) AOA 0° , (b) AOA 10° and (c) AOA 20°	59
Fig. 4-16 Comparison of coefficients between surface oil flow visualizations and CFD results for $\beta=15^\circ$, (a) 0° AOA, (b) 5° AOA, (c) 10° AOA, (d) 15° AOA, (e) 20° AOA.	64

Fig. 4-17 Comparison of coefficients between surface oil flow visualizations and CFD results for $\beta=30^\circ$, (a) 0° AOA, (b) 5° AOA, (c) 10° AOA, (d) 15° AOA, (e) 20° AOA.	65
Fig. 4-18 Possible flow structures and streamlines for explaining surface oil flow patterns.	66
Fig. 4-19 Plot of (a) C_L and (b) C_D for PP250 for varying ϕ	68
Fig. 4-20 Comparison of coefficients between wind tunnel (WT) and CFD results for (a) $\phi=-15^\circ$ and (b) $\phi=-30^\circ$	69
Fig. 4-21 Flow fields generated for $\phi=-15^\circ$, (a) AOA 0° , (b) AOA 15° and (c) AOA 25°	70
Fig. 4-22 Flow fields generated for $\phi=-30^\circ$, (a) AOA 0° , (b) AOA 15° and (c) AOA 25°	71
Fig. 4-23 Comparison of coefficients between surface oil flow visualizations and CFD results for $\phi=-15^\circ$, (a) 0° AOA, (b) 5° AOA, (c) 10° AOA, (d) 15° AOA, (e) 20° AOA.	74
Fig. 4-24 Comparison of coefficients between surface oil flow visualizations and CFD results for $\phi=-30^\circ$, (a) 0° AOA, (b) 5° AOA, (c) 10° AOA, (d) 15° AOA, (e) 20° AOA.	75
Fig. 5-1 Plot of (a) C_L and (b) C_D for D150 for varying AOA.	77
Fig. 5-2 Plot of pitching moment coefficient, C_M , against effective angle of attack, α_{eff} for D150.	78
Fig. 5-3 Comparison of (a) coefficient of lift, C_L , and (b) coefficient of drag, C_D , against AOA between experimental and numerical simulation results.	79
Fig. 5-4 C_p plots at (a) 1/4 semi-span of wing, (b) 1/2 semi-span of wing, and (c) 3/4 semi-span of wing.	81
Fig. 5-5 Plot of streamlines and C_p across the wing and fuselage at (a) 4° AOA, (b) 8° AOA, (c) 10° AOA, (d) 16° AOA and (e) 20° AOA.	82
Fig. 5-6 Surface oil flow visualizations for (a) 0° AOA, (b) 4° AOA, (c) 8° AOA, (d) 10° AOA, (e) 16° AOA and (f) 20° AOA.	83
Fig. 5-7 Surface streamlines based on wall shear for (a) 0° AOA, (b) 4° AOA, (c) 8° AOA, (d) 10° AOA, (e) 16° AOA and (f) 20° AOA.	85
Fig. 5-8 Plot of (a) C_L and (b) C_D for D150 for varying β	87
Fig. 5-9 Plot of (a) yawing moment coefficient, C_N and (b) rolling moment coefficient, C_l , against sideslip angle, β	87

Fig. 5-10 Comparison of coefficients between WT and CFD results for (a) $\beta=15^\circ$ and (b) $\beta=30^\circ$	88
Fig. 5-11 Flow fields generated for $\beta=15^\circ$, (a) AOA 0° , (b) AOA 10° , (c) AOA 15° and (d) AOA 20°	89
Fig. 5-12 Flow fields generated for $\beta=30^\circ$, (a) AOA 0° , (b) AOA 10° , (c) AOA 15° and (d) AOA 20°	91
Fig. 5-13 Comparison of coefficients between surface oil flow visualizations and CFD results for $\beta=15^\circ$, (a) 0° AOA, (b) 5° AOA, (c) 10° AOA, (d) 15° AOA, (e) 20° AOA.	95
Fig. 5-14 Comparison of coefficients between surface oil flow visualizations and CFD results for $\beta=30^\circ$, (a) 0° AOA, (b) 5° AOA, (c) 10° AOA, (d) 15° AOA, (e) 20° AOA.	96
Fig. 5-15 Streamlines for roll up vortex for $\beta=30^\circ$	97
Fig. 5-16 Plot of (a) C_L and (b) C_D for D150 for varying ϕ	98
Fig. 5-17 Comparison of coefficients between WT and CFD results for (a) $\phi=-15^\circ$ and (b) $\phi=-30^\circ$	99
Fig. 5-18 Flow fields generated for $\phi=-15^\circ$, (a) AOA 0° , (b) AOA 10° , (c) AOA 15° and (d) AOA 20°	100
Fig. 5-19 Flow fields generated for $\phi=-30^\circ$, (a) AOA 0° , (b) AOA 10° , (c) AOA 15° and (d) AOA 20°	101
Fig. 5-20 Streamlines of vortex forming on left wing at $\phi=-30^\circ$, AOA 15°	102
Fig. 5-21 Comparison of coefficients between surface oil flow visualizations and CFD results for $\phi=-15^\circ$, (a) 0° AOA, (b) 5° AOA, (c) 10° AOA, (d) 15° AOA, (e) 20° AOA.	104
Fig. 5-22 Comparison of coefficients between surface oil flow visualizations and CFD results for $\phi=-30^\circ$, (a) 0° AOA, (b) 5° AOA, (c) 10° AOA, (d) 15° AOA, (e) 20° AOA.	105
Fig. 6-1 Comparison of (a) C_L , (b) C_D and (c) L/D for B150 and D150 for varying AOA.....	107
Fig. 6-2 Plot of pitching moment coefficient, C_M , against effective angle of attack, α_{eff} for B150 and D150.....	108
Fig. 6-3 Comparison of (a) coefficient of lift, C_L , and (b) coefficient of drag, C_D , against AOA between experimental and numerical simulation results.....	109
Fig. 6-4 C_p plots at (a) 1/4 semi-span of front wing, (b) 1/4 semi-span of rear wing, (c) 1/2 semi-span of front wing, (d) 1/2 semi-span of rear wing, (e) 3/4 semi-span of front wing, (f) 3/4 semi-span of rear wing.....	111

Fig. 6-5 Plot of streamlines and C_p across the front wing (left) and rear wing (right) at (a) 4° AOA, (b) 8° AOA, (c) 10° AOA, (d) 16° AOA and (e) 20° AOA.....	112
Fig. 6-6 Surface oil flow visualizations for (a) 0° AOA, (b) 4° AOA, (c) 8° AOA, (d) 10° AOA, (e) 16° AOA and (f) 20° AOA.....	116
Fig. 6-7 Surface streamlines based on wall shear for (a) 0° AOA, (b) 4° AOA, (c) 8° AOA, (d) 10° AOA, (e) 16° AOA and (f) 20° AOA.	117
Fig. 6-8 Plot of (a) C_L and (b) C_D for D150 for varying β	119
Fig. 6-9 Plot of (a) yawing moment coefficient, C_N and (b) rolling moment coefficient, C_l , against sideslip angle, β	119
Fig. 6-10 Comparison of coefficients between WT and CFD results for (a) $\beta=15^\circ$ and (b) $\beta=30^\circ$	120
Fig. 6-11 Flow fields generated for $\beta=15^\circ$, (a) AOA 0° , (b) AOA 5° and (c) AOA 15°	122
Fig. 6-12 Flow fields generated for $\beta=30^\circ$, (a) AOA 0° , (b) AOA 5° and (c) AOA 15°	123
Fig. 6-13 Comparison of coefficients between surface oil flow visualizations and CFD results for $\beta=15^\circ$, (a) 0° AOA, (b) 5° AOA, (c) 10° AOA, (d) 15° AOA, (e) 20° AOA.	126
Fig. 6-14 Comparison of coefficients between surface oil flow visualizations and CFD results for $\beta=30^\circ$, (a) 0° AOA, (b) 5° AOA, (c) 10° AOA, (d) 15° AOA, (e) 20° AOA.	127
Fig. 6-15 Plot of (a) C_L and (b) C_D for D150 for varying ϕ	129
Fig. 6-16 Comparison of coefficients between WT and CFD results for (a) $\phi=-15^\circ$ and (b) $\phi=-30^\circ$	129
Fig. 6-17 Flow fields generated for $\phi=-15^\circ$, (a) AOA 0° , (b) AOA 5° and (c) AOA 15°	130
Fig. 6-18 Flow fields generated for $\phi=-30^\circ$, (a) AOA 0° , (b) AOA 5° and (c) AOA 15°	131
Fig. 6-19 Streamlines of vortices forming for AOA 15° , (a) $\phi=-15^\circ$ and (b) $\phi=-30^\circ$ ...	131
Fig. 6-20 Comparison of coefficients between surface oil flow visualizations and CFD results for $\phi=-15^\circ$, (a) 0° AOA, (b) 5° AOA, (c) 10° AOA, (d) 15° AOA, (e) 20° AOA.	133
Fig. 6-21 Comparison of coefficients between surface oil flow visualizations and CFD results for $\phi=-30^\circ$, (a) 0° AOA, (b) 5° AOA, (c) 10° AOA, (d) 15° AOA, (e) 20° AOA.	134

Fig. 7-1 Superimposed images of the deflections captured for AOA 8° (a) from the rear, (b) from the side and AOA 10° (c) from the rear, (d) from the side.....	136
Fig. 7-2 Comparison between (a) deflected wing geometry used in numerical simulations and (b) deflections captured in wind tunnel experiments.....	137
Fig. 7-3 Failure to identify the wing due to poor colour contrast.	137
Fig. 7-4 Comparison of C_L obtained from wind tunnel experiments and numerical simulations with and without the deflections taken into account.	138
Fig. 7-5 Comparison of C_P from numerical simulations along the upper surface for the (a) front wing and (b) rear wing at AOA= 8° , and the (c) front wing and (d) rear wing at AOA= 10°	139
Fig. 7-6 Determination of wing curvature for the D150 configuration.	141
Fig. 7-7 Determination of wing deflection and twist for the D150 configuration.	141
Fig. 7-8 Wing curvatures determined for D150.....	142
Fig. 7-9 Comparison of wingtip deflection with C_L for D150 configuration.	143
Fig. 7-10 Change in wing twist for D150 configuration.....	143
Fig. 7-11 Determination of wing curvature for the B150 configuration.....	144
Fig. 7-12 Determination of wing deflection and twist for the B150 configuration.	144
Fig. 7-13 Wing curvatures determined for (a) front wing and (b) rear wing of B150..	145
Fig. 7-14 Comparison of wingtip deflection with C_L for B150 configuration.....	145

List of Tables

Table 2-1 Limit and accuracy of sting balance	17
Table 2-2 Correction for moments.....	17
Table 2-3 Wing parameters for the different configurations.	23
Table 3-1 Comparison of wind tunnel results with CFD for RAE 2822.	32
Table 3-2 Comparison of wind tunnel results with CFD for DLR F6.	35
Table 3-3 Comparison of grid discretization error.....	39
Table 7-1 Maximum wing tip deflection magnitudes.	137

List of Symbols

b	Semi-span [m]
C_d	Sectional coefficient of drag
C_D	Coefficient of drag
C_{Di}	Induced drag coefficient
$C_{Di,e}$	Induced drag coefficient for elliptical lift distribution
C_l	Sectional coefficient of lift
C_L	Coefficient of lift
C_l	Coefficient of rolling moment
$C_{l\beta}$	$dC_l/d\beta$ [$1/^\circ$]
C_M	Coefficient of pitching moment
$C_{M\alpha}$	$dC_M/d\alpha$ [$1/^\circ$]
C_N	Coefficient of yawing moment
$C_{N\beta}$	$dC_N/d\beta$ [$1/^\circ$]
C_P	Coefficient of pressure
C_Y	Coefficient of lateral force
d	Distance [m]
e	span efficiency factor

F_x, F_y, F_z	Forces measured in the Cartesian coordinates [N]
G	Gap [m]
k	Turbulence kinetic energy [m^2/s^2]
l	Length [m]
M_x, M_y, M_z	Moments measured in the Cartesian coordinates [Nm]
u	Velocity [m/s]
u'	Velocity fluctuations [m/s]
u_τ	$(\tau_w/\rho)^{1/2}$, Friction velocity [m/s]
V	$G/2b$, Vertical aspect ratio (gap to span)
W	Weight loading on beam [N]
x, y, z	Cartesian coordinates
x', y', z'	Transformed Cartesian coordinates
y^+	Normal distance to the wall in boundary-layer wall units
Z	Sectional modulus [m^3]
α	Angle of attack, (AOA) [$^\circ$]
α_{eff}	$\alpha - \alpha_{0L}$ [$^\circ$]
α_{0L}	Zero lift angle of attack [$^\circ$]
β	Sideslip angle [$^\circ$]
δ_{ij}	Kronecker delta

μ	Dynamic viscosity [Ns/m ²]
μ_t	Turbulent (eddy) viscosity [Ns/m ²]
ρ	Density [kg/m ³]
σ	Stress [N/m ²]
τ_w	Surface shear stress [N/m ²]
ν	Kinematic viscosity [m ² /s]
ϕ	Roll angle [°]
ω	Specific dissipation rate [1/s]
ACARE	Advisory Council for Aeronautics Research in Europe
AIAA	American Institute of Aeronautics and Astronautics
CFD	Computational Fluid Dynamics
C.G.	Centre of Gravity
CPACS	Common Parametric Aircraft Configuration Schema
DLR	Deutsches Zentrum für Luft-und Raumfahrt e. V.
DPW	Drag Prediction Workshop
FSI	Fluid-Structure Interaction
IATA	International Air Transport Association
ICAO	International Civil Aviation Organization
KTH	Kungliga Tekniska Högskolan

MAC	Mean Aerodynamic Chord
MDO	Multi-Disciplinary Optimization
ONERA	Office National d'Etudes et de Recherches Aérospatiales
RAE	Royal Aircraft Establishment
RANS	Reynolds-Averaged Navier-Stokes
SA	Spalart-Allmaras
SST	Shear-Stress Transport
STL	Stereo Lithography
TiGL	TiGL Geometry Library
VLM	Vortex Lattice Method
WT	Wind Tunnel
XML	Extensible Markup Language

Chapter 1 Introduction

1.1 Background

Air travel has significantly increased over the past, especially so in the last two decades and is still projected to grow in the near future. Based on the statistical figures published by the International Civil Aviation Organization (ICAO), ICAO (2014) reports an average annual growth of 5.2% and 3.7% in terms of passenger count and freight tonnage from the years 2005 to 2014. Notwithstanding any internationally major situations such as terrorism, financial crises or pandemics, these numbers are set to grow in the future. Although the oil prices as of late have dropped, they still account for 19% of an airline's operating cost, as reported by the International Air Transport Association (IATA) in their annual review (IATA, 2017). Coupled with an increasing emphasis on reducing carbon footprints and combustion emissions through projects such as Vision 2020 and FlightPath 2050 supervised by the Advisory Council for Aeronautics Research in Europe (ACARE), there is increasing pressure on airline operators to still remain profitable whilst aiming to reduce environmental pollution. It was reported by Cavallaro and Demasi (2016) that without radical changes to the conventional design (monoplane configuration), the improvements would only be minimal. As such, there is renewed interest in unconventional aircraft configurations which may potentially improve aerodynamic performance and decrease fuel consumption. The configuration of choice presented in this thesis is the joined-wing configuration.

1.2 Literature Review

1.2.1 Overview of Joined Wings

The joined-wing concept was first introduced and patented by Wolkovitch in 1976. Wolkovitch (1986) describes it as wings lined up in such a way that both the frontal and planar views result in a diamond shape. This is achieved by extending the horizontal stabilizer, located on the vertical stabilizer, with an anhedral angle and a forward sweep, and connecting it with the main wing, which has a dihedral angle and a rearward sweep. This connection can either be on the tip or any point along the main wing. However, the definition of the joined wing also includes configurations where vertically placed wings join the main wing and the horizontal stabilizer. Figure 1-1 depicts a general configuration.

There are its advantages and disadvantages, which are discussed next. The first advantage is its higher stiffness when compared to its conventional counterpart. However, for meaningful comparison, both configurations should have the same parameters such as lift and drag, or wingspan and aspect ratio (Cavallaro and Demasi, 2016). Samuels (1982) compared an equivalent joined-wing configuration joined at the tips to a conventional monoplane with the same lift and drag. It was found that the joined-wing configuration had a smaller span, lighter structural weight and lower tip deflections through finite element analysis (Samuels, 1982). It is also important to note that under different optimization criteria, such as minimum induced drag, it is also possible to have joined-wing configuration that results in a large deformation despite the increased stiffness. Unlike the conventional wing, which can be modelled as a cantilever, in the joined wings, the front wing is supported by the rear wing. Intuitively,

Fig. 1-1 (a) General joined-wing configuration and (b) control surface deflections for direct lift and lateral force (Wolkovitch, 1986).

this configuration allows it to withstand more bending force than conventional wing. However, one must take caution and consider the whole joined-wing configuration in its entirety and not only focus on the buckling of the supporting wing (Cavallaro and Demasi, 2016). In fact, Cavallaro et al. (2014) showed that stiffening the wing not in compression is more beneficial in eliminating the aeroelastic instabilities.

The second advantage is its lighter weight compared to the conventional configuration. This will not be true for all cases. However, with the proper sizing and designing of geometric parameters, substantial weight savings can be obtained. Miura et al. (1985) showed that the joined-wing configuration enjoy weight savings of around 22% to 35% over the conventional wing configuration, depending on the aspect ratio. Weight can also be saved on the fuselage structure. In the conventional configuration, the wings exert an upward lift force, while the tail exerts a downward trimming force on the fuselage. For the joined-wing configuration, both the wings exert an upward lift force. This results in a smaller net bending moment exerted on the fuselage for the joined-wing configuration. The fuselage for the joined-wing configuration can thus be made lighter.

Fig. 1-2 (a) Span efficiencies, e , of different wing configurations and (b) reduction of induced drag against gap for different configurations (Kroo, 2001).

In the transonic regime, the joined-wing configuration also enjoys smaller wave drag due to the smoother transition of cross sectional area along the plane, according to the Whitcomb area rule. Bushnell (2000) and Sellers III et al. (2003) highlighted the possibility of minimal wave drag through the extreme arrow wing concept, not dissimilar to the joined-wing configuration, introduced by Pfenninger and Vemuru (1988). Further work by Komadina et al. (2002) produced a supersonic joined-wing design suitable for both civilian and military applications. Also, with the additional wings and control surfaces, the joined-wing configuration is more manoeuvrable, due to direct lift and lateral force control. Figure 1-2 shows how the flaps can be orientated to produce the lift and lateral force.

Another advantage is its improved aerodynamic performance in certain conditions. The induced drag of the joined-wing configuration is smaller due to the reduction of the wing tip vortices. This results in a larger span efficiency factor, e , for the joined wing, due to the lower induced drag. The span efficiency factor e is given by the ratio of the induced drag coefficient, C_{Di} , to the induced drag coefficient for an elliptical lift distribution, $C_{Di,e}$. Note that the span efficiency for an elliptic lift distribution on a monoplane is 1. Indeed, this joined-wing concept is not an entirely new concept, merely an extension of the non-planar closed system of lifting surfaces, with other variants such

as the ring wing (Terry, 1964), the box plane (Miranda, 1972) (as cited in Kroo, 2001, pp. 612) the strut- and truss-braced wings (Pfenninger, 1954; Kulfan and Vachal, 1978; Grasmeyer et al., 1998), the SensorCraft (Johnson, 2001; Blair and Canfield, 2002; Martinez et al., 2008), PrandtlPlane (Frediani et al., 2003; Frediani, 2005) and others, which all exhibit the advantage of lower induced drag. Demasi et al. (2014) introduced an analytical formulation for the determination of the minimum induced drag for generic non-planar wing systems, and also verifies the span efficiencies of other non-planar configurations previously introduced. Discussion on the induced drag is discussed in the following section.

During cruise conditions for maximum endurance, with ratio of lift to drag at its highest, the zero lift drag and induced drag are the same in magnitude (Anderson, 2008). However, at lower speeds during takeoff or landing, the induced drag would be much larger compared to the zero lift drag, forming approximately 80% to 90% of the total drag (Kroo, 2001). Hence aircraft designers look to reduce the induced drag, which in turn would improve the range as well as allow for a larger maximum take-off weight with the same climb performance and runway takeoff length.

1.2.2 Parameters of Joined Wings

To better understand the overall performance of the joined wing, a look was first made into the sectional properties. When the joined wing is cut in the stream wise direction along its span to obtain a two dimensional (2D) view of the airfoil profiles, one would notice that it would resemble closely to that of a biplane, albeit with some minor differences. The following will explain some of the parameters specific to the biplane

configuration. Stagger refers to the horizontal placement along the stream wise direction of the upper wing with respect to the lower wing. A positive stagger implies that the upper wing is placed ahead of the lower wing. Negative stagger implies the opposite. Gap, G , refers to vertical distance between the trailing edge of the front wing and the leading edge of the rear wing. Unlike stagger, there is no negative gap. Another parameter similar to gap is the vertical aspect ratio, V , which is the ratio of the gap to the full span. Decalage refers to the angle of attack, α , of the lower wing with respect to the upper wing. A positive decalage means that the lower wing is at a higher angle of attack than the upper wing. Overhang refers to the ratio of the spans of the lower wing to the upper wing. Like the gap, there is no negative overhang. However, a value of less than 1 implies that the lower wing is shorter than the upper wing while a value of more than 1 means that the lower wing is longer than the upper wing. Figure 1-3 illustrates the parameters described.

In addition to aforementioned parameters described, the other changeable parameters include the sweep and dihedral, which are the same for single wing systems. Work has been done on how each of the parameters, as well as combinations of these parameters, affect the lift and drag (Knight & Noyes, 1929a; Knight & Noyes, 1929b; Knight & Noyes, 1929c). The aerodynamics of the flow is that the rear wing would be affected by the downwash from the front wing while the front wing would be affected by the upwash from the rear wing. As such, the more important parameters are the gap, vertical aspect ratio and the stagger.

Fig. 1-3 (a) Stagger (Knight and Noyes, 1929a), (b) gap (Bagwill and Selburg, 1996), (c) decalage, and (d) overhang (Knight and Noyes, 1929b).

Prandtl (1924) concluded that for a fixed span, biplanes have less induced drag than single wing, and a larger gap reduces the induced drag. Prandtl (1924) also concluded that for a fixed span, minimum induced drag is occurs when overhang is 1, i.e. both wings have the same span, and when both wings produce the same lift. Prandtl (1924) goes on to investigate the effects of adding more wings, and eventually arrives at an approximate generalized formula for the best wing configuration, which resembled the joined-wing configuration joined at the tips, except that in this case, the wings are connected to each other with vertical wings. DeYoung (1980) and Demasi et al. (2016a) also showed that this joined-wing configuration joined at the tips, or box-wing configuration, is equivalent to the multi-wing case with an infinite number of same-span wings and vertical aspect ratio.

Fig. 1-4 (a) Diagram of Prandtl's Best Wing System (Prandtl, 1929) and (b) lift distribution for minimum induced drag (Frediani, 2005).

Frediani et al. (1999) confirms Prandtl's findings and explains the lift distribution required. Frediani (2005) describes it as butterfly-shaped on the vertical wings and an overlay of a constant part and an elliptic part on the horizontal wings. Figure 1-4 illustrates the best wing system, as well as the lift distribution required for minimal induced drag. Frediani's results match closely with that of Prandtl at small vertical aspect ratios of 0.1 to 0.2. However, at larger vertical aspect ratios, Prandtl's equation was found to have overestimated the possible reduction in induced drag (Frediani, 2005). As the gap to span ratio tend to infinity, Prandtl (1924) predicted an 84% reduction in the induced drag, whereas Rizzo (2009) (as cited in Schiktanz & Scholz, 2011, pp. 604) predicted 57% reduction in the induced drag. In Prandtl's honour, Frediani dubbed this particular configuration as the PrandtlPlane, but for the purpose of this thesis, it is considered as a variant to the joined-wing aircraft.

However, more recent works have pointed out otherwise. Demasi et al. (2016c) and Demasi et al. (2016d) showed that while the earlier presented lift distribution is an acceptable approximation for the optimal induced drag at lower vertical aspect ratios, it does not hold at larger vertical aspect ratios. As the vertical aspect ratio tends to infinity, the induced drag also tends to 0 (Demasi et al., 2016b; Demasi et al., 2016d). This is due to the reduction in the circulation gradients at the wing tips. The previous works (Frediani, 2005; Rizzo, 2009) were working on the premise where the lift was

equally distributed amongst both the top and bottom wings, which led to penalizing results. While there is an optimal value to the induced drag, Cavallaro and Demasi (2016) explains that there are an infinite number of aerodynamic loads that can achieve this optimal value of induced drag. The key is that adding a constant circulation to the system would not affect the wake, and hence the induced drag.

When considering the joined wing connected with vertical wings, there are two more parameters which can be changed: twist and cant angles, as shown in figure 1-5. The twist angle refers to the angle the tip chord of the upper wing makes with respect to the flight direction when looking down from the top view. A positive twist angle means that the leading edge span would be shorter than that of the trailing edge. The cant angle refers to the angle the vertical wing makes with the vertical axis from the bottom wing. A positive cant angle means that the upper wing is longer than the lower wing.

Another parameter would be the joint location. Wolkovitch (1986), Kroo et al. (1991) recommend a joint location at around 60% to 70% of the span instead of the tip joint as discussed earlier in view of the compromise between aerodynamic and structural constraints. Kroo et al. (1991) further state that the joint location has to be considered on a case-by-case basis and that there is no hard and fast rule on the optimum location. However, Bagwill and Selburg (1997) found that the twist angle did not significantly affect the induced drag at low values of lift coefficient. The cant angle however did play a role indirectly. With a positive cant angle, the span is increased and hence naturally the induced drag is reduced if all other parameters are kept constant. Bagwill and Selburg (1997) concluded that configurations with cant angles greater than 30° , the

Fig. 1-5 (a) Twist angle viewed from the top view and (b) cant angle (Bagwill and Selburg, 1997).

aerodynamic performance can be better than a baseline configuration single wing with the same plan form area but larger aspect ratio.

Indeed, the main parameters that affect are the stagger and gap. Bagwill and Selburg (1996) found that the positive stagger configuration performed better than the negative stagger configuration in terms of the induced drag for a gaps ranging from 0.25 to 1 times of the average of the tip chords of the upper and lower wings. This is also more evident at higher angles of attack, since the gap for the positive stagger configuration increases whereas the gap for the negative stagger configuration decreases. In addition, in the case of a negative stagger, there might be adverse effects at higher angles of attack (Galinski et al., 2017). With a negative stagger, at a higher angle of attack, the front and rear wings are nearly level, and the flow separation from the front wing would impinge and affect the flow on the rear wing. The gap plays a significant role. Bagwill and Selburg (1996) also found that on both a negative and positive stagger, increasing the gap from 0.25 times of the average tip chords to a factor of one brought about a 6%-7% increase in the ratio of the lift to induced drag.

However, the difference in the induced drag between the negative and positive stagger configurations is in contrary with Munk's Stagger Theorem (Munk, 1919). Munk (1919) stated that the induced drag would not change when the lifting surfaces are moved in the stream-wise direction. Khan (2010) showed this numerically. This discrepancy is possibly due to the fact that in Munk's formulation, the wake was assumed to be rigid (Cavallaro and Demasi, 2016). By allowing the wake to be convected by the local free-stream (free wake), the induced drag was found to be reduced at lower vertical aspect ratios (Schirra et al., 2014; Gagnon and Zingg, 2016).

1.2.3 Works done so far

The discussion for the induced drag stems from Prandtl's lifting line theory, which was originally developed for unswept wings, which models the wing as a single horseshoe vortex. For swept wings, the vortex lattice method (VLM), first introduced by Faulkner in 1943 (Mason, 1998), splits up the wing along the span-wise direction into many parts and uses multiple horseshoe vortexes to represent them. The number of horseshoe vortexes used is dependent on the level of accuracy required. Most of the studies mentioned earlier used the VLM to compute the lift and induced drag. VLM gives reasonably accurate results given the fast turnaround time needed for the computations.

However, the VLM faces some limitations, including small angle approximation as well as thin airfoil. It is also not suitable for solving transonic compressible flow, even with the Prandtl-Glauert corrections (Drela & Youngren, 2010). However, this does not mean that the earlier discussion on minimum reduced drag does not hold. The argument still holds due to Munk's Stagger Theorem (Munk, 1919), as long as the lift circulation

is unchanged (Mason, 1997). The sweep angle also does not affect it (Frediani, 2005). Hence, if only searching for an optimal minimum for induced drag, the sweep, dihedral, or twist angles, need not be considered so long as the circulation distribution is achieved (Munk, 1919). In addition, VLM is unable to solve for the zero lift drag as it is based on potential flow, where the flow is inviscid and irrotational. Another disadvantage is the inability to accurately model the effects of the fuselage as well.

To date, Gall (1984), Gall and Smith (1985), Gallman et al. (1993), Bagwill and Selberg (1996), Bagwill and Selberg (1997), Kang et al. (2008), Cuerno-Rejado et al. (2010), Salam and Bil (2012), Waeterschoot (2012) and Jemitola (2012), just to name a few, used VLM for the aerodynamic analysis and conceptual design of the joined-wing design. In addition, Cavallaro (2014) focused on the study of the aeroelastics of joined-wing configurations, developing in-house codes to couple the aerodynamics and the structural deformations. There are a few that used higher fidelity codes that solved the Euler equations of the Navier-Stokes equations, such as Hashimoto et al. (1989), Tyler et al. (2002), Bottoni and Scanu (2004), Sivaji (2004) and Khan (2010), just to name a few, but the numbers are much lower compared to the earlier list. A study that conducted a full Reynolds-averaged Navier-Stokes (RANS) simulation is even far rarer. To the author's knowledge, only Sivaji (2004), and more recently Sieradzki et al. (2016) conducted such simulations, but only limited to a very small number of cases. This is due to the fact that most of the studies conducted were preliminary investigations, where a large number of configurations in a large design space were considered and hence a low computation time is required. However, in this thesis, a more in depth look into the aerodynamic behaviour is taken, so a higher fidelity code that can more accurately model the flow, at the cost of additional computational resources, is used.

Experimentally, wind tunnel experiments have been carried out by Smith and Stonum (1989), Galinski (1992), Corneille (1999), Barcala et al. (2011) and Perez-Alvarez et al. (2016), just to name a few. Scaled flight tests have been carried out by Richards et al. (2012), and Galinski et al. (2014), just to name a few. In addition, IDINTOS (Tuscan Innovative Seaplane, acronym in Italian) was developed in Italy based on the PrandtlPlane concept. It was a light amphibious 2-seater plane (Frediani et al., 2015b). It was a laborious process, from the design stage (Frediani et al., 2015a; Cipolla et al., 2013a), to the testing involving both wind tunnel (Cipolla et al., 2013b) and numerical studies (Viti, 2013; Maggiari, 2013), scaled flight tests by remote control (Frediani et al., 2015a; Frediani et al., 2015b) and finally to the working prototype.

1.3 Research Aims and Scope

As shown in the literature review, most of the previous studies were focused on quickly cycling through a large number of possible configurations in the myriad of parameters, to select an optimal design for a specific condition. Hence, simpler codes were used. Computationally, there were very few studies that ran RANS simulations. Experimentally, most wind tunnel studies were focused on the lift and drag with regards to the angle of attack, but not really looking into any forms of yaw or roll angles. In addition, there is little done in terms of understanding the nature of the flow.

In this thesis, the following would be explored:

1. To elucidate the aerodynamic characteristics associated with both positive and negative stagger joined-wing configurations through force and moment measurements in wind tunnel tests at lower Reynolds number first
2. To corroborate the wind tunnel findings with RANS simulation to gain more insights on the nature of the flow
3. To look into the impacts of utilizing additive manufacturing for fabricating the wind tunnel model

1.4 Organization of Thesis

The thesis is organized as such: In chapter 2, the wind tunnel experimental setup and procedures undertaken is presented. In chapter 3, the numerical setup and procedures undertaken is presented. In addition, the choice of the turbulence model and validation is discussed. In chapter 4, the results for the PP250 model are presented and discussed. In chapter 5, the results for the D150 model, a baseline model, are presented and discussed. In chapter 6, the results for the B150 model, a joined-wing variant of the D150 are presented and discussed. In chapter 7, the findings from a preliminary investigation of the impacts of using additive manufacturing to produce the wind tunnel models are presented. In chapter 8, the conclusion and possible future works is presented.

Chapter 2 Experimental Setup and Procedures

2.1 Overview

The experiments were conducted in a closed-loop wind tunnel, where the wind tunnel test model is mounted on a sting balance in the test section. The aerodynamic forces and moments experienced on the model were measured by the sting balance. In addition, surface oil flow visualization was used to elucidate the surface flow fields. In this chapter, the aforementioned experimental setup and procedures undertaken in this thesis are discussed in detail, which will include the wind tunnel, the sting balance, the wind tunnel model, as well as, the surface oil flow visualization.

2.2 Experimental Apparatus

2.2.1 Close-loop Wind Tunnel

A subsonic closed-looped wind tunnel with a test section size measuring 0.72m (H) $\times$ 0.78m (W) $\times$ 2m (L) was used for this study. A motor drives a fan providing the wind flow in the test section. The maximum operating free-stream velocity is 90m/s, and is controlled manually by adjusting the power and voltage supplied to the motor through a frequency inverter. The speed is determined from the dynamic pressure measured by a pitot static tube located at the start of the test section. The pressure transducer can measure up to 103kPa, with an uncertainty of 1kPa. A stream-wise turbulence intensity level of 0.1% is achieved through the use of a honeycomb flow straightener, three anti-turbulence screens and a 9:1 contraction chamber.

2.2.2 Sting Balance

A sting balance was used to measure the aerodynamic forces and moments acting on the mounted test models. Table 2-1 displays the full scale (FS) limits and accuracy of the sting balance. The angle-of-attack (AOA), α , can be varied from -10° to 25° . The positioning accuracy for the AOA is 0.1° . The sideslip angle, β , can be manipulated from -40° to 40° . The roll angle, ϕ , similarly can be adjusted from -45° to 45° . The positioning accuracy for both the sideslip angle and roll angle is 0.5° . Figure 2-1 shows the coordinate reference system used in this study. The y-axis is in the direction coming out of the page. The right hand rule is used to define the positive moment direction. These variations were done through computer control and all data acquisition was performed by a National Instruments data-acquisition platform using LabView^{TM} software. During the experiments, the data was acquired at a sampling rate of 47Hz for 30 seconds for each AOA. Three different cases were considered: variation of the AOA only, variation of the AOA and sideslip angle and variation of the AOA and roll angle. The aircraft model was designed to be aligned with the angles of the balance system. The sting balance has a cylindrical protrusion which is housed within the cylindrical fuselage. Hence, the angle at which the balance is at directly translates to the angle of the model. However, the centre of gravity (C.G.) of the model did not coincide with the moment reference point of the sting balance. This did not pose a problem for the forces measured, but the pitching and yawing moments had to be corrected for accordingly. The C.G. of the wind tunnel models were ahead of the sting balance moment reference point, seen in figure 2-2, and table 2-2 lists the corrections required.

Fig. 2-1 Coordinate system used in this thesis.

Table 2-1 Limit and accuracy of sting balance

Component	FS limit	Uncertainty of FS
Drag force	500N	0.11%
Lateral force	600N	0.22%
Lift force	600N	0.14%
Rolling moment	50Nm	0.19%
Pitching moment	36Nm	0.14%
Yawing moment	36Nm	0.25%

Fig. 2-2 Shift in moments.

Table 2-2 Correction for moments

Sting (Body Axes)	Wind Tunnel Model
M_y	$M_y' = M_y - F_z \cdot d$
M_z	$M_z' = M_z + F_y \cdot d$

2.2.3 Wind Tunnel Model

2.2.3.1 Wind Tunnel Model Configurations

Three joined-wing test models were studied in this thesis. All three models were provided by the German Aerospace Centre (DLR). All geometric information of the test models were contained in the Common Parametric Aircraft Configuration Schema (CPACS) file format. CPACS is a text based file, based on Extensible Markup Language (XML), and was introduced by Nagel et al. (2012) to tackle the issue of managing the vast amount of data generated when doing a multi-disciplinary optimization (MDO) of an aircraft design. In addition to the geometry, results from aerodynamic analysis can also be stored in CPACS. TiGL Geometry Library (TiGL) can then be used to visualize the geometric information stored in CPACS, and also be exported into neutral Computer-Aided Design (CAD) formats such as Stereo Lithography (STL), which can be used directly for additive manufacturing. This simplifies the design process as all the required information is stored in a single file. Figure 2-3 show how the data is arranged in CPACS and how that data is visualized in TiGL respectively.

The models were then scaled down to fit the wind tunnel test section and minimize wind tunnel wall effects. The PP250 joined-wing design was based on a previous study done by Bottoni and Scanu (2004). For simplification, the fuselage was modelled as a sharp-tipped ogive cylinder with a length of 0.528m, diameter of 0.06m and fineness ratio of 2.5. The fineness ratio of the ogive cone of the fuselage is defined as the ratio of length of the cone measured from the tip to the base to the diameter of the base. While a higher fineness ratio could lead to improved lift-to-drag performances (Mayo, 1965),

Fig. 2-3 (a) Data organization in CPACS (Rizzi et al., 2012) and (b) TiGL view of joined-wing design

Fig. 2-4 CAD representations of (a) PP250, (b) D150 and (c) B150.

the physical geometry was constrained by the size of the test section. Hence, a moderate value of 2.5 was used instead. The second D150 test model is a replica of the Airbus A320 configuration and serves as a baseline case for benchmarking purposes. In this case, the fuselage was not simplified but was a scaled down version of the Airbus A320 fuselage such that it has a comparable fuselage diameter as the PP250 model for a more consistent comparison. There was a sharp curvature at the end of the fuselage to accommodate the sting. Hence, the horizontal tails were not incorporated. Lastly, the B150 test model has the same fuselage as the D150 configuration but with a different joined-wing design. It was designed and optimized by DLR, through the vortex lattice code, Tornado, developed by the Royal Institute of Technology (KTH), Sweden, to have a similar planform area as the D150 test model wings. The B150 and D150 are designed for the same conditions, payload and mission profile. The PP250 was designed to carry 258 passengers, with a range of 11000km and cruising speed of Mach 0.85. The B150/D150 configurations were designed to carry 150 passengers with a slightly shorter

Fig. 2-5 Planform view of PP250.

range of 3500km and a cruising speed of Mach 0.78. The B150 was expected to have a lower drag coefficient at cruising conditions when compared to the D150. Figures 2-4 to 2-7 show the different configurations, and table 2-3 shows the values of the different parameters.

Fig. 2-6 Planform view of D150.

Fig. 2-7 Planform view of B150.

Table 2-3 Wing parameters for the different configurations.

Parameter	PP250	D150	B150
Span [m]	0.372	0.49	0.38
Reference area [m ²]	0.022	0.0256	0.0252
Forward wing sweep [°]	40	27.5	25
Aft wing sweep [°]	-14.9	N.A.	-25
Forward wing dihedral [°]	3.5	5	-1
Aft wing dihedral [°]	-11	N.A.	1
Forward wing root twist [°]	1	4	0.7
Forward wing tip twist [°]	-3.5	0.75	5
Aft wing root twist [°]	1	N.A.	-0.9
Aft wing tip twist [°]	-3	N.A.	4.2
Forward wing taper ratio	0.21	0.25	0.23
Aft wing taper ratio	0.71	N.A.	0.23
Mean aerodynamic chord [m]	0.032	0.06	0.04
Wing tip stagger normalized by MAC	2.9	N.A.	1.9
Wing tip gap normalized by MAC	1.7	N.A.	1
Wing tip decalage [°]	-0.5	N.A.	-0.8
Vertical aspect ratio	0.15	N.A.	0.11

2.2.3.2 Wind Tunnel Model Sizing

The wind tunnel model was tested with a free-stream velocity of 15-60m/s. Assuming the flow along the wind tunnel walls can be represented by the flow along a flat plate, based on the Blasius solution to the laminar boundary layer thickness (Schlichting, 1979), the thickest calculated laminar boundary layer thickness based on the test conditions at the end of the test section is 0.007m. However, a look at the local Reynolds number of 1,950,000 indicates that the boundary layer would most likely be turbulent at that location since it is much larger than the critical Reynolds number at transition of 500,000 for flat plate flow. For the Blasius solution of the turbulent boundary layer thickness (Schlichting, 1979), the thickest calculated boundary layer thickness is 0.041m. The span of the largest model measures 0.49m. This gives a clearance of 0.145m between the wing tips and the wind tunnel walls, which is much larger than the boundary layer thickness from the wind tunnel wall.

While the test velocities and size of the models do not allow for direct scaling and comparison to the actual and full-scale conditions due to the limitations of the wind tunnel, these experiments would still be useful in providing some much needed fundamental understanding of the flow behaviour. In addition, the experimental data would be vital in validating in verifying the accuracy of the numerical model and solver used in this study, which would be described in the next chapter. This would give more confidence in the solver when the full-scale aircraft is simulated. However, caution must be taken when interpreting the wind tunnel results, as the results from the low speed experiments do not directly correspond to the high speed results due to the compressibility effects as well as the thermal effects.

2.2.3.3 Wind Tunnel Model Material Selection

The wind tunnel model fuselage was made out of 6061 aluminium alloy, with a maximum specified surface roughness of Ra 40 microns. The joined wings were manufactured through additive manufacturing with polyamide (PA) 2200, or nylon. This was due to physical scale of the wind tunnel model. The wings were thin, especially at the joints, which made it difficult to manufacture conventionally. The thickest portion of the forward wing tip profile for the PP250 measures 0.002m. Nylon was chosen after various considerations such as material characteristic strength, surface finishing and cost. The intrinsic strength of the material used for the joined wing is crucial, since the full brunt of the aerodynamic forces would be acting on the wings throughout the entire experiment. The nylon material used in this study has a reported flexural modulus of 1500MPa and flexural strength of 58MPa, which were deemed to be sufficiently robust for the wind tunnel experiments to be conducted. Several

assumptions were made in determining the suitability of nylon. The calculations were made based on the D150 model, which was the conventional monoplane design. Since the joined-wing configurations had multiple wings to spread the load and stresses, the monoplane wing should have to withstand more stress. Another assumption made was the even load distribution and constant cross section with no taper or sweep, so that the generalized equation for a cantilever beam can be used (Jones et al., 2004). The cross section taken was at the root, since it is at the root which the highest stress was found, and was assumed to take the area of a rectangle formed by the chord length and the thickest section of the airfoil. The load on the wing as taken to be at a lift coefficient of 2, which would be very high and unlikely, but provides a good safety buffer.

$$\sigma = \frac{Wl}{2Z} \quad (2-1)$$

where σ is the shear stress, W is the load on the beam, l is the length of the beam and Z is sectional modulus of the beam. The calculated stress on each wing at the root was 1.45MPa, which was lower than the given flexural strength of 58MPa. In the course of the experiments, neither of the wings broke, further validating the robustness of nylon.

Surface finishing is critical as well, as it impacts the location and extent of flow transitions, subsequently affecting the resultant aerodynamic behaviour, forces and moments. For a satisfactorily smooth surface finishing, the additive manufactured wings were carefully sanded down using 2000 grit sandpaper. Although Traub (2013) reported that the surface finish of additive manufacturing has minimal effect on the aerodynamic performance, he also noted that some of the surface irregularities, a by-product of this manufacturing technique, may act as localized trip points and should be removed.

Fig. 2-8 DSLR camera and its position circled in red for capturing (a) deflections in the z-axis from the side and (b) span-wise curvatures.

However, as a result of this material selection, substantial wing deflections were observed during testing, and measures had been undertaken to quantify these deflections.

2.3 Capturing Deflections

To quantify any wing deflections, a digital single-lens reflex (DSLR) Canon EOS 550D camera equipped with a 50mm, $f1.4$ lens was used to capture images of the test model from two orientations before and during the experiments. Side-view images of the test model were taken outside of the test-section to quantify deflections in the vertical axis, while span-wise curvatures of the wings were captured by positioning the camera inside the wind tunnel test section. For the second orientation, the camera was mounted onto a plate with pivot joints such that the camera height and angle could be easily adjusted. During the experiments, the plate could be adjusted to match the AOA of the test model so that the camera faced the model normally. The two images are then superimposed and compared through the use of a digitizer and MATLAB^{TM}. Figure 2-8 shows the camera setup by the side of as well as inside the wind tunnel test section respectively.

2.4 Surface Oil Flow Visualizations

To better elucidate the flow features, surface oil flow visualizations were carried out. A quick survey of some existing literature done on surface oil flow visualizations by Maltby (1962), Settles (1986), Merzkirch (1987), Ristic (2007), Lu (2010) and Mahmood (2011), suggests a suspension of fine pigment, depending on the colour of the surface being applied to, in some mixture of mineral oil. The pigments in the oil film applied would leave streaks behind as the flow develops, and the remaining patterns could be photographed for interpretation.

The common pigments include titanium dioxide, which is white in colour and makes it useful when the surface is black, and lampblack (carbon), which is conversely black in colour and gives a good contrast when the surface is white. Alternatively, luminescent pigments can also be used to good effect.

As for the mixture of mineral oil, it depends on the test conditions and varies from case to case. Ideally, the oil should only start flow when the air flow reaches the desired testing conditions. In order to achieve that, the viscosity of the oil can be adjusted by either adding additives, such as naphtha, to thin it or by using a more viscous oil as the base solution. Hence, there is no hard and fast rule for the recipe of the mixture and some trial and error is required before the optimal mixture can be obtained. In addition to the viscosity, as the wind tunnel in this thesis is a closed-loop one, an additional safety consideration of low flammability was taken into account as the oil might be carried downwind to the motor, which can get hot after prolonged usage.

In this study, a mixture of white oil and Carlupe^{TM} oil treatment, a commercially available fuel additive, was used in the ratio of 4:1. The pigment used was a luminescent zinc sulphide, which glows a neon green in the absence of light. The wind tunnel model was spray painted with a matt black finish, to provide a contrast to the luminescent pigment. The entire model was brushed on evenly with the mixture before starting the experiments. As done previously when capturing the wing deflections, the same DSLR camera was placed at the same location to capture the patterns formed through surface oil flow visualization. To capture the patterns on the upper surfaces of the wings, the model was rotated on the sting to face the camera normally. This was done as there is no optical access from the top of the test section of the wind tunnel. As a result, there was some smudging of the patterns caused by the flow of the oil due to gravity when orientating the model to capture the patterns on the upper surfaces of the wings.

Chapter 3 Numerical Setup and Procedures

3.1 Overview

Reynolds-averaged Navier-Stokes (RANS) equations can be obtained through the Navier-Stokes equations by decomposing the variables into their mean and fluctuating components (represented by the overbar and prime symbols respectively), and taking a time average (Reynolds, 1895). This is done due to the inexact nature of turbulence. The resulting equations are as follows:

$$\frac{\partial \rho}{\partial t} + \frac{\partial}{\partial x_i}(\rho u_i) = 0 \quad (3-1)$$

$$\frac{\partial}{\partial t}(\rho u_i) + \frac{\partial}{\partial x_j}(\rho u_i u_j) = -\frac{\partial p}{\partial x_i} + \frac{\partial}{\partial x_j} \left[\mu \left(\frac{\partial u_i}{\partial x_j} + \frac{\partial u_j}{\partial x_i} - \frac{2}{3} \delta_{ij} \frac{\partial u_k}{\partial x_k} \right) \right] + \frac{\partial}{\partial x_j}(-\rho \overline{u'_i u'_j}) \quad (3-2)$$

While this approach may simplify things somewhat, it also gives rise to additional terms, Reynolds stresses, which require additional equations and relations for closure (Wilcox, 1998). This is where turbulence modelling comes about, to attempt to provide the equations needed to solve for these additional terms. One of the common approaches is to employ the Boussinesq hypothesis (Boussinesq, 1877), which introduces the idea of turbulent (or eddy) viscosity, μ_t , and turbulence kinetic energy, k .

$$-\rho \overline{u'_i u'_j} = \mu_t \left(\frac{\partial u_i}{\partial x_j} + \frac{\partial u_j}{\partial x_i} \right) - \frac{2}{3} \left(\rho k + \mu_t \frac{\partial u_k}{\partial x_k} \right) \delta_{ij} \quad (3-3)$$

The Spalart-Allmaras (SA) and Shear-Stress Transport (SST) k - ω turbulence models are based on this approach, and their suitability for this thesis examined. The Spalart-Allmaras model is a one equation model proposed by Spalart and Allmaras (1992). When calculating the Reynolds stresses, the last term in the equation (3-3) is ignored as the turbulence kinetic energy, k , is not calculated in the Spalart-Allmaras model (one equation model), and the $\partial u_k / \partial x_k$ term, mass continuity equation for incompressible flows, should be zero (ANSYS, 2013). The SST k - ω model is a two equation model proposed by Menter (1994). The turbulence kinetic energy, k , and the specific dissipation rate, ω , are modelled in the SST model. The model parameters can also be found in the FLUENT theory guide (ANSYS, 2013).

3.2 Choice of Turbulence Models

Numerical simulation validations against experimental results for the choice of the turbulence models were carried out at transonic Mach numbers for three different cases: 2D airfoil, 3D wing, and 3D wing body, and at a subsonic speed for a 2D airfoil as well. The lift and drag coefficients (C_L and C_D) have been checked for convergence, both within each mesh and between the coarse, medium and fine meshes, and that y^+ is sufficiently low to resolve for the viscous sublayer. The y^+ is described as the non-dimensionalized distance from the wall based on the viscous sublayer (Wilcox, 1998) and can be expressed as the following equation:

$$y^+ \equiv \frac{u_\tau y}{\nu} = \sqrt{\frac{\tau_w}{\rho}} \frac{y}{\nu} \quad (3-4)$$

where y is the distance from the wall, ν is kinematic viscosity, u_τ is the friction velocity and is defined as the square root of the quotient of the dividend, τ_w , the surface shear stress and the divisor, ρ , the density. In addition, for flows experiencing adverse pressure gradients or undergoing separation, wall functions generally do not perform well. Hence the viscous sublayer should be fully simulated with at least 10 nodes within in for accuracy (ANSYS 2013).

3.2.1 High Speed Cases

3.2.1.1 2D Airfoil

The 2D airfoil chosen was the Royal Aircraft Establishment (RAE) 2822 transonic airfoil, and wind tunnel experiments were carried out at Mach 0.725, and an AOA of 2.92° . The airfoil coordinates and wind tunnel results were obtained from Cook et al. (1979). The values of interest were the sectional lift and drag coefficients (C_l and C_d) as well as the coefficient of pressure (C_p). The boundary conditions were corrected for based on steps outlined by Garner et al. (1966) and Barlow et al. (1999). Table 3-1 and figure 3-1 show the comparison between the experimental and simulation results. The results show a slight difference between both turbulence models, though they both overestimate the C_l and C_d . In terms of the C_p distribution, the SA model (blue dashed line) more accurately predicts the shock location as compared to the SST model (green dashed line). However, in terms of absolute values, near the shock location, the SST model is closer to the experimental results (red dots) than the SA model.

Table 3-1 Comparison of wind tunnel results with CFD for RAE 2822.

Parameter	Experiment	CFD SA	CFD SST
C_l	0.7280	0.7397	0.7350
C_d	0.0127	0.0158	0.0153

Fig. 3-1 Comparison of C_p distribution for RAE 2822.

3.2.1.2 3D Wing

The 3D wing chosen was the Office National d'Etudes et de Recherches Aéropatiales (ONERA) M6 wing, and wind tunnel experiments were carried out at Mach 0.84, and an AOA of 3.06° . The airfoil coordinates and wind tunnel results were obtained from Schimtt and Charpin (1979). The values of interest were the distribution of C_p across seven different span-wise locations across the wing, as shown in figure 3-2, where b is the semi-span and y is the distance from the wing root along the span-wise direction. It is found that for this case, the results of both the SA model and SST model overlap each other from sections 1 to 4 and very minimal differences can be seen in sections 5 to 7. The simulation results for sections 1 and 2 correlate well with the experimental data. However, the correlation for sections 3 and 4 are not so ideal. In section 4, the CFD

Fig. 3-2 Location of comparison for C_p for ONERA M6.

simulation do not capture the shock at all. In section 7, there is also a large deviation between the CFD results and experimental data. This can be partly attributed to the fact that the wing actually experienced some deflections in the experiment, which were not modelled in the simulations. In terms of the C_L and C_D , the differences are minimal as well. The SA model predicts a C_L of 0.266 and a C_D of 0.0177, while the SST model predicts 0.263 and 0.0174 respectively.

3.2.1.3 3D Wing Body

The 3D wing body configuration chosen was the DLR F6, and wind tunnel experiments were carried out at Mach 0.752, and an AOA of 0.49° , which was published in the 2nd AIAA CFD Drag Prediction Workshop (DPW). The mesh used, generated by ANSYS with ICEM, was also obtained from the DPW website. The values of interest were the distribution of C_p across eight different span-wise locations across the wing, as shown in figure 3-4, as well as the C_L and C_D . It is found that for this case, there are minimal

Fig. 3-3 Comparison of C_p distribution for M6 wing at (a) section 1, (b) section 2, (c) section 3, (d) section 4, (e) section 5, (f) section 6 and (g) section 7.

Fig. 3-4 Location of comparison for C_p for DLR-F6 (Godard, 2003).

Table 3-2 Comparison of wind tunnel results with CFD for DLR F6.

Parameter	Experiment	CFD SA	CFD SST
C_L	0.4984	0.5192	0.4969
C_D	0.0295	0.0322	0.0290

differences between the results of both the SA model and SST model. The simulation results for sections 2 to 5 correlate well with the experimental data. However, the correlation for sections 1 and 6 to 8 are not so ideal, as seen in figure 3-5. This can again be partly attributed to the wing deflections not accounted for in the simulations. Table 3-2 shows the comparison of the C_L and C_D . The C_L and C_D calculated using the SST model are actually quite close, with only 0.3% and 1.7% difference between the experimental C_L and C_D values. The SA model do not perform as well, with a 4.2% and 9.2% difference respectively. Overall, both the SA and SST models can be used with fairly high confidence. Considering that the SA is only a one equation model, compared to the SST model which is a two equation model, with half the computational resources required and a relatively similar result, the SA model will be used in this thesis.

Fig. 3-5 Comparison of C_p distribution for F6 at (a) section 1, (b) section 2, (c) section 3, (d) section 4, (e) section 5, (f) section 6, (g) section 7 and (h) section 8.

3.2.2 Low Speed Case

3.2.2.1 2D Airfoil

As an additional check to the suitability of the SA turbulence model at lower speeds to suit the wind tunnel conditions, the SD7003 airfoil was chosen. At relatively low Reynolds number (10^5), Ma and Liu (2009) also suggested that RANS with SA would be sufficiently accurate. In addition, Tang (2008) and Catalano and Tognaccini (2010) reported satisfactory results by using the SA turbulence model for the SD7003 airfoil at a Reynolds number of 60,000. Moreover, Rumsey and Spalart (2009) noted that no better models than the SA or SST turbulence models have been identified for low speed flows.

The airfoil geometry and wind tunnel results were obtained from Selig et al. (1995). The sectional lift and drag coefficients were compared, as seen in figure 3-6. Generally, the trends of the C_l and C_d are similar, although the C_d is somewhat overestimated, especially at the lower AOAs. This is because at the lower AOAs, the flow still remains largely laminar, but due to the nature of the numerical simulation, it is assumed to be entirely turbulent. The drag prediction can be improved through the use of transitional turbulence models, which empirically determines the onset of transition through the momentum-thickness Reynolds number. However, that would require additional computational resources, as there two additional equations to be solved.

3.3 Numerical Setup and Grid Independence

As discussed earlier, steady state RANS with SA turbulence model was deemed accurate enough for use in this thesis. In addition, to further ensure accuracy, the

Fig. 3-6 Comparison of (a) C_l and (b) C_d of SD7003 airfoil.

domain boundary was placed at a distance of at least 10 times the fuselage length away from the model, which works out to be more than the recommended length of 100 times the reference chord by Slotnick et al. (2011). In this study, the inlet and outlet were placed 20 times the fuselage length upstream and downstream of the model respectively, while remaining boundaries were placed at a distance of 10 times the fuselage length, as seen in figure 3-7. Inflation layers of prisms were used in order to ensure a y^+ of less than or equal to 1, in line with recommendations for best practices by Slotnick et al. (2011). For the investigation of the change in pitch alone, only half the model was meshed with a symmetric boundary condition enforced in order to save on computational resources required. A grid size sensitivity study with three different sizes for the three different models was conducted at AOA 0° , and the numerical uncertainty is reported in table 3-3 as per standards outlined by Celik (n.d.) for the Journal of Fluids Engineering. Although the grid convergence index for CL for the B150 is very high, on closer inspection, this is due to the oscillatory nature of the solution as the grid size was refined. There is less than 4% difference between the CL of the coarse, medium, and fine meshes. For the investigation with the variation in pitch and roll and pitch and yaw, the same settings used to generate the half body medium mesh were kept when generating the mesh for the full body.

Table 3-3 Comparison of grid discretization error.

Parameter	PP250	D150	B150
Coarse Mesh Size (10^6)	3.8	3.5	5.7
Medium Mesh Size (10^6)	6.7	7.7	12.4
Fine Mesh Size (10^6)	12.6	15.8	20.2
Grid Convergence Index (C_L)	0.43%	0.17%	104.7%
Grid Convergence Index (C_D)	2.9%	0.04%	2.2%

Fig. 3-7 (a) Domain boundary size in terms of fuselage length and (b) surface mesh of the different configurations.

Chapter 4 Aerodynamic Characterization of PP250

4.1 Overview

In this chapter, the results for the PP250 model are presented and discussed. The results are arranged according to the following cases considered: variations in AOA, variations in AOA and sideslip angle as well as variations in AOA and roll angle. The results obtained were from wind tunnel experiments, numerical simulations and surface oil flow visualizations, and are compared side by side for a more holistic analysis.

4.2 Variations in AOA

For this case, the AOA was varied from -10° to 25° . The free-stream velocity for the wind tunnel experiments was set at 30m/s, with a resultant Reynolds number of 62,000 based on the test conditions. The numerical simulations were run based on the same conditions matching that of the wind tunnel experiments for consistent comparison and validation. The surface oil flow visualizations were conducted at selected angles near the stall regions found to better understand the aerodynamic characteristics.

4.2.1 Variations in AOA: Wind Tunnel Results

For the wind tunnel results at $Re=62,000$, displayed in figure 4-1, a linearly increasing trend of the lift coefficient, C_L , is observed. There are no significant drops in the C_L which indicates stall. However, two regions, where the C_L values are found to stagnate, are at the AOAs of 6° to 8° and 18° to 20° . This characteristic differs

Fig. 4-1 Plot of (a) C_L and (b) C_D for PP250 for varying AOA.

significantly from a standard 2D airfoil where a sharp drop the C_L is observed at the stall angle. This serves to show that this configuration will not experience a sudden loss of lift across this range of AOA. Due to limitations of the sting, larger AOAs could not be tested. Due to the nature of the experimental results, how the stall occurs cannot be analyzed in this section, but would be further discussed later on with the aid of supplementary data from the numerical simulations as well as the flow visualizations. The drag polar, however, exhibits a typical parabolic profile.

For longitudinal static stability, the pitching moment slope, $C_{M\alpha}$, has to be negative (Jacobs and Ward, 1935). Figure 4-2 shows the plot of the pitching moment against the effective AOA, α_{eff} , where the zero-lift angle, α_{0L} , has been accounted for. Based on the wind tunnel lift results, the zero-lift angle is approximately $\alpha_{0L} = -1.8^\circ$ and that the current configuration is largely statically stable in the longitudinal direction. However, it has a slightly negative C_M at $\alpha_{\text{eff}} = 0^\circ$. In addition, the pitching moment slope becomes positive between $\alpha_{\text{eff}} 8^\circ$ and 10° , indicating longitudinal instability,

Fig. 4-2 Plot of pitching moment coefficient, C_M , against effective angle of attack, α_{eff} for PP250

which roughly coincides at the region where the C_L starts to stagnate. Note that however, a positive trim AOA of 2° was reported by Bottoni and Scanu (2004) for the same joined-wing configuration. The analysis on the longitudinal stability done by Bottoni and Scanu (2004) were purely based on numerical simulations. In addition, due to the simplification of the fuselage shape taken in this study, this could have led to the discrepancy with previous study.

4.2.2 Variations in AOA: Numerical Simulation Results

When comparing the numerical simulation results with the experimental results, the overall trends are matching. The trends of the drag coefficient, C_D , obtained through numerical simulations are quite favourable and matching above AOA 8° . Below AOA of 8° , there is a slight discrepancy between the wind tunnel and numerical simulations results. This is probably due to the sensitivity and uncertainty of the sting balance at lower angles-of-attack due to the much smaller drag forces. In addition, for the numerical simulations, the entire flow field is assumed to be turbulent, which will

Fig. 4-3 Comparison of (a) coefficient of lift, C_L , and (b) coefficient of drag, C_D , against AOA between experimental and numerical simulation results.

increase the predicted C_D , as this is not the case. In reality, the flow will be laminar initially at some parts before experiencing transition and becoming turbulent. For the C_L , the numerical simulations over-predict the first stall angle and fails to identify the second stall angle. The discrepancy in the stall angle with the experimental results is likely to be caused by the deflections of the 3D-printed wing. In the numerical simulations, the shapes of the wings are assumed to be rigid. However, this is not the case in reality. Due to it being 3D-printed out of a slightly flexible material, discernible wing deflections were observed in the $Re=62,000$ case as the angle-of-attack increased above 8° .

Although the numerical simulations overestimate the stall angles, an insight can still be obtained from how the stall occurs and develops. C_p plots of the upper surface of the wings were plotted at 4° , 8° , 10° and 12° AOAs, where 10° AOA is identified as the stall angle from the numerical simulations. The locations of the wings chosen were $1/4$, $1/2$ and $3/4$ of the semi-span. From figure 4-4, the numerical simulations suggest that leading edge stall first starts to occur at $1/4$ of the semi-span of the front wing as seen by the characteristic kink in the C_p plot. As the AOA increases, the characteristic kink

becomes less pronounced. Since it is near the fuselage, the flow around that junction is heavily influenced by the flow upstream at the sharp ogive nose. As seen from figure 4-5, which is a plot of the streamlines at junction between the wing root and the fuselage, as the AOA increases, the recirculation region travels further upwind towards the leading edge, which is typical for wings mounted low on the fuselage (Jacobs and Ward, 1935). For the rear wing, leading edge separation is not observed. The flow across the rear wing is generally smooth for 4° to 10° AOA, shown in figure 4-6. As the AOA is increased, the wingtip vortices get stronger, as seen in the increase of the curvature of the streamlines towards the inboard section of the wing. At 12° AOA, the flow starts to separate near the $3/4$ semi-span location, resulting in a span-wise flow into the inboard section, causing a reduction in the lift generation at the $1/2$ semi-span location, as seen in the sharp decrease in magnitude of C_p in figure 4-4.

Fig. 4-4 C_p plots at (a) 1/4 semi-span of front wing, (b) 1/4 semi-span of rear wing, (c) 1/2 semi-span of front wing, (d) 1/2 semi-span of rear wing, (e) 3/4 semi-span of front wing, (f) 3/4 semi-span of rear wing.

Fig. 4-5 Plot of streamlines and C_p at junction between the front wing and fuselage at (a) 4°, (b) 8°, (c) 10° and (d) 12° AOA.

Fig. 4-6 Plot of streamlines and C_p at along the rear wing at (a) 4°, (b) 8°, (c) 10° and (d) 12° AOA.

4.2.3 Variations in AOA: Surface Oil Flow Visualization Results

The surface oil flow visualization served to better understand the flow characteristics across the range of AOA. The angles chosen were based on the stall angles found through the wind tunnel tests, 6° to 8° and 18° to 20°. Additional points were chosen before and after stall. However, due to limitations in the experimental setup, a lack of

optical access necessitated the need to rotate the model. Hence some of the patterns were smudged due to gravity which makes the analysis more complicated.

At AOA 0° , two distinct lines devoid of pigments on the front wing can be seen, shown in 4-7(a). These two lines indicate both a transition from laminar to turbulent boundary layer and flow reattachment. This flow transition is due to the adverse pressure gradient present in the region ahead of the wing on the fuselage (Boermans et al., 1998), as seen in figure 4-8. A similar trend can also be seen on the rear wing, where a similar line spanning the rear wing indicates the laminar to turbulent transition. The reattachment suggests a laminar separation bubble forming.

At AOA 4° , both the transition point and the flow separation on the front wing are moved ahead as compared to that at AOA 0° . The turbulent transition on the rear wing less uniform and becomes wavier as compared to previously at AOA 0° . As the AOA increases to 6° , the flow separation on the outboard section of the front wing is more pronounced as compared to at AOA 4° . Onset of wingtip stall can also be detected. The pattern on the rear wing is similar to that at AOA 4° . Further increasing the AOA to 8° , the flow separation on the inboard section near the wing root become more pronounced. Wingtip stall develops and expands inwards. The patterns on the rear wing were too smudged to be analyzed. At AOA 10° , the situation at the wing root remains similar to that at AOA 8° . However, the stall region developing from the wingtip expands further inwards. Likewise, the patterns on the rear wing could not be analysed due to the smearing.

Fig. 4-7 Surface oil flow visualizations for (a) 0° AOA, (b) 4° AOA, (c) 6° AOA, (d) 8° AOA, (e) 10° AOA, (f) 18° AOA, (g) 20° AOA and (h) 24° AOA.

For the second stall region identified, for AOAs 18°, 20° and 24°, the oil flow patterns for the front wing are generally similar. The oil can be seen accumulating in pools in the

Fig. 4-8 Flow pattern observed on wing of sailplane (Boermans et al., 1998).

intersection between the wing root and the fuselage, suggesting a recirculation zone. The main difference is in the location. For AOA 18° , it is located near the leading edge, while for AOAs 20° and 24° , the pools are located further downstream near the trailing edge. For the rear wing, at AOA 18° , an accumulation of oil can be seen near the trailing edge of the wing root. At AOA 20° , at the same area, a swirling pattern can be seen with a visible focal point. Further increasing to AOA 24° , the swirling pattern breakdowns and spreads outwards towards the outboard section of the wing.

When comparing the surface oil flow visualizations patterns with the surface streamlines based on the wall shear obtained from the numerical simulations, some similarities and differences can be observed. For AOA 0° , the transition lines cannot be seen as the flow was assumed to be fully turbulent in the numerical simulations. In addition, to verify that transition is indeed taking place, a 2D numerical simulation of the root chord was conducted. The use of the transition model was only one-off. It was not used for the other 3D cases to reduce the computational resources required, as an additional 2 variables have to be modelled. On the whole, the fully turbulent model is still able to predict the lift and drag forces fairly well, although when comparing against the flow visualization experiments, there were some discrepancies. The transition SST turbulence model by Langtry and Menter (2009) was used. To model the transition, two more parameters, γ and Re_0 , which were the intermittency and the transition onset

Fig. 4-9 (a) Mesh of airfoil, (b) wall shear stress across the upper surface of the airfoil, (c) streamlines across the airfoil and (d) plot of intermittency, γ .

criteria. An intermittency of 0 represented laminar flow, while a value of 1 represented turbulent flow. Any value in between would indicate transition. The domain size was set at 10 chord lengths upstream, above and below the airfoil, and 20 chord lengths downstream. An unstructured quadrilateral mesh was created, with inflation layers added to ensure a low y^+ for accuracy. Eggenpieler (2012) recommended a y^+ of less than 1 and a cell expansion ratio across the wall of about 1.1, which were adhered to when generating the mesh. The resulting mesh had an element count of 57,000, and the y^+ value across the airfoil was 0.16 at its peak value. Based on the wall shear stress across the airfoil surface, it reaches 0 at two points: 0.475 and approximately 0.9 of the chord, as seen in figure 4-9(b). The first point when it reaches zero represents the separation point, while the second point represents the reattachment point. The point where the wall shear experiences a sudden increase at approximately 0.7 of the chord

should represent the transition from laminar to turbulent. Looking at the surface oil flow visualizations, the transition and reattachment point appears to be slightly ahead of what the numerical simulations predict. However, note that in the experiments, the flow is 3D in nature and hence is affected by the adverse pressure gradient due to the junction between the fuselage and the wings.

For AOA's 4° and 6° , the flow separation along the front wing closely resembles that of the surface oil flow visualizations. In the numerical simulations, this separation line can be seen clearly across the entire wing. However, in the surface oil flow visualizations, this is not so obvious due to the smudging of the oil due to gravity. As for AOA 8° , the flow separation near the wing root was accurately predicted. However, for AOA 10° , the numerical simulations severely overestimate the degree of flow separation. In the surface oil flow visualizations, the flow separation starts to occur near the mid chord of the wing root. However, in the numerical simulations, the flow separation is seen to occur near the leading edge instead.

For the higher AOA's, the degree of accuracy of the numerical simulations is not so desirable, as seen from the failure to predict the second stall region. The degree of flow separation on the front wing is under predicted. As for the rear wing, at AOA 18° , the accumulation of oil near the trailing edge of the wing root can be seen in the numerical simulations. In addition, the swirling pattern which occurs at AOA 20° in the surface oil flow visualizations is reproduced in the numerical simulations, albeit at a higher AOA of 24° . This shows the suitability of this particular simulation and turbulence model selection at lower AOA's and its limitations at higher AOA's.

Fig. 4-10 Surface streamlines based on wall shear for (a) 0° AOA, (b) 4° AOA, (c) 6° AOA, (d) 8° AOA, (e) 10° AOA, (f) 18° AOA, (g) 20° AOA and (h) 24° AOA.

4.3 Variations in AOA and Sideslip Angle

For the wind tunnel experiments, the AOA and sideslip angle were varied from -10° to 25° and 5° to 30° respectively at 5° intervals for a combined 8×6 test matrix. The numerical simulations and surface oil flow visualizations were run based on the same

Fig. 4-11 Plot of (a) C_L and (b) C_D for PP250 for varying β .

conditions as the wind tunnel experiments, but only for two sideslip angles of 15° and 30° .

4.3.1 Variations in AOA and Sideslip Angle: Wind Tunnel Results

Figure 4-11 displays the wind tunnel results at $Re=62,000$, for the various AOAs, as the sideslip angle β changes. It is found that the C_L remains relatively constant at the given AOA as the sideslip angle β increases. This demonstrates the lift independence of this configuration on the sideslip angle β , making it stable to sudden changes in the lateral direction of the oncoming free-stream. The resulting flow separation behind the fuselage region has little impact on the lift generation from the wings. The C_D increases as the sideslip angle β increases, as a larger proportion of the fuselage is opposing the oncoming free stream.

Fig. 4-12 Plot of (a) yawing moment coefficient, C_N and (b) rolling moment coefficient, C_l , against sideslip angle, β .

For the directional stability, the yawing moment slope, $C_{N\beta}$, should be positive, to exhibit the 'weathercock' effect and push the aircraft back into the direction of the incoming flow, while for the lateral stability, the rolling moment slope $C_{l\beta}$, should be negative (Nelson, 1989). Figure 4-12 shows the both the plot of the yawing moment, C_N , against the sideslip angle, β , and the plot of the rolling moment, C_l , against the sideslip angle, β .

The resulting $C_{N\beta}$ is generally positive, which indicates directional stability. Any gust will be mitigated even if no corrective measures are taken. However, there are still instances where the $C_{N\beta}$ becomes 0 and even negative. These instances occur at AOA 15°, when β is more than 25° and at AOA 25°, when β ranges between 10° and 20°. This is still satisfactory as the aircraft is not expected to operate at these range of angles for a prolonged period of time.

The resulting $C_{l\beta}$ is negative, which indicates lateral stability. However, at 0° AOA, it is almost statically neutral, as compared to the higher AOAs, with an almost flat $C_{l\beta}$ slope. This implies that the roll situation would not worsen if left unchecked and manual pilot

Fig. 4-13 Comparison of coefficients between wind tunnel (WT) and CFD results for (a) $\beta=15^\circ$ and (b) $\beta=30^\circ$.

intervention would be required to right the perturbation at 0° AOA. In addition, in the range of the AOAs tested, between a sideslip angle of 5° and 10° , $C_{l\beta}$ is flat. This indicates that this configuration is susceptible to spiral instability due to the dominance of directional stability.

4.3.2 Variations in AOA and Sideslip Angle: Numerical Simulation

Results

For the comparison between the wind tunnel (WT) results and the numerical simulation (CFD) results where the AOA and sideslip angle were varied, the trends for the C_L , C_D , and C_Y are similar for the sideslip angle of 15° . The C_L slope has two distinct gradients, and the change occurs around AOA 10° , where stall occurs when purely varying the AOA as seen in the previous section. The gradient decreases after AOA 10° . The C_D shows a similar parabolic trend, while the C_Y remains relatively constant. Beyond the sideslip angle of 15° , there is a large discrepancy especially in the C_L . Figure 4-13 shows the comparison between the WT results and CFD results for the sideslip angles of 15° and 30° . One of the reasons for the large discrepancy found for case of varying

the sideslip angle is probably due to the blockage of the wind tunnel. Upon increasing the sideslip angle, the amount of blockage of the cross section of the wind tunnel increases, from not just the model, but also the sting balance. This increase in the blockage causes the true free stream velocity across the test section to increase, causing the numerical results to under predict the forces experienced on the model.

Based on the numerical simulations, the flow fields could be visualized for both sideslip angles of 15° and 30° . AOAs of 0° , 10° and 20° were chosen as based on the wind tunnel and numerical simulation results, the gradient of the lift slope changes after AOA of 5° . The flow fields were generated over the front wings and the rear wings separately for clarity. For the case with the sideslip angle at 15° , shown in figure 4-14, at AOA 0° , the flow across both the front and rear wings are rather clean with no span-wise flow across them. At AOA 10° , some span-wise flow starts to develop across the leeward front left wing, which is angled further behind to the incoming free stream as compared to the windward front right wing. Some signs of flow separation can also be seen on the windward front right wing, near the midsection of the wing. In addition, flow separation is also observed on the outboard section of the leeward rear left wing. This causes the lift generation to reduce. When the AOA is further increased to 20° , the magnitude of the span-wise flow on the front wings increases. The span-wise flow can be seen developing right from the leading edge of the leeward front left wing root. Span-wise flow also occurs along the windward rear right wing. The flow separation on the outboard section of the leeward rear left wing is also intensified and widened, as seen in the change in the C_p distribution. Furthermore, due to the inherent geometry of this geometry, above AOA 10° , the flow past the rear wings is influenced by the wake

Fig. 4-14 Flow fields generated for $\beta=15^\circ$, (a) AOA 0° , (b) AOA 10° and (c) AOA 20° .

region of the front wings. The loss of the lift generation on the wings due to the span-wise flow is probably the cause of the reduction in the lift curve slope after AOA 10° .

For the case with sideslip angle at 30° , the similar trend as the 15° case is observed, as seen in figure 4-15. However, at AOA 0° , the vortex formed by the ogive front of the fuselage can be seen to be impinging on the junction between the left wing root and the fuselage. The flow across the rear wings is generally clean and attached. The little twists of the streamlines towards the back of the rear wings are due to the wing tip vortices as well as the junction flow between the rear wing and the fin.

At AOA 10° , the roll up vortex from the ogive front accelerates the flow past the leading edge of the left wing and also results in a large degree of span-wise flow across the leeward front left wing. On the windward front right wing, some span-wise flow near the trailing edge can be seen. This suggests flow separation is occurring near the trailing edge. Towards the rear of the model, the most obvious flow structure would be the flow separation with a recirculation zone across the rear fin. A similar flow separation region is also found on the leeward rear left wing just above the fin.

At AOA 20° , the accelerated flow across the leeward front left wing is much larger than that at AOA 10° , as seen from the colour of the streamlines as well as the C_p distribution. On the windward front right wing, the trailing edge stall has developed into full blown leading edge stall at approximately one quarter the semi-span location. The flow separation region on the leeward rear left wing remains at relatively the same location as compared to the AOA 10° case. However, when comparing this flow separation location with the case where the sideslip angle was at 15° and AOA at 20° , its location is much further inboard as compared to the latter. In general, the degree of the flow separation in the case of the 30° sideslip angle is more severe, which leads to a reduced C_L and increased C_D when compared to the case of the 15° sideslip angle.

Fig. 4-15 Flow fields generated for $\beta=30^\circ$, (a) AOA 0° , (b) AOA 10° and (c) AOA 20° .

4.3.3 Variations in AOA and Sideslip Angle: Surface Oil Flow Visualization Results

The surface oil flow visualization serves to better understand the flow characteristics across the range of AOA, although some of the inevitable smudging makes the analysis slightly more complex. Figure 4-16 shows the case of 15° sideslip angle while figure 4-17 shows the case of 30° sideslip angle for the range of AOAs tested. For all cases, the radial position of the flow separation line on the fuselage decreases as the AOA increases. The flow separation line creeps closer to the right side of the fuselage. As the AOA is increased, the stagnation point where the incoming free stream encounters the fuselage head point is shifted downwards, hence the flow has to travel a slightly longer distance radially to reach the same point.

For the case of 15° sideslip angle, at AOA 0° and AOA 5°, the flow patterns on the wings are similar to that of one with no sideslip angle, the difference being it slightly skewed due to the sideslip. At AOA 10°, span-wise flow can be detected on both the front wings through the presence of lines devoid of pigment. The span-wise flow on the windward front right wing starts at approximately 0.6 of the semi-span while it starts near the wing root on the leeward front left wing. The large accumulation on of oil on the both of the rear wings indicates a region of separation with re-circulation. It is interesting to note that the pools of oil accumulate near the location of the junction between the fin and the wing. The local flow properties must have been affected by the flow near this junction. In addition, another accumulation point can be seen on approximately 0.7 of the semi-span of the leeward rear left wing. As the AOA increases from 10° to 15° and 20°, the oil patterns observed on the wings were same in the nature. The main difference is that as the AOA increases, the location at which the span-wise

flow develops on the windward left front wing moves progressively inboard towards the fuselage.

For the case of 30° sideslip angle, at AOA 0° , the flow patterns on the wings are slightly similar to that of one with no sideslip angle. Aside from the skewed patterns, the reattachment line is not observed on the leeward front left wing. In addition, an accumulation of oil is observed on the leeward rear left wing, just above the fin. At AOA 5° , a clear line devoid of pigment is seen traversing stream-wise across the leeward front left wing at approximately 0.25 of the semi-span. This is postulated to be formed due to the roll up vortex formed from the ogive front of the fuselage impinging on the wing. At AOA 10° , this flow pattern is also observed. In addition, span-wise flow is also detected on both the front wings at AOA 10° . Location-wise, they are similar to that of the case where the sideslip angle was 15° at the same AOA. At AOA 15° , two spots are identified as possible centres for the respective vortices to develop. One is near the leading edge at approximately 0.2 of the semi-span while the other is on the trailing edge at approximately 0.4 of the semi-span. Span-wise flow is also observed on the windward rear right wing, traversing approximately 60% of the semi-span. At AOA 20° , the span-wise flow on the windward rear right wing extends to up to approximately 80% of the semi-span. In addition, there is an extra region void of oil and pigment near the wing root of the leeward front left wing. This is probably due to the reattachment of the recirculating flow at fuselage-wing junction.

When comparing the surface oil flow visualizations patterns with the surface streamlines based on the wall shear obtained from the numerical simulations, some similarities and differences can be observed. The flow separation patterns on the

fuselage look similar to the wall shear patterns obtained from the numerical simulations. The main difference is that the numerical simulations tend to over predict the locations of the separation lines. This is understandable as the flow is assumed to be entirely turbulent in the simulations, so the flow can remain attached for a longer distance as compared to the experiments. However, there are several points where the simulation failed to predict certain regions of the flow separation with recirculation, especially at fin-rear wing junctions.

Nonetheless, it still proves a useful tool in complementing the analysis of the surface oil flow patterns. The flow field is 3D in nature, but the surface oil flow visualizations can only capture and display information in 2D, which makes interpreting an abstract and complicated task. For example, in figure 4-18, some examples of how the surface oil flow patterns could have been formed were explained through the streamlines obtained through the numerical simulations. Four distinct features were chosen from two particular cases, where the sideslip angle was fixed at 30° , while the AOAs were varied between 10° and 20° .

In figure 4-18(a), at a sideslip angle of 30° and an AOA of 10° , two clear lines devoid of pigment are observed on the leeward front left wing. The line closer to the fuselage is likely to be formed due to the acceleration imparted by the roll up vortex created from the ogive front of the fuselage impinging on the wing. On the other hand, the other line is formed by the characteristic span-wise flow seen in typical swept back wings.

In figure 4-18(b), at a sideslip angle of 30° and an AOA of 20° , on the leeward front left wing, two clear lines are similarly observed. This time around, the former line is formed

when the flow across the fuselage is drawn in towards the junction between the wing and fuselage due to the pressure gradient. This flow then merges with the span-wise flow.

In figure 4-18(c), on the windward rear right wing, a forked line is observed forming. The fork is likely formed due to the acceleration of the flow near the junction between the vertical fin and the wing. In figure 4-18(d), the flow structure and recirculation on the leeward rear left wing are displayed. Based on the streamlines, "tornado-like" vortices are formed from the foci and link together to form an "arch-like" structure.

Fig. 4-16 Comparison of coefficients between surface oil flow visualizations and CFD results for $\beta=15^\circ$, (a) 0° AOA, (b) 5° AOA, (c) 10° AOA, (d) 15° AOA, (e) 20° AOA.

Fig. 4-17 Comparison of coefficients between surface oil flow visualizations and CFD results for $\beta=30^\circ$, (a) 0° AOA, (b) 5° AOA, (c) 10° AOA, (d) 15° AOA, (e) 20° AOA.

Fig. 4-18 Possible flow structures and streamlines for explaining surface oil flow patterns.

4.4 Variations in AOA and Roll Angle

For the wind tunnel experiments, the AOA and roll angle were varied from -10° to 25° and -5° to -30° respectively at 5° intervals, forming an 8×6 test matrix. The numerical simulations and surface oil flow visualizations were run based on the same conditions as the wind tunnel experiments, but only for two roll angles of -15° and -30° .

4.4.1 Variations in AOA and Roll angle: Wind Tunnel Results

Figure 4-19 displays the wind tunnel results at $Re=62,000$, for the various AOAs, as the roll angle ϕ changes. It is found that both C_L and C_D remain relatively constant at the given AOA as the magnitude of the roll angle ϕ increases, although there is a slight decrease in the C_L as the roll angle ϕ goes past -20° . There is no sudden decrease in the C_L , which makes this configuration less susceptible to stall due to the roll angle ϕ and more suitable for aggressive roll manoeuvres at large AOAs. The relatively constant C_D also suggests that the degree of flow separation remains relatively the same.

4.4.2 Variations in AOA and Roll Angle: Numerical Simulation Results

For the comparison between the WT and CFD results where the AOA and roll angle were varied, the trends and magnitudes for the coefficients are similar, even at the largest roll angle of -30° , as shown in figure 4-20. The C_L slope looks similar to the case when purely varying the AOA as seen in the previous section. There is a similar stall region between AOAs 5° and 10° , but no stall could be seen between AOAs 20° and 25° . The C_D shows a similar parabolic trend. The C_Y remains relatively constant after

Fig. 4-19 Plot of (a) C_L and (b) C_D for PP250 for varying ϕ .

AOA 5°. Note that the discrepancy here is not as large as in the case of varying the sideslip angle, further substantiating the fact that the cause for the discrepancy for the between the WT and CFD results is due to the increased blockage.

Based on the numerical simulations, the flow fields were visualized for both roll angles of -15° and -30°. AOAs of 0°, 10° and 20° were chosen as based on the wind tunnel and numerical simulation results, the AOA of 10° is the critical point where the gradient of the lift slope changes. For clarity, the flow fields were generated over the front wings and the rear wings separately. It has been noted that the flow fields for both roll angles of -15° and -30° are very similar to each other, with only minor and subtle differences. It should be expected, given the similar lift curve slopes. At AOA 0°, the flow across both the front and rear wings are rather clean with no span-wise flow across them. At AOA 10°, span-wise flow can be observed originating from the trailing edge of the wing root, traversing the entire span of the front wings. On the rear wings, span-wise flow is observed on the midsection. At AOA 20°, the span-wise flow across the front wings have edged upstream towards the leading edge, spanning for approximately half of the outboard segment of the wings. On the rear wings, flow separation with recirculation is observed. The recirculation is seen to developed from the rolled up

Fig. 4-20 Comparison of coefficients between wind tunnel (WT) and CFD results for (a) $\phi = -15^\circ$ and (b) $\phi = -30^\circ$.

vortices formed near the wing tips. The recirculation zone on the rear right wing covers nearly the entire right wing, whereas the recirculation on the rear left wing spans approximately half of the left wing. This is probably because the rear right wing, with an anhedral angle of 11° and coupled with the extra rotation caused by the roll, forms a large angle with the free-stream whereas the roll cancels out the anhedral angle of the rear left wing.

Fig. 4-21 Flow fields generated for $\phi = -15^\circ$, (a) AOA 0° , (b) AOA 15° and (c) AOA 25° .

Fig. 4-22 Flow fields generated for $\phi = -30^\circ$, (a) AOA 0° , (b) AOA 15° and (c) AOA 25° .

4.4.3 Variations in AOA and Roll Angle: Surface Oil Flow Visualization Results

Figure 4-23 shows the case of -15° roll angle while figure 4-24 shows the case of -30° roll angle for the range of AOAs tested. Flow separation on the fuselage is only

observable at AOAs 15° and 20° for both roll angles. For both roll angles at AOA 0° and AOA 5° , the flow patterns on the wings are similar to that of one with no roll angle. The main difference is the increased waviness of the turbulent transition line on the rear right wing which has been rolled to a lower level of elevation. At AOA 10° for both roll angles, span-wise flow can be detected on both the front wings through the presence of lines devoid of pigment. The patterns are not very obvious due to the smudging of the patterns. In addition, with the heavy smudging of the patterns on the rear right wing due to an accumulation of the oil during the experiment, this suggests a large region of separation with re-circulation on it, especially near the wing root. At AOA 15° for both roll angles, the span-wise flow and pattern on the front wings becomes much more obvious. Likewise, an accumulation of oil on the rear right wing is again observed near the wing root, indicating a region of separation with re-circulation. At AOA 20° for roll angles, the span-wise flow on the front wings that was seen at the lower AOAs is also detected. For the -15° roll case, a triangular swirl is observed on the rear right wing, near the location where the fin connects with the rear right wing. For the -30° roll case, a spot located near the trailing edge at approximately half of the semi-span of the left rear wing is identified as the centre at which the recirculation develops from.

When comparing the surface oil flow visualizations patterns with the surface streamlines based on the wall shear obtained from the numerical simulations, some similarities and differences can be observed. Although not very obvious, the wall shear streamlines on the rear wing tips for AOAs 10° to 20° match closely to the curved streaks produced through the surface oil flow visualizations due to the wing tip vortices. In addition, the triangular swirl observed at -15° roll angle and AOA 20° is also reproduced in the numerical simulations. In addition, the flow separation patterns on the

fuselage look similar to the wall shear patterns derived from the numerical simulations. However, at AOA 10° for both cases, where the numerical simulations predict flow separation on the fuselage, the surface oil flow visualizations are not able to show. This is probably due to the nature of the numerical simulations; the fully turbulent assumption causes the simulation to over predict the separation location. In addition, due to the orientation at which the images were taken, any separation occurring on the lower half of the fuselage below the horizontal axis would not be seen. Furthermore, there are several points where the simulation failed to predict certain regions of the flow separation with recirculation, especially at the rear right wing root at AOAs 10° and 15° for both roll cases.

Fig. 4-23 Comparison of coefficients between surface oil flow visualizations and CFD results for $\phi = -15^\circ$, (a) 0° AOA, (b) 5° AOA, (c) 10° AOA, (d) 15° AOA, (e) 20° AOA.

Fig. 4-24 Comparison of coefficients between surface oil flow visualizations and CFD results for $\phi = -30^\circ$, (a) 0° AOA, (b) 5° AOA, (c) 10° AOA, (d) 15° AOA, (e) 20° AOA.

Chapter 5 Aerodynamic Characterization of D150

5.1 Overview

In this chapter, the results for the D150 reference model are presented and discussed. The results are arranged according to the following cases considered: variations in AOA, variations in AOA and sideslip angle as well as variations in AOA and roll angle. The results obtained were from wind tunnel experiments, numerical simulations and surface oil flow visualizations, and are compared side by side for a more complete analysis.

5.2 Variations in AOA

For this case, the AOA was varied from -10° to 25° . The free-stream velocity for the wind tunnel experiments was set at 30m/s, with a resultant Reynolds number of 116,250 based on the test conditions. The numerical simulations were run based on the same conditions matching that of the wind tunnel experiments for consistent comparison and validation. The surface oil flow visualizations were conducted at selected angles before, during and after the stall regions found to better understand the aerodynamic characteristics.

5.2.1 Variations in AOA: Wind Tunnel Results

For the wind tunnel results at $Re=116,250$, displayed in figure 5-1, a linearly increasing trend of the lift coefficient, C_L , up to an AOA of 9° is observed. After AOA of 9° , a region where the C_L stagnates is found, up to an AOA of 14° . After which, the C_L is

Fig. 5-1 Plot of (a) C_L and (b) C_D for D150 for varying AOA.

seen to dip and reaches its local minimum at an AOA of 16° . Subsequently, the C_L is observed to increase again. Eder et al. (2006) also reported similar findings in the behaviour of the C_L with a similar wing-fuselage model without the vertical tail at a higher Reynolds number, but their investigation stopped at AOA 15° . This characteristic differs slightly from a standard 2D airfoil where the C_L does not recover and increase after the stall angle. In comparison with the PP250, for the D150, the region of linear increment of the C_L is larger. Due to limitations of the sting, larger AOAs could not be tested. Due to the nature of the experimental results, how the stall occurs cannot be analyzed in this section, but would be further discussed later on with the aid of supplementary data from the numerical simulations as well as the flow visualizations. The drag polar exhibits a typical parabolic profile.

For the D150 configuration, the pitching moment slope, $C_{M\alpha}$, is negative, and the pitching moment, C_M , about the C.G. at 0° AOA is also negative. Figure 5-2 shows the plot of the pitching moment against the effective AOA, α_{eff} , where the zero-lift angle, α_{0L} , has been accounted for. Based on the wind tunnel lift results, the zero-lift angle is approximately $\alpha_{0L} = -2.4^\circ$ and that the current configuration is largely statically stable in

Fig. 5-2 Plot of pitching moment coefficient, C_M , against effective angle of attack, α_{eff} for D150

the longitudinal direction. With a slightly negative C_M at $\alpha_{\text{eff}}=0^\circ$, implying that it cannot be trimmed for level flight without incorporating additional flight controls. However, note that the horizontal tail was omitted in the test model due to physical constraints. With the horizontal tail added, although the zero lift angle would be slightly increased due to the downward force exerted by the horizontal tail, this downward force would add a positive restoring moment to the aircraft, hence allowing for trimmed level flight. In addition, the pitching moment slope becomes positive between α_{eff} 11° and 17° , indicating longitudinal instability, which again roughly coincides at the region where the C_L starts to stagnate.

5.2.2 Variations in AOA: Numerical Simulation Results

When comparing the numerical simulation results with the experimental results, the overall trends are matching. The trends of the drag coefficient, C_D , obtained through numerical simulations are quite favourable and matching above AOA 8° . Below AOA of 8° , there is a slight discrepancy between the wind tunnel and numerical simulations

Fig. 5-3 Comparison of (a) coefficient of lift, C_L , and (b) coefficient of drag, C_D , against AOA between experimental and numerical simulation results.

results. This is probably due to the fact that for the numerical simulations, the entire flow field is assumed to be turbulent, which would over predict C_D , as this is not the case. In reality, the flow would be laminar initially at some parts before becoming turbulent. For the C_L , the numerical simulations manage to identify the both the AOAs where the C_L first starts to stagnate at AOA 10° as well as the local minima at AOA 16°. The discrepancy is that between the AOA where the C_L first starts to stagnate and the local minima, in the wind tunnel results, the C_L remains relatively constant whereas in the numerical results, the C_L actually decreases. This is likely to be caused by the deflections of the 3D-printed wing. In the numerical simulations, the shapes of the wings are assumed to be rigid. However, this is not the case in reality. Due to it being 3D-printed out of a slightly flexible material, discernible wing deflections were observed in the $Re=116,250$ case as the angle-of-attack increased above 8°.

Based on the numerical simulations, the stall mechanism is investigated and explored. C_p plots of the upper surface of the wings at 1/4, 1/2 and 3/4 of the semi-span are plotted for AOAs of 4°, 8°, 10°, 16° and 20°, where 10° AOA is identified as the onset of stall from the numerical simulations. From the C_p plots and streamlines, at AOA 4°,

the flow across the wings are generally cleanly attached to the wing, except near the junction between the wing root and the fuselage. At AOA 8° , some span-wise flow indicating separation can be seen developing at approximately 0.7 of the semi-span from the streamline plots. Based on the C_p plots at $3/4$ of the semi-span, a relatively flat region, which is a sign of flow separation, develops from 0.6 of the normalized chord. This is typical for swept back wings, where separation first starts to occur from the wing tips. At AOA 10° , the flow separation regions from the wingtips have developed up to $1/2$ the semi-span. From the C_p plots, a drop in the peak value as compared to AOA 8° at $1/2$ the semi-span can be seen. The kink in the C_p for the first 0.2 of the normalized chord suggests leading edge separation. At $1/4$ the semi-span, the flow is still relatively clean and attached, while the flow is totally separated from the wing at $3/4$ the semi-span. At AOAs 16° and 20° , the flow is cleanly separated from the entire wing. Sharp drops in the peak C_p values at $1/4$ and $1/2$ the semi-span indicate the flow separation, causing a loss in lift generation. The C_p plots for AOAs 16° and 20° are rather similar in terms of trends and magnitudes. Hence, it can be deduced that the increase in the C_L after the local minima is in fact due to the fuselage and its increased upwards force acting on the entire model due to the increase in the AOA.

Fig. 5-4 C_p plots at (a) 1/4 semi-span of wing, (b) 1/2 semi-span of wing, and (c) 3/4 semi-span of wing.

Fig. 5-5 Plot of streamlines and C_p across the wing and fuselage at (a) 4° AOA, (b) 8° AOA, (c) 10° AOA, (d) 16° AOA and (e) 20° AOA.

5.2.3 Variations in AOA: Surface Oil Flow Visualization Results

The surface oil flow visualization serves to better understand the flow characteristics across the range of AOA. The angles chosen were based on the stall region found through the wind tunnel tests, with additional points before and after for comparison at AOAs 0°, 4°, 8°, 10°, 16° and 20°.

Fig. 5-6 Surface oil flow visualizations for (a) 0° AOA, (b) 4° AOA, (c) 8° AOA, (d) 10° AOA, (e) 16° AOA and (f) 20° AOA.

At AOA 0° , a laminar separation bubble is seen forming on the wings, similar to the PP250 configuration. At AOA 4° , there is a region near the wing root and fuselage junction where the density of the oil is different from the rest of the wing. This is the recirculation zone, caused by the adverse pressure gradient in the region ahead of the wing on the fuselage. Slight signs of flow separation through span-wise flows can be seen on the wing tips. As the AOA is increased to 8° , the flow recirculation zone near the wing root and the flow separation at the wing tips become more prominent and

obvious. At 10° , the flow separation regions from the wing tips have developed up to almost $1/2$ the semi-span of the wing. For 16° and 20° , the surface oil patterns are similar to each other. The flow is no longer attached along the wings, and the patterns are caused by the flow recirculation region above the wings. Eder et al. (2006) also reported similar findings on the progression of flow separation on the wing up to stall through the use of tufts on their wind tunnel model.

When comparing the surface oil flow visualizations patterns with the surface streamlines based on the wall shear obtained from the numerical simulations, some similarities and differences can be observed. For the fuselage, at the larger AOA's, a unique bell-shaped separation can be seen in the surface oil flow visualizations near the front. However, this separation is not predicted by the numerical simulations. It has been noted that this pattern seem to originate from approximately the same location. This is probably due to the difficulty of machining out a perfectly smooth and continuous spline for the profile of the fuselage. Hence the surface irregularity acts as a local obstruction and separates the flow. At AOA 0° , the numerical simulation fails to predict the laminar separation bubble due to the nature of the turbulence model used. At AOA 4° , the size of the recirculation zone is under predicted by the numerical simulations. The patterns for AOA's 8° and 10° are a close match and greatly aids in the understanding of the flow topology and the interpreting of the surface oil flow visualizations. However, at the larger AOA's of 16° and 20° , the wall shear streamlines seem to be unable to display the recirculation.

Fig. 5-7 Surface streamlines based on wall shear for (a) 0° AOA, (b) 4° AOA, (c) 8° AOA, (d) 10° AOA, (e) 16° AOA and (f) 20° AOA.

5.3 Variations in AOA and Sideslip Angle

For the wind tunnel experiments, the AOA and sideslip angle were varied from 0° to 20° and 5° to 30° respectively at 5° intervals for a combined 5×6 test matrix. The numerical simulations and surface oil flow visualizations were run based on the same conditions as the wind tunnel experiments, but only for two sideslip angles of 15° and 30° .

5.3.1 Variations in AOA and Sideslip Angle: Wind Tunnel Results

Figure 5-8 displays the wind tunnel results at $Re=116,250$, for the various AOAs, as the sideslip angle β changes. It is found that the C_L remains relatively constant at the given AOA as the sideslip angle β increases, and only drops when β reaches 30° . Notice that the C_L values are almost identical for AOAs 10° , 15° and 20° for small values of β . This is also in line with the case of no sideslip, where the C_L values are at approximately the same levels for those particular AOAs. This demonstrates the lift independence of this configuration on the sideslip angle β , making it stable to sudden changes in the lateral direction of the oncoming free-stream. The resulting flow separation behind the fuselage region has little impact on the lift generation from the wings. The C_D increases as the sideslip angle β increases, as a larger proportion of the fuselage is opposing the oncoming free stream, resulting in a more bluff body flow situation.

For directional stability, a positive $C_{N\beta}$ is required. The $C_{N\beta}$ for the D150 is generally positive, indicating directional stability. However, at AOAs 20° and 25° , the $C_{N\beta}$ is close to zero and even negative on one occasion for small values of β up to 10° . This is still satisfactory as the aircraft is not expected to operate at these range of angles for a prolonged period of time.

For lateral stability, a negative $C_{l\beta}$ is required. The $C_{l\beta}$ for the D150 is generally negative, indicating directional stability. However, the gradient becomes positive between β of 25° and 30° for both AOAs 15° and 20° . In addition, the slope for AOA 15° is more gentle as compared to the other AOAs. This is probably due to the fact that the stall has affected the lift generation most severely at that region of AOA as seen in

Fig. 5-8 Plot of (a) C_L and (b) C_D for D150 for varying β .

Fig. 5-9 Plot of (a) yawing moment coefficient, C_N and (b) rolling moment coefficient, C_l , against sideslip angle, β .

the previous section. This is still satisfactory as the aircraft is not expected to operate at these range of angles for a prolonged period of time. As anticipated, there should not be much problems with the stability of a well established design that is still in operation currently.

5.3.2 Variations in AOA and Sideslip Angle: Numerical Simulation

Results

For the comparison between the WT results and the CFD results where the AOA and sideslip angle were varied, the trends for the C_L , C_D , and C_Y are similar for the sideslip angles investigated at 15° and 30°. The C_L slope has two distinct gradients, and

Fig. 5-10 Comparison of coefficients between WT and CFD results for (a) $\beta=15^\circ$ and (b) $\beta=30^\circ$.

the change occurs around AOA 10° . The gradient decreases after AOA 10° . The C_D shows a similar parabolic trend, while the C_Y shows a slightly increasing trend. Beyond the sideslip angle of 15° , there is a large discrepancy especially in the C_L and C_D . Figure 5-10 shows the comparison between the WT results and CFD results for the sideslip angles of 15° and 30° . One of the reasons for the large discrepancy found, as mentioned in the previous chapter, is probably due to the increased blockage ratio in the test section as the sideslip angle is increased.

The flow fields were visualized for both sideslip angles of 15° and 30° based on the results of the numerical simulations. AOAs of 0° , 10° , 15° and 20° were chosen as based on the wind tunnel and numerical simulation results, the gradient of the lift slope experiences a reduction after AOA of 10° . For the case with the sideslip angle at 15° , shown in figure 5-11, at AOA 0° , the flow across both the front and rear wings are rather clean with no span-wise flow them. At AOA 10° , some span-wise flow starts to develop across the leeward left wing, which is angled further behind to the incoming free stream as compared to the windward right wing, starting from approximately 40% of its semi-span. On the windward right wing, flow separation with recirculation

Fig. 5-11 Flow fields generated for $\beta=15^\circ$, (a) AOA 0° , (b) AOA 10° , (c) AOA 15° and (d) AOA 20° .

develops at approximately 40% of its semi-span to nearly the wing tip. At AOAs 15° and 20° , flow separation with recirculation spanning the entire wings is observed. The main difference is that at AOA 15° , the flow accelerates from under the fuselage and impinges on the leading edge of the leeward left wing near the wing root and forms a vortex which subsequently connects with the span-wise flow. At AOA 20° , this leading edge vortex is not formed. In addition, the wake from the windward right wing impinges directly on the vertical tail fin.

For the case with the sideslip angle at 30° , shown in figure 5-12, at AOA 0° , the flow accelerates from under the fuselage and forms a roll up vortex which flows over junction between the leeward left wing root section and the fuselage. This roll up vortex is present in all the range of AOAs tested. The flow over the windward right wing is clean and attached. At AOA 10° , some span-wise flow starts to develop across the leeward left wing from approximately 20% of its semi-span at the leading edge. On the windward right wing, flow separation with recirculation develops at approximately 40% of its semi-span to nearly the wing tip. At AOA 15° , an angled vortex, formed by the flow accelerating from under the fuselage can be seen forming on the from the connection point of the fuselage and the leeward left wing leading edge, developing into span-wise flow across the entire wing. The windward right wing experiences flow separation with recirculation across its entire semi-span. At AOA 20° , a vortex similarly forms on the leading edge of the leeward left wing. However, the nature of this vortex is slightly different from that at AOA 15° , as seen from the speed of the streamlines as well as the C_p distribution. In this case, the vortex has a "tornado-like" structure, forming on the upper surface of the wing, before developing into a span-wise flow.

Fig. 5-12 Flow fields generated for $\beta=30^\circ$, (a) AOA 0° , (b) AOA 10° , (c) AOA 15° and (d) AOA 20° .

5.3.3 Variations in AOA and Sideslip Angle: Surface Oil Flow Visualization Results

The surface oil flow visualization serves to provide complementary data on the flow on top of the force balance readings from the wind tunnel experiment. Figure 5-13 shows the case of 15° sideslip angle while figure 5-14 shows the case of 30° sideslip angle for the range of AOAs tested. For the flow separation line on the fuselage, as the AOA increases, the flow separation gradually edges towards to the right side of the fuselage. This is due to the change in the stagnation point on the fuselage due to the change in the AOA. In addition, for both sideslip angles of 15° and 30° , more complex surface flow patterns can be seen forming at AOA 20° . This is most likely due to secondary flow separation and reattachment.

For the case of 15° sideslip angle, at AOA 0° , the flow patterns on the wings are practically the same to that of one with no sideslip angle. At AOA 5° , the recirculation zone is only observed on the leeward left wing root but not on the windward right wing root. At AOA 10° , on the windward right wing, at approximately half of the semi-span, a pool of oil is found to accumulate near the leading edge, suggesting region of separation with re-circulation. On the leeward left wing root, the recirculation zone has extended upstream to reach near the leading edge. Span-wise flow due to flow separation from the wing tips up to approximately half of the semi-span of the leeward left wing was also detected. At AOA 15° , the flow patterns on the windward right wing are similar to case with no sideslip angle, implying that the flow is no longer attached along the wing, and forms a recirculation region above it. On the leeward left wing root, the formation of a vortex can be seen originating from the leading edge up to more than half of the local chord. Span-wise flow then develops from there and extends across the

leeward left wing to the wing tip. At AOA 20° , the flow patterns on both wings are similar to case with no sideslip angle.

For the case of 30° sideslip angle, at AOA 0° , the laminar separation bubble can still be seen on the windward right wing. However, on the leeward left wing, the bubble is disrupted near the wing root. This is probably due to the flow accelerating from under the fuselage, thus forming a roll up vortex which impacts on the wing-fuselage junction which disrupts the bubble. At AOA 5° , the effect of this flow is more evident, with a clear line on the leeward left wing demarcating the region where this roll up vortex impinges on the wing. As with the case of sideslip angle 15° , the recirculation zone on the windward right wing root is not observed. At AOA 10° , on the windward right wing, at approximately 0.25 of the semi-span, a pool of oil is found to accumulate near the leading edge, suggesting region of separation with re-circulation. On the leeward left wing, on top of the roll up vortex impingement, span-wise flow is also detected across the wing. At AOA 15° , the flow features are similar to the previous case of AOA 10° , minus the accumulation of oil on the leading edge of the windward right wing. At AOA 20° , the effect of the roll up vortex is not evident. However, span-wise flow can be observed on both wings.

When comparing the surface oil flow visualizations patterns with the plotted surface streamlines based on the wall shear obtained from the numerical simulations, some similarities and differences can be observed. The flow separation patterns on the fuselage look similar to the wall shear patterns obtained from the numerical simulations, especially at the higher AOAs of 20° for both sideslip angles, with the more complex patterns due to the secondary flow separation and reattachment. Although the exact

positioning may differ, this is mainly due to the turbulence modelling used. Other similarities include the recirculation zones near the leeward left wing root trailing edge at AOAs 0° and 5° , the vortex formation on the leeward left wing at AOA 15° , for sideslip angle of 15° as well as the roll up vortex for the cases at sideslip angle of 30° . Figure 5-15 illustrates a clearer picture of the roll up vortex with the streamlines plotted. Some of the differences would be the location of the centre of the vortex at AOA 10° for both sideslip angles. Numerical simulations predict it forming in the centre of the wing, while the surface oil flow visualizations show it forming closer to the leading edge of the wing. Still, with these discrepancies, the numerical simulations greatly aid with the understanding of the flow structures.

Fig. 5-13 Comparison of coefficients between surface oil flow visualizations and CFD results for $\beta = 15^\circ$, (a) 0° AOA, (b) 5° AOA, (c) 10° AOA, (d) 15° AOA, (e) 20° AOA.

Fig. 5-14 Comparison of coefficients between surface oil flow visualizations and CFD results for $\beta=30^\circ$, (a) 0° AOA, (b) 5° AOA, (c) 10° AOA, (d) 15° AOA, (e) 20° AOA.

Fig. 5-15 Streamlines for roll up vortex for $\beta=30^\circ$.

5.4 Variations in AOA and Roll Angle

For the wind tunnel experiments, the AOA and roll angle were varied from 0° to 20° and -5° to -30° respectively at 5° intervals, forming a 5×6 test matrix. The numerical simulations and surface oil flow visualizations were run based on the same conditions as the wind tunnel experiments, but only for two roll angles of -15° and -30° .

5.4.1 Variations in AOA and Roll Angle: Wind Tunnel Results

Figure 5-16 displays the wind tunnel results at $Re=116,250$, for the various AOAs, as the roll angle ϕ changes. It is found that both C_L and C_D show a gradually decreasing trend at the given AOA as the magnitude of the roll angle ϕ increases. The gentle decrease in the C_L makes this configuration less susceptible to stall due to sudden changes in ϕ . It is also interesting to note that the C_L values for AOAs 10° , 15° and 20° are very similar up to $\phi=-25^\circ$. The relatively constant C_D also suggests that the degree of flow separation remains relatively the same.

Fig. 5-16 Plot of (a) C_L and (b) C_D for D150 for varying ϕ .

5.4.2 Variations in AOA and Roll Angle: Numerical Simulation

Results

For the comparison between the WT and CFD results where the AOA and roll angle were varied, the trends and magnitudes for the coefficients are similar, even at the largest roll angle of -30° at the lower AOAs ($\leq 10^\circ$), as shown in figure 5-17. For the case of $\phi = -15^\circ$, as the AOA is increased beyond 10° , the numerical results show a dip at AOA 15° before recovering at AOA 20° , similar to case when only the AOA is varied. For the case of $\phi = -30^\circ$, as the AOA is increased beyond 10° , the C_L remains relatively constant at AOA 15° before decreasing at AOA 20° . However, the WT results show that C_L remain relatively constant between AOAs 10° and 20° for both roll angles. The C_D shows a similar parabolic trend. The C_Y gradually increases from AOA 0° to AOA 10° , then remains relatively constant thereafter.

Based on the numerical simulations, the flow fields were visualized for both roll angles of -15° and -30° . AOAs of 0° , 10° , 15° and 20° were chosen as based on the wind tunnel and numerical simulation results, the AOA of 10° is the critical point where the gradient of the lift slope changes. It is found that the flow fields generally develop

Fig. 5-17 Comparison of coefficients between WT and CFD results for (a) $\phi = -15^\circ$ and (b) $\phi = -30^\circ$.

in a similar fashion for both roll angles of 15° and -30° , shown in figures 5-18 and 5-19 respectively. At AOA 0° , the flow across both the front and rear wings are rather clean and attached to the surface of the wings. At AOA 10° , flow separation is observed originating from the wing tips, traversing almost half the semi-span of the wings. At AOAs 15° and 20° , the flow separation occurs across the entire span of the wings. For $\phi = -15^\circ$, this flow separation causes the C_L to dip from AOA 10° to AOA 15° . The C_P distribution on the wings at AOAs 15° and 20° at $\phi = -15^\circ$ are similar in terms of magnitude. Hence, the recovery of the C_L at AOA 20° is most likely due to the fuselage and the increased upwards force due to the increase in the AOA. For $\phi = -30^\circ$, although flow separation also occurs across the entire wing, at AOA 15° , a vortex is observed forming on the leading edge of the left wing at approximately 0.2 semi-span from the wing root, as seen in figure 5-20. This vortex changes the C_P distribution significantly, thus increasing the lift generation. Hence, the dip that was experienced at $\phi = -15^\circ$ for AOA 15° is not observed at $\phi = -30^\circ$ for AOA 15° .

Fig. 5-18 Flow fields generated for $\phi = -15^\circ$, (a) AOA 0° , (b) AOA 10° , (c) AOA 15° and (d) AOA 20° .

Fig. 5-19 Flow fields generated for $\phi = -30^\circ$, (a) AOA 0° , (b) AOA 10° , (c) AOA 15° and (d) AOA 20° .

Fig. 5-20 Streamlines of vortex forming on left wing at $\phi=-30^\circ$, AOA 15° .

5.4.3 Variations in AOA and Roll Angle: Surface Oil Flow Visualization Results

Figure 5-21 shows the case of -15° roll angle while figure 5-22 shows the case of -30° roll angle for the range of AOAs tested. For both roll angles, the surface oil flow patterns on the wings resemble that of the case without any roll for the lower AOAs. For both roll angles, at AOA 0° , laminar separation bubbles are formed on both wings. A small recirculation zone is also detected at the wing-fuselage junction near the trailing edge. At AOA 5° , the laminar separation bubble is no longer present, and the recirculation zone expands outwards slightly. At AOA 10° , flow separation from the wing tips develops inboard and meet with the recirculation at the wing-fuselage junction. At this AOA, in comparison, the flow separation line at $\phi=-30^\circ$ is longer than that at $\phi=-15^\circ$. At AOA 15° , vortices are seen forming at the leading edge of both the wings, at the wing-fuselage junction. Span-wise flow is then observed to originate from the edge of the vortices all the way to the wing tips. At AOA 20° , for $\phi=-15^\circ$, the oil flow patterns on each wing are symmetric and resemble the case with no roll angle, suggesting a flow recirculation region above the wings. However, for the $\phi=-30^\circ$ case, the flow patterns are asymmetric. On the right wing, the patterns are similar as the case

with no roll angle, with the span-wise flow originating from the trailing edge of the wing root. On the left wing, the span-wise flow originates from the leading edge instead.

When comparing the surface oil flow visualizations patterns with the surface streamlines based on the wall shear obtained from the numerical simulations, some similarities and differences can be observed. For the fuselage, at the larger AOAs, flow separation can be seen in the surface oil flow visualizations originating from near the front. It is noted that this point of origin did not change with the AOA. However, this separation is not predicted by the numerical simulations. This is probably due to the difficulty of machining out a perfectly smooth and continuous spline for the profile of the fuselage. The surface irregularity might have been the cause of flow separation. For the wings, at AOA 0° , the numerical simulations do not predict the laminar separation bubble due to the choice of turbulence model. At AOA 5° , the numerical simulations seem to under predict the size of the recirculation zone. The wall shear patterns for AOA 10° match quite well with the surface oil flow visualizations at both roll angles. However, at AOA 15° , the numerical simulations only predict a vortex forming on the leading edge of the left wing for $\phi = -30^\circ$, and thus causing the discrepancy between the wind tunnel experimental results.

Fig. 5-21 Comparison of coefficients between surface oil flow visualizations and CFD results for $\phi = -15^\circ$, (a) 0° AOA, (b) 5° AOA, (c) 10° AOA, (d) 15° AOA, (e) 20° AOA.

Fig. 5-22 Comparison of coefficients between surface oil flow visualizations and CFD results for $\phi = -30^\circ$, (a) 0° AOA, (b) 5° AOA, (c) 10° AOA, (d) 15° AOA, (e) 20° AOA.

Chapter 6 Aerodynamic Characterization of B150

6.1 Overview

In this chapter, the results for the B150 model, a variant of the D150 reference model, are presented and discussed. The results are arranged according to the following cases considered: variations in AOA, variations in AOA and sideslip angle as well as variations in AOA and roll angle. The results obtained were from wind tunnel experiments, numerical simulations and surface oil flow visualizations, and are compared side by side for a more complete analysis.

6.2 Variations in AOA

For this case, the AOA was varied from -10° to 25° . The free-stream velocity for the wind tunnel experiments was set at 30m/s, with a resultant Reynolds number of 77,500 based on the test conditions. The numerical simulations were run based on the same conditions matching that of the wind tunnel experiments for consistent comparison and validation. The surface oil flow visualizations were conducted at selected angles before, during and after the stall regions found to better understand the aerodynamic characteristics.

6.2.1 Variations in AOA: Wind Tunnel Results

For the wind tunnel results at $Re=77,500$, displayed in figure 6-1, a linearly increasing trend of the lift coefficient, C_L , with two distinct slopes was observed. There are no significant drops in the C_L which indicates stall. However, the C_L values are found to

Fig. 6-1 Comparison of (a) C_L , (b) C_D and (c) L/D for B150 and D150 for varying AOA.

start to taper off at AOA 4°, stagnate between AOAs of 8° to 10° before starting to increase again. However, the lift curve slope after AOA 10° is more shallow as compared to that before AOA 4°. The drag polar exhibits a typical parabolic profile.

In comparison with the reference D150, although the B150 does not experience a drop in the C_L after AOA 14°, it experiences the stagnation in the C_L much earlier, and the overall magnitude is generally lower than the reference model. The drag of the B150 is also generally higher than the reference D150 across the AOAs tested. However, the difference is only slight, and the D150 model is also missing its horizontal tail, which would add a penalty to the total drag. In addition, due to the reduced C_L values of the B150, the maximum lift to drag ratio of the B150 (≈ 11) is consequently affected and pales in comparison to that of the D150 (≈ 17). Incidentally, a lift to drag ratio of 17 is

Fig. 6-2 Plot of pitching moment coefficient, C_M , against effective angle of attack, α_{eff} for B150 and D150.

the ballpark figure for a commercial airliner at cruise, which gives extra confidence to the wind tunnel results. Hence, based on this wind tunnel experiment for this given configuration, there is no visible benefit of the reduction in the induced drag of the joined-wing configuration over the conventional design. Even though optimization done at Mach 0.78 with the Tornado vortex lattice code shows a drag reduction for the B150 over the conventional design, the Prandtl-Glauert corrections might not have been able to accurately model the flow. In addition, the vortex lattice code ignores the effect of the fuselage, which may have caused extra drag penalties due to the interaction between the fuselage and the wings. Furthermore, the code only calculates the induced drag and ignores the skin friction drag.

For the B150 configuration, the pitching moment slope, $C_{M\alpha}$, is found to be generally negative, and the pitching moment, C_M , about the C.G. at 0° AOA is positive, as seen in figure 6-2. Based on the wind tunnel lift results, the zero-lift angle is approximately $\alpha_{0L} = -1.3^\circ$ and that the current configuration is largely statically stable in the longitudinal direction. However, the pitching moment slope becomes 0 between $\alpha_{eff} 5^\circ$

Fig. 6-3 Comparison of (a) coefficient of lift, C_L , and (b) coefficient of drag, C_D , against AOA between experimental and numerical simulation results.

and 7° , indicating longitudinal neutrality, and even positive between α_{eff} 8° and 11° , indicating longitudinal instability. This region roughly coincides at the region where the C_L starts to taper off and stagnate before increasing again. In comparison with the D150 configuration, the $C_{M\alpha}$ for the B150 is steeper than that of the D150, indicating stronger resistance to change in AOA, due to the rear wings of the B150 imparting a restoring moment. However, the B150 has a much smaller stable operating region before instability sets in as compared to the D150.

6.2.2 Variations in AOA: Numerical Simulation Results

When comparing the numerical simulation results with the experimental results, the overall trends and magnitudes are matching, as seen in figure 6-3. The trends of the drag coefficient, C_D , obtained through numerical simulations are very favourable and matching above AOA 6° . Below AOA of 6° , the numerical simulations results slightly over predicts the C_D as compared to the wind tunnel results. This is most probably due to the fully turbulent nature of the numerical simulations. For the C_L , the numerical simulations manages to identify the region where the C_L tapers off, stagnate, then

increase again. However, the value of C_L is over predicted by the numerical simulations at AOAs below 0° .

Based on the numerical simulations, the stall mechanism was investigated and explored through the C_P plots and streamline plots. The C_P was taken across the upper surface of the wings at 1/4, 1/2 and 3/4 of the semi-span were plotted for AOAs of 4° , 8° , 10° , 16° and 20° . At AOA 4° , the flow across the wings are generally clean and remained attached.

At AOA 8° , on the front wing, flow separation develops from the wing tips up to approximately 1/2 of the semi-span. From the C_P plot at 1/2 of the semi-span of the front wing, there is a drop in the peak C_P value from AOA 4° and a flat portion occurring at 0.2 of the normalized chord. From the C_P plot at 3/4 of the semi-span of the front wing, there is a sharp drop in the peak C_P value, and the flat portion occurs at approximately 0.4 of the normalized chord and stretches to the end of the chord. On the rear wing, a flow recirculation zone develops at the wing-fuselage junction. Some span-wise flow is observed originating from the wing tip, then spreading inboard to approximately 1/2 of the semi-span. This is also captured on the C_P plot at 1/2 the semi-span of the rear wing, with a long flat portion starting at approximately 0.6 of the normalized chord.

At AOA 10° , on the front wing, the flow separation develops even further inboard as compared to AOA 8° , as seen from the streamlines and C_P distribution. There is a decrease in the peak C_P value at 1/2 the semi-span as compared to that at AOA 8° , but the C_P distribution at 3/4 the semi-span is similar to that AOA 8° , indicating similar

Fig. 6-4 C_p plots at (a) 1/4 semi-span of front wing, (b) 1/4 semi-span of rear wing, (c) 1/2 semi-span of front wing, (d) 1/2 semi-span of rear wing, (e) 3/4 semi-span of front wing, (f) 3/4 semi-span of rear wing.

extent of flow separation. At AOA 10° , on the rear wing, the flow recirculation zone is still present at the wing-fuselage junction. However, the span-wise originating from the wing tip has also developed further inboard, all the way to the wing root. This is evident in the C_p plots, with the flat portions at both 1/4 and 1/2 of the semi-span starting from approximately 0.4 of the normalized chord.

Fig. 6-5 Plot of streamlines and C_p across the front wing (left) and rear wing (right) at (a) 4° AOA, (b) 8° AOA, (c) 10° AOA, (d) 16° AOA and (e) 20° AOA.

At AOA 16° , on the front wing, the flow separation develops even further inboard, as compared to AOA 10° , up to the wing root, as seen from the streamlines and C_p distribution. There is a sharp drop in the peak C_p value at $1/4$ the semi-span from that at AOA 10° , and the C_p distribution almost resembles a flat line. The same "flat" C_p distribution is also observed at $1/2$ and $3/4$ the semi-span of the front wing, indicating flow separation across the entire front wing. At AOA 16° , on the rear wing, the flow recirculation zone is still present at the wing-fuselage junction. However, wing tip vortices driving the span-wise flow increases, as seen from the intensification of the C_p and streamlines. This causes the C_p magnitudes at $3/4$ and $1/2$ the semi-span of the rear wing to increase as compared to AOA 10° from 0.4 of the normalized chord onwards. This causes the lift generation on the rear wing to increase, contributing to the increase in the total C_L even though the entire front wing has stalled at AOA 16° .

At AOA 20° , on the front wing, the situation is similar to that at AOA 16° , with minimal differences in terms of C_p distributions and magnitude. On the rear wing, the wing tip vortices further intensify, as seen in the increase in the C_p magnitudes at $3/4$ of the semi-span of the rear wing. However, this impact only affects the outboard half of the rear wing, as the C_p distributions at $1/4$ and $1/2$ of the semi-span of the rear wing remains similar in terms of magnitudes. It can be deduced that the further increase in the total C_L from AOA 16° to 20° is due to the fuselage and its increased upwards force acting on the entire model due to the increase in the AOA.

6.2.3 Variations in AOA: Surface Oil Flow Visualization Results

The surface oil flow visualization serves to better understand the flow characteristics across the range of AOA. The angles chosen were based on the stall region found through the wind tunnel tests, with additional points before and after for comparison at AOAs 0° , 4° , 8° , 10° , 16° and 20° .

At AOA 0° , a faint line could be seen on the front wing. This should be the point where the laminar flow turned turbulent. In comparison with the D150, no laminar separation bubble is observed this time. No discernible features are also seen on the rear wing. At AOA 4° , flow separation is observed to have occurred on the outboard section of the front wing near the wing tips. The smudge is caused by the dripping of the accumulation of oil, due to separation. On the rear wing, the faint line near the leading edge should be the flow transition point. At AOA 8° , the flow separation at the wing tips have progressed inboard to approximately $1/2$ of the semi-span. The span-wise flow is a result of this separation. In addition, vortices are formed near the wing root of the front wing. On the rear, flow separation also occurs on the wing tips. At AOA 10° , the flow features are similar to that at AOA 8° . The vortices on the front wings have moved slightly inboard towards the fairing. In addition, the flow recirculation at the rear wing-fuselage junction could be detected with an accumulation of oil near the rear wing root. At AOA 16° , the vortices on the front wing have moved slightly inboard again onto the fairing. The origin of the span-wise flow on the front wing is further inboard, close to the leading edge of the front wing root. For the rear wing, span-wise flow could be seen originating from the wing tip towards the wing root, slightly in front of the trailing edge. At AOA 20° , the vortices on the fairing are displaced downstream and are located near the middle of the normalized chord, instead of being near the leading edge at AOA

16°. The origin of the span-wise flow on the front wing is also displaced downstream, coinciding with the vortices. For the fuselage, at the larger AOA, a unique bell-shaped separation can be seen in the surface oil flow visualizations near the front, and the location remains constant regardless of the AOA.

When comparing the surface oil flow visualizations patterns with the surface streamlines based on the wall shear obtained from the numerical simulations, some similarities and differences could be observed. Flow separation is not predicted to occur on the fuselage at any AOA. This suggested that the flow separation observed in the surface oil flow experiments is possibly a consequence of surface irregularities due to machining. At AOA 0°, no discernible features are observed from the numerical simulations. At AOA 4°, flow separation on the front wing tip is correctly predicted by the numerical simulations. As for the flow separation predicted to occur on the rear wing near the trailing edge, a very faint line can be seen near the trailing edge of the rear wing for the surface oil flow visualizations, which could correspond to the numerical simulations. At AOA 8° and 10°, vortices on the front wing, including the shift inboard due to change in the AOA, and the flow recirculation on the rear wing are accurately captured by the numerical simulations. The location of the vortices on the front wing is also accurately predicted at AOA 16°. However, at AOA 20°, the numerical simulations fails to predict the shift downstream of the vortices.

Fig. 6-6 Surface oil flow visualizations for (a) 0° AOA, (b) 4° AOA, (c) 8° AOA, (d) 10° AOA, (e) 16° AOA and (f) 20° AOA.

Fig. 6-7 Surface streamlines based on wall shear for (a) 0° AOA, (b) 4° AOA, (c) 8° AOA, (d) 10° AOA, (e) 16° AOA and (f) 20° AOA.

6.3 Variations in AOA and Sideslip Angle

For the wind tunnel experiments, the AOA and sideslip angle were varied from 0° to 20° and 5° to 30° respectively at 5° intervals for a combined 5×6 test matrix. The numerical simulations and surface oil flow visualizations were run based on the same conditions as the wind tunnel experiments, but only for two sideslip angles of 15° and 30°, similar to the D150 configuration.

6.3.1 Variations in AOA and Sideslip Angle: Wind Tunnel Results

Figure 6-8 displays the wind tunnel results at $Re=77,500$, for the various AOA, as the sideslip angle β changes. The C_L shows a gradual decreasing trend as β increases. In comparison with the D150 configuration, the C_L magnitudes are lower in the B150 configuration. The C_D values are comparable.

For the directional stability, the $C_{N\beta}$ for the B150 is generally positive, even at AOA 25° , which is not the case for the D150. However, for the B150, at AOA 10° and 15° , the $C_{N\beta}$ becomes negative when β becomes more than 25° . This is still satisfactory as the aircraft is not expected to operate at these range of angles for a prolonged period of time.

For lateral stability, a negative $C_{l\beta}$ is required. In this aspect, the B150 shows lateral instability past sideslip angle of 5° , with the $C_{l\beta}$ turning positive for most of the AOA tested. This gives rise to poor flight handling qualities, easily leading to spiral instability. In comparison, the D150 is laterally stable on most occasions.

6.3.2 Variations in AOA and Sideslip Angle: Numerical Simulation Results

For the comparison between the WT results and the CFD results where the AOA and sideslip angle were varied, the trends for the C_L , C_D , and C_Y are similar for the sideslip angles investigated at 15° and 30° . The C_L slope has two distinct gradients, and the change occurs around AOA 5° . The gradient decreases after AOA 5° . The C_D shows a similar parabolic trend, while the C_Y remains relatively constant. For both sideslip

Fig. 6-8 Plot of (a) C_L and (b) C_D for D150 for varying β . Fig. 6-9 Plot of (a) yawing moment coefficient, C_N and (b) rolling moment coefficient, C_l , against sideslip angle, β .

angles, the CFD systematically under predicts the C_L and C_D from AOAs 10° to 20° , and over predicts the C_Y systematically for the entire range of AOA tested.

The flow fields were visualized for both sideslip angles of 15° and 30° based on the results of the numerical simulations. For clarity, the flow fields were generated over the front wings and the rear wings separately. AOAs of 0° , 5° and 15° were chosen as based on the wind tunnel and numerical simulation results, the gradient of the lift slope experiences a reduction after AOA of 5° . For the case with the sideslip angle at 15° , shown in figure 6-11, at AOA 0° , the flow across both the front and rear wings are rather clean with no span-wise flow over them. However, a small region of flow recirculation is observed at the root section of the leeward rear left wing, near the

Fig. 6-10 Comparison of coefficients between WT and CFD results for (a) $\beta=15^\circ$ and (b) $\beta=30^\circ$.

trailing edge. At AOA 5° , flow separation occurs on the wing tips of both the front wings, resulting in some span-wise flow. Likewise, on the windward rear right wing, some span-wise flow is detected from the wing tip. In addition, the flow recirculation at the root section of the leeward rear left wing grows in size. At AOA 15° , the flow is almost totally detached from the windward front right wing. A vortex is also formed near the centre of the front fairing. On the leeward front left wing, a vortex is formed on the leading edge at approximately 0.2 of the semi-span, accelerating the flow and causing span-wise flow over the outboard section. On the windward rear right wing, span-wise flow traverses the entire span of the wing. On the leeward rear left wing, the flow is also totally detached, forming a region of recirculation.

For the case with the sideslip angle at 30° , shown in figure 6-12, at AOA 0° , the flow across the front wings are mostly clean, with a small vortex forming on the leading edge near the wing-fairing junction. However, for the leeward rear left wing, a vortex roll up from the front of the fuselage impinges and accelerates the flow over the wing-fuselage junction. In addition, a region of flow recirculation is formed next to the point of impact of the roll up vortex. At AOA 5° , on the front wings, the size of the vortex grows, and flow separation starts to develop from the wing tips, resulting in some span-wise flow.

The flow on the rear wings is similar to that at AOA 0° , with the exception of span-wise flow developing over the wing tips. At AOA 15° , the flow is almost totally detached from the windward front right wing. On the leeward front left wing, a vortex is formed on the leading edge at approximately 0.2 of the semi-span, accelerating the flow and causing span-wise flow over the outboard section. The strength of this vortex is much stronger than of sideslip angle 15° at the same AOA, as shown by the difference in the intensity of the C_p distribution. In addition, another roll up vortex from the front of the fuselage impinges and accelerates the flow over the wing-fairing junction. On the windward rear right wing, span-wise flow traverses the entire span of the wing. On the leeward rear left wing, the flow is also totally detached, forming a region of recirculation.

Fig. 6-11 Flow fields generated for $\beta=15^\circ$, (a) AOA 0° , (b) AOA 5° and (c) AOA 15° .

Fig. 6-12 Flow fields generated for $\beta=30^\circ$, (a) AOA 0° , (b) AOA 5° and (c) AOA 15° .

6.3.3 Variations in AOA and Sideslip Angle: Surface Oil Flow

Visualization Results

The surface oil flow visualization serves to provide complementary data on the flow on top of the force balance readings from the wind tunnel experiment. Figure 6-13 shows the case of 15° sideslip angle while figure 6-14 shows the case of 30° sideslip angle for

the range of AOAs tested. For the case of 15° sideslip angle, at AOA 0° , on the leeward rear left wing, a small accumulation of oil can be seen near the wing-fuselage junction on the trailing edge. This suggests some flow recirculation. As the AOA is increased to 5° , flow separation starts to occur from the wing tips of the front wings. For the rear wings, span-wise develops on the windward rear right wing, while the recirculation region on the leeward rear left wing expands and grows in size. At AOA 10° , the flow separation on the front wings develop further inboard, and a vortex can be seen forming near the wing-fairing junction, on the leading edge of the windward front right wing. The other features are similar to that AOA 5° . At 15° , the vortex on the windward front right wing moves slightly inboard and downstream in comparison to AOA 10° onto the fairing. Another vortex can be observed on the leading edge of the leeward front left wing, and span-wise flow develops from there, up to the wing tip. The flow features on the rear wings are still similar to that at AOA 10° . At AOA 20° , the size of the vortex on the windward front right wing moves even further downstream, while the vortex on the leeward front left wing reduces in size.

For the case of 30° sideslip angle, at AOA 0° , on the windward front right wing, a vortex can be seen forming near the wing-fairing junction on the leading edge. Span-wise flow is also detected on the windward front right wing. However, on the leeward rear left wing, there is a small patch near the wing-fuselage junction that seems relatively untouched. This is probably due to the roll up vortex from the front of the fuselage impinging on that region. Next to that region is an accumulation of oil, which suggests some flow recirculation. At AOA 5° , on top of the vortex near the wing-fairing junction on the leading edge on the windward front right wing, another vortex can be seen on the centre of the fairing. The other flow features are similar to that at AOA 5° .

At AOAs 10° and higher, the vortices are still present on the front wings, but the smudging on the oil patterns make it hard to determine the exact location. In addition, the effect of the roll up vortex on the leeward rear left wing is reduced and is shifted forwards to the leeward front left wing, just next to the wing-fairing junction.

When comparing the surface oil flow visualizations patterns with the surface streamlines based on the wall shear obtained from the numerical simulations, some similarities and differences can be observed. The numerical simulations are able to predict the locations of the vortices, as well as the impact of the roll up vortices from the fuselage on the wings. However, the predicted locations of the vortices are usually more downstream as compared to the surface oil flow visualization experiments.

Fig. 6-13 Comparison of coefficients between surface oil flow visualizations and CFD results for $\beta = 15^\circ$, (a) 0° AOA, (b) 5° AOA, (c) 10° AOA, (d) 15° AOA, (e) 20° AOA.

Fig. 6-14 Comparison of coefficients between surface oil flow visualizations and CFD results for $\beta = 30^\circ$, (a) 0° AOA, (b) 5° AOA, (c) 10° AOA, (d) 15° AOA, (e) 20° AOA.

6.4 Variations in AOA and Roll Angle

For the wind tunnel experiments, the AOA and roll angle were varied from 0° to 20° and -5° to -30° respectively at 5° intervals, forming an 5×6 test matrix. The numerical simulations and surface oil flow visualizations were run based on the same conditions as the wind tunnel experiments, but only for two roll angles of -15° and -30° .

6.4.1 Variations in AOA and Roll Angle: Wind Tunnel Results

Figure 6-15 displays the wind tunnel results at $Re=77,500$, for the various AOAs, as the roll angle ϕ changes. It is found that both C_L and C_D show a gradually decreasing trend at the given AOA as the magnitude of the roll angle ϕ increases. The gentle decrease in the C_L , which makes this configuration less susceptible to stall due to sudden changes in ϕ . This behaviour is similar to that of the D150 configuration.

6.4.2 Variations in AOA and roll angle: Numerical Simulation Results

For the comparison between the WT and CFD results where the AOA and roll angle were varied, the trends and magnitudes for the coefficients were similar, even at the largest roll angle of -30° , as shown in figure 6-16. For both the C_L and C_Y , there is decrease in the gradient of the slope after AOA 5° . After AOA 5° , the increase in C_L becomes slower, whereas for the C_Y , it becomes relatively constant.

Based on the numerical simulations, the flow fields were visualized for both roll angles of -15° and -30° . AOAs of 0° , 5° and 15° were chosen as based on the wind tunnel and numerical simulation results, the AOA of 5° is the critical point where the gradient of

Fig. 6-15 Plot of (a) C_L and (b) C_D for D150 for varying ϕ . Fig. 6-16 Comparison of coefficients between WT and CFD results for (a) $\phi = -15^\circ$ and (b) $\phi = -30^\circ$.

the lift slope changes. It is found that the flow fields generally developed in a similar fashion for both roll angles of 15° and -30° , shown in figures 6-17 and 6-18 respectively. At AOA 0° , the flow across both the front and rear wings are clean and attached to the surface of the wings. At AOA 5° , flow separation is observed originating from the wing tips on the front wings, spanning approximately 0.4 of the semi-span. On the rear wings, for $\phi = -15^\circ$, the flow is clean. For $\phi = -30^\circ$, a small recirculation zone is observed at the wing-fuselage junction, near the wing root. At AOA 15° , flow separation occurs across the entire span of the wings. On the front wings, two vortices are seen developing, with one on each wing. The focus of the vortex on the right wing is closer towards the fairing, while the vortex on the left wing is slightly angled and situated near the leading edge of the wing root. Based on the streamlines and C_p

Fig. 6-17 Flow fields generated for $\phi = -15^\circ$, (a) AOA 0° , (b) AOA 5° and (c) AOA 15° .

distribution, the vortex on the left wing is stronger for the case of $\phi = -30^\circ$ as compared to that of $\phi = -15^\circ$. Figure 6-19 shows the two vortices forming for both roll angles at AOA 15° .

Fig. 6-18 Flow fields generated for $\phi = -30^\circ$, (a) AOA 0° , (b) AOA 5° and (c) AOA 15° .

Fig. 6-19 Streamlines of vortices forming for AOA 15° , (a) $\phi = -15^\circ$ and (b) $\phi = -30^\circ$.

6.4.3 Variations in AOA and Roll Angle: Surface Oil Flow Visualization Results

Figure 6-20 shows the case of -15° roll angle while figure 6-21 shows the case of -30° roll angle for the range of AOAs tested. Based on the surface oil flow visualization experiments, the flow is observed to develop in similar fashion for both cases. At AOA 0° , a faint line can be seen on the front wings. This should be the point where the laminar flow turned turbulent, just like the case with no roll angle. At AOA 5° , flow separation starts from the wing tips of the front wings, while some flow recirculation can be seen on the rear wings, near the wing-fuselage junction. At AOA 10° , the flow separation on the front wing develops further inboard, and two vortices, one on each wing, can be seen forming near the wing-fairing junction on the leading edge. The sizes of the vortices are asymmetric, possibly due to the roll angle. Span-wise flow can be seen developing from next to the vortex, up to the wing tip. At AOA 15° , the vortices reduce in size and move inboard, as compared to AOA 10° . This trend is also continued at AOA 20° , as the vortex on the right wing moves onto the fairing.

When comparing the surface oil flow visualizations patterns with the surface streamlines based on the wall shear obtained from the numerical simulations, some similarities and differences can be observed. The numerical simulations are able to predict the locations of the vortices. However, the predicted locations of the vortices are usually more downstream as compared to the surface oil flow visualization experiments.

Fig. 6-20 Comparison of coefficients between surface oil flow visualizations and CFD results for $\phi = -15^\circ$, (a) 0° AOA, (b) 5° AOA, (c) 10° AOA, (d) 15° AOA, (e) 20° AOA.

Fig. 6-21 Comparison of coefficients between surface oil flow visualizations and CFD results for $\phi = -30^\circ$, (a) 0° AOA, (b) 5° AOA, (c) 10° AOA, (d) 15° AOA, (e) 20° AOA.

Chapter 7 Characterization of Wing Deflections¹

7.1 Overview

In this chapter, the impact of utilizing additive manufacturing, to produce the wings of the wind tunnel models, on the experimental and numerical simulation results is briefly looked into. Due to the complex and intricate nature of the wings of the wind tunnel model, additive manufacturing was chosen to take advantage of its flexibility and strengths in producing parts that might be difficult to manufacture through conventional machining methods. Since the 3D-printed wings were made through sintering, the melting of powder to form a solid, it would not be unreasonable to assume isotropic mechanical properties (Proto Labs, n.d.). The elastic modulus and shear modulus of the printed nylon provided by EOS GmbH (n.d.) are 1700MPa and 762 MPa respectively at 20°C. The calculated Poisson's ratio (Roylance, 2008) based on isotropic assumptions is 0.115. As a result, deflections were observed during the experiments. In order to account for these deflections, images of the deflections were captured during the experiments to be quantified. These deflections can be used in the future to validate future fluid-structure interaction (FSI) simulations.

7.2 Wing Deflections of PP250

To look into whether variations in the lift performance caused by wing deflections at the stall point can be systematically corrected for, two AOAs of 8° and 10° were chosen. Figure 7-1 shows the deflections captured for both AOA 8° and AOA 10° from the rear

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¹ Part of this chapter has been accepted for publication entitled "Wind tunnel testing of additive manufactured joined-wing aircraft components," in *Rapid Prototyping Journal*.

Fig. 7-1 Superimposed images of the deflections captured for AOA 8° (a) from the rear, (b) from the side and AOA 10° (c) from the rear, (d) from the side.

and side. The images with the deflections were then superimposed on the stationary images for comparison.

For the AOA 8° , the white nylon wings provide good contrast against the black wind tunnel background and a MATLAB^{TM} script is able to correctly identify the wing deflections. The original stationary wing is demarcated in red while the deflected wing is marked out in blue in figure 7-2. Figure 7-2 also shows the geometry used in the numerical simulations.

For AOA 10° however, the increase in the pitch angle rendered part of the background to include the lightly coloured wind tunnel wall. This resulted in poor colour contrast and the deflection could not be identified with the same script, as seen in figure 7-3. In this case, a digitizer was used instead to visually pick out the deflections where 10 equidistant points were marked along the joined-wing trailing-edges to create smooth

Table 7-1 Maximum wing tip deflection magnitudes.

Wing/AOA	z [m]
Upper wing/8° AOA	0.0067
Lower wing/8° AOA	0.0068
Upper wing/10° AOA	0.0064
Lower wing/10° AOA	0.0086

Fig. 7-2 Comparison between (a) deflected wing geometry used in numerical simulations and (b) deflections captured in wind tunnel experiments.

Fig. 7-3 Failure to identify the wing due to poor colour contrast.

splines to model the curved wings. The wingtip deflections for both AOAs are summarized in table 7-1.

A few assumptions were taken for this numerical simulation for simplification. Firstly, only the curvatures of the upper and lower wings were accounted for. This was because the contribution to both the lift and drag forces from the middle wing was insignificant compared to those by the upper and lower wings. Hence the deflection of the middle wing was neglected although all the three wing sections (i.e. upper wing, lower wing, as well as the middle wing connecting the upper and lower wings) were observed to have

Fig. 7-4 Comparison of C_L obtained from wind tunnel experiments and numerical simulations with and without the deflections taken into account.

deflected. Secondly, the lower wing root was assumed to originate from the same location as before. Thirdly, the vertical tail remained non-deflected and was at the same location as before. Lastly, the deflections were only accounted for the vertical axis. The deflections in the horizontal axis, or the in the direction of the free-stream, were much smaller compared to the deflections in the vertical axis and were hence ignored. This assumption also greatly simplified the modifications required. For a fair comparison, the mesh created for this new geometry was similar in count to that used in the previous analysis and all other conditions and settings remained the same.

The discrepancy between the experimental and simulation C_L has decreased from 14% for both AOAs, to 8% and 6% for 8° and 10° AOAs respectively after the deflections were incorporated to the numerical simulations, and is shown in figure 7-4. On the other hand, the maximum discrepancy between the experimental and simulation C_D remains at 15%, regardless of whether deflections were taken into consideration. The outstanding discrepancies could be attributed to numerical errors typically encountered

Fig. 7-5 Comparison of C_p from numerical simulations along the upper surface for the (a) front wing and (b) rear wing at $AOA=8^\circ$, and the (c) front wing and (d) rear wing at $AOA=10^\circ$.

during numerical simulations, as well as the middle wing which was not modelled here. To better understand the effects of including the wing deflections into the numerical simulations, the C_p along the upper surface of the wing, as determined from the numerical simulations, are plotted at three different span-wise positions: 1/4, 1/2 and 3/4 of the semi-span (b). They are plotted against locations along the surface in the stream-wise direction normalized by the sectional chords for 8° and 10° AOAs with and without wing deflections taken into account. When comparing the C_p plots at different span-wise locations along the wing in figure 7-5, it can be seen that the magnitudes of the peak C_p are generally higher when the wing deflections were not included. On the other hand, exceptions occur at 8° AOA at 1/4s location and 10° AOA at 3/4s location

of the front wing. This produces higher overall C_L magnitudes and leads to instances where numerical simulation C_L magnitudes were higher than those obtained experimentally.

7.3 Wing Deflections of D150

Leveraging upon the experience from capturing the deflection in the PP250 case, to prevent cases where the deflections could not be detected in MATLAB^{TM} due to poor colour contrast, the following measures were taken. Firstly, the camera at the rear was set to be level with wings of the wind tunnel model when at AOA 0° and stayed fixed throughout the whole procedure. This prevents the issue of poor contrast due to the similarly coloured wind tunnel walls. This also greatly sped up the process as the camera now does not have to be adjusted for each angle.

Secondly, to further improve the colour contrast, the trailing edges of the wings were traced out with colour pencils. For the D150, red was used. It was found that the images were slightly shifted upwards due to the flow conditions in the test section. Cross-correlation was used to shift the images so that they would coincide with each other. A smooth quadratic spline was fitted to the data points with the least squares method, with a constraint to start from (0, 0). Figures 7-6 show an example of the process and result of the quantification of the wing curvature of the D150.

Thirdly, the leading edge and trailing edge of the wingtips were marked out with a felt-tip marker. By comparing the images before and after the deflections, the displacement vectors from the leading edge to the trailing edge were used to determine the average

Fig. 7-6 Determination of wing curvature for the D150 configuration.

Fig. 7-7 Determination of wing deflection and twist for the D150 configuration.

distances of the vectors between the two red dots and hence wing deflection and wing twist. Figure 7-7 shows the data collection procedure for the D150 configuration pictorially.

It was found that the wing curvatures could be satisfactorily represented by a quadratic curve as seen in figure 7-8. The magnitude of the wing curvature was first found to increase from AOA 0° to AOA 8° . It then decreases at AOA 10° , before increasing again from AOA 15° onwards. The end points of the curvatures at AOA 8° and AOA 20° were found to be approximately the same. Generally, the changes in the wing curvature and deflections should follow a similar trend to the lift curve, since the deformation in the wing structure is dependent on the aerodynamic forces acting on it.

This is the case and can be seen when the wing deflections were plotted with the lift curve on the same graph in figure 7-9. The local peak magnitude of the deflections occurs 1° earlier than the point where the C_L reaches its local maximum and starts to stagnate. In the region where the C_L stagnates, the deflection magnitude decreases. The

Fig. 7-8 Wing curvatures determined for D150.

local minima of the deflection magnitude occurs 4° earlier than the point where C_L reaches its local minima and starts to increase again. There exists a slight phase lag between the deflection and the C_L . In contrast, changes in the wing twist do not show any obvious correlation with the wing deflections or lift curve, as seen in figure 7-10. In fact, wing twist was observed to fluctuate severely at higher AOAs due to wing flutter. However, wing twist is estimated to reach a maximum of less than 1° throughout the entire range of AOAs tested, which is negligible as compared to wing deflection.

7.4 Wing Deflections of B150

For the B150 configuration, the same procedure as the D150 was undertaken. For the trailing edges, red, green and yellow were used to differentiate between the upper, lower and middle wings respectively. Figures 7-11 show an example of the process and result of the quantification of the wing curvature of the B150. However, for the B150 configuration, due to the connecting wing between the upper and lower wings, the markers on the trailing edge of the upper wing and the leading edge of the lower wing could not be captured on the image. Hence, the wing twist could not be adequately

Fig. 7-9 Comparison of wingtip deflection with C_L for D150 configuration.

Fig. 7-10 Change in wing twist for D150 configuration.

determined. Figure 7-12 shows the same procedure done for the B150 configuration and how two of the markers for wing twist could not be captured.

Similar trends in the wing deflections for the D150 case were found in the B150 case. Quadratic splines were sufficient in representing the wing curvatures, as seen in figure 7-13. For the front wing, the curvature increases as the AOA increases, with the exception between AOAs 5° and 10° . The end points of the curvatures at those two AOAs practically coincide. The main difference is that the curvature at AOA 10° is more pronounced than that of at AOA 5° . Based on this alone, it suggests stall and loss

Fig. 7-11 Determination of wing curvature for the B150 configuration.

Fig. 7-12 Determination of wing deflection and twist for the B150 configuration.

of lift on the outboard sections of the front wing. The curvature of the rear wing increases as the AOA increases with no exception. The rear wing is in the downwash of the front wing, causing stall of the rear wing to be delayed.

The trends in the C_L and the wing deflection are compared in figure 7-14. The B150 does not experience stall per se, but at an AOA of approximately 5° , the slope of the C_L decreases. Similar to the D150 case, the decrease in the slope of the wing deflection occurs at an AOA of approximately 4° , which is 1° earlier than the lift curve slope. It is interesting to note that at lower AOAs of less than 5° , both the lower and upper wings deflect by the same amount. However, generally at higher AOAs, between 5° and 13° , the front wing deflects more than the rear wing while the opposite occurs at AOAs greater than 13° with one or two odd exceptions. This implies that the middle wing is actually under tension between AOAs 5° and 13° and under compression at AOAs larger than 13° . Another implication is that at lower AOAs, both wings contribute equally to the lift generation. At higher AOAs, the lift generation distribution between

Fig. 7-13 Wing curvatures determined for (a) front wing and (b) rear wing of B150.

Fig. 7-14 Comparison of wingtip deflection with CL for B150 configuration.

the two wings are not symmetric. This would also mean structurally, both the lower and upper wing need not have the same stiffness, should the application of this configuration include high AOAs in the flight envelope, depending on the lift distribution of the wings. In contrast, the structural design of the middle wing would need more consideration in light of the different forces it might experience at different operating regions of the flight envelope.

Chapter 8 Conclusions

An experimental and numerical study was carried out to investigate the aerodynamic characteristics, as well as the flow behaviour, of two joined-wing configurations, as well as a typical commercial aircraft configuration for reference, at various orientations by permuting a combination of AOAs, sideslip angle as well as roll angle. The fuselage of the model was machined out of aluminium, while the wings were fabricated out of nylon through additive manufacturing due to the shape of the joined-wing configuration. Force balance measurements and surface oil flow visualization were carried out in the wind tunnel experiments, and numerical simulations were carried out with a commercially available code for corroboration. While the test conditions and model size did not allow for direct comparison to the actual flight conditions due to the constraints on the experimental setup, these experiments were still useful in aiding with the understanding of the fundamental flow behaviour.

8.1 Summary

Based on the wind tunnel experiments, the C_L of the joined-wing configurations were found to stagnate first as compared to the typical aircraft configuration and were also generally lower. There are several reasons for this. Firstly, there was a need to evenly distribute the lift distribution between the front and rear wings for the joined-wing configuration for longitudinal stability considerations. Secondly, for the PP250, with the large anhedral angle of the rear wings, the vertical component of force generated by the rear wings contributing to the lift was reduced. At higher AOAs, the rear wings were in the downwash of the front wings, thereby reducing the effective angle of attack. For the

B150, instead of the typical wash-out on swept wings, the optimization process resulted in wash-in on the wings. This wing twist distribution causes the wing tips to stall much earlier, resulting in less lift. As for the C_D , when comparing the B150 over the D150, there was no significant advantage of the joined-wing configuration over the typical configuration at the tested Reynolds numbers. The advantage of reduced induced drag was not sufficient to overcome the penalties in the skin friction drag due to the side wings. The numerical simulations also showed the same conclusion. In terms of the longitudinal stability, the joined-wing configurations showed no instability problems. In fact, the restoring moment for the joined-wing configuration was much stronger, due to the rear wings. In terms of the directional stability, the joined-wing configurations also performed on par with the typical aircraft configuration. However, for the lateral stability, there are some concerns with the B150 configuration, which might lead to spiral instability.

For the numerical simulations, they were sufficiently accurate, especially for the B150. Although the C_D was consistently over predicted in the numerical simulations at the lower AOAs, this was due to the entire flow field being assumed to be turbulent due to the turbulence model used. The numerical simulations were also unable to accurately model the flow physics of the laminar separation bubble at the lower AOAs. In addition, the discrepancies with the experimental results were higher when combinations of roll and yaw were implemented. However, the numerical simulations still proved useful in providing insights on the complex flow structures, especially with the combinations of roll and yaw, which were difficult to glean from the surface oil flow visualizations alone due to the 2D nature of the results when the flow field is highly 3D.

In addition, due to the material selection and physical size of the model, significant wing deflections were observed. These deflections were recorded and quantified. Preliminary attempts at incorporating these deflections were carried for the PP250, with improved matching of the C_L between the experimental and numerical results. For the B150 and D150 configurations, these deflections were photographed and digitized, then reproduced as a smooth curve through regression. The deflection magnitudes show a strong correlation with the lift curve, which was expected as the deflections are dependent on the aerodynamic forces acting on it.

8.2 Recommendations for Future Work

Based on the current work, the following possible avenues for further research are as proposed:

1. Further flow visualization techniques such as smoke injection would provide additional qualitative data, on top of the surface oil flow visualization, for added ease of analysis. 2D, or maybe even 3D, particle image velocimetry (PIV) can be used to collect quantitative data for analysis.
2. For the numerical simulations, at this low test speed, other turbulence models should be explored, especially those that include laminar-turbulent transition modelling, such as the transition SST model.
3. In addition, for the numerical simulations, higher velocities closer to the designed cruise speeds should be used. This would result in transonic effects and shocks that may

alter the conclusion of the aerodynamic efficiency of the joined-wing configuration. The low speed simulations were only done due to the limits on the experimental setup.

4. Due to the material selection, significant deflections were observed. For the numerical simulations, more can be done in terms of incorporating the wing deflections into the geometry, or even conducting fluid-structure interaction (FSI) simulations to predict the wing deflections. This would mean doing time-dependent and coupled simulations, instead of the current steady-state simulations. The recorded and quantified wing deflections from the wind tunnel experiments would be useful for validation.

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List of Publications

1. Journal Articles

Teo Z.W., New T.H., Li S., Pfeiffer T., Nagel B. and Gollnick V. Wind tunnel testing of additive manufactured joined-wing aircraft components. *Rapid Prototyping Journal*. (Accepted for publication, 2017).

Teo Z.W., New T.H., Li S., Pfeiffer T., Nagel B. and Gollnick V. Comparison of the Aerodynamic Performance between a Conventional Monoplane and Joined Wing with varying Pitch. (Under preparation).

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2. Conference Articles

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