A Comprehensive Methodology for Design of a Circulation Control Small-Scale Unmanned Aircraft

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A Comprehensive Methodology for Design of a Circulation Control Small-Scale Unmanned Aircraft

A Dissertation
Presented to
the Faculty of the Daniel Felix Ritchie School of Engineering and Computer Science
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In Partial Fulfillment
of the Requirements for the Degree
Doctor of Philosophy

by
Konstantinos Kanistras
March 2016
Advisors: Kimon P. Valavanis, Ph.D. and Matthew J. Rutherford, Ph.D.
Abstract

Unmanned Aerial Vehicles (UAVs) have become increasingly prevalent and important for a wide spectrum of civilian and military operations. When focusing on small-scale fixed-wing UAVs, payload, power and energy requirements limit considerably their utilization and flexibility allowing them to complete only those specific missions they are designed for. Circulation Control (CC) is an active flow control method used to produce increased lift over the traditional systems (flaps, slats, etc...) currently in use. This dissertation focuses on the foundations of a comprehensive methodology from design to implementation and experimental testing of Coandă-based Circulation Control Wings (CCW) for unmanned aircraft. The research goes beyond the current state of the art by demonstrating the feasibility of CC as applied to small-scale UAVs. 2-D and 3-D wind tunnel tests at Mach numbers of 0.03, with momentum coefficients of blowing ($C_\mu$) ranging from 0.0 to 0.3 are conducted. It is found that CC blowing is effective at all cases enabling the wing to achieve high lift-to-drag ratios and high lift augmentation during takeoff. The wind tunnel results indicate that upper slot blowing using CC can be effective for lift enhancement even at low blowing rates. Through flight testing it is confirmed that CC can be applied to small-scale UAVs resulting in significant runway reduction up to 53%. The technology described herein can be made suitable for use on commercial
airliners, cargo planes and personal aerial vehicles because equipping these aircraft with cruise-efficient high-lift devices can give the user more valid runway choices at existing airports and help alleviate environmental noise problems near airports by allowing steeper climb-outs.
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# Table of Contents

1 Introduction ........................................ 1
   1.1 Motivation ........................................ 3
   1.2 Problem Statement ................................ 4
   1.3 Method of Approach ................................ 5
   1.4 Summary of Achievements & Contributions ....... 7
   1.5 Dissertation Outline ............................ 9

2 Literature Review .................................... 10
   2.1 Numerical Analysis ............................... 10
   2.2 Experimental Work ............................... 14
       2.2.1 CC on Full Scale Aircraft ................. 14
       2.2.2 CC on Unmanned Aerial Vehicles .......... 19
   2.3 Discussion ........................................ 23

3 Fundamentals of Circulation Control ................. 26
   3.1 Vorticity & Circulation ......................... 26
       3.1.1 Kevin’s & Helmholtz’s Vortex Theorems .... 29
       3.1.2 Finite Wing Theory .......................... 30
   3.2 Coandă Effect .................................... 32
       3.2.1 2-D Analysis ................................. 33
       3.2.2 Coandă Jet Circulation Control .......... 34

4 Model Description ..................................... 36
   4.1 Coandă Surfaces ................................ 36
       4.1.1 Airfoil Shapes ............................... 38
       4.1.2 Dual Radius Flap Geometry ................ 41
       4.1.3 CCW Wind Tunnel Model .................... 44

5 Plenum Design ........................................ 46
   5.1 Plenum Geometry ................................. 47
# Wind Tunnel & Instrumentation

6.1 Wind Tunnel ................................................. 57
6.2 Force Balance ............................................... 58
   6.2.1 The Concept ............................................ 59
   6.2.2 Setup ................................................... 60
   6.2.3 Calibration ............................................. 61
6.3 Pitot Probe .................................................. 65

# Wind Tunnel Results & Discussion

7.1 No-Blowing & Blowing Case Comparison ...................... 70
   7.1.1 Effect of blowing on lift coefficient $C_L$ ........... 70
   7.1.2 Lift-to-Drag Ratio ..................................... 72
7.2 NACA 0015 CCW Configuration ................................ 84
   7.2.1 Wind Tunnel Test Conditions .......................... 84
   7.2.2 Cruise Flight Performance, 0° Flap Deflection .......... 86
   7.2.3 High-Lift Takeoff Performance, 30° Flap Deflection .... 88
   7.2.4 High-Lift Takeoff Performance, 60° Flap Deflection .... 90
   7.2.5 Further Data Analysis .................................. 92

# UC²AV: Unmanned Circulation Control Aerial Vehicle

8.1 The Platform ............................................... 95
8.2 Air Supply Unit & Air Delivery System ...................... 97
   8.2.1 Air Supply Unit ........................................ 97
   8.2.2 Air Delivery System .................................... 98
8.3 Circulation Control System .................................. 99
8.4 Circulation Control Wing .................................... 100
   8.4.1 CCW Structural Analysis ............................... 100
   8.4.2 NACA 0015 Wing Implementation ........................ 107
   8.4.3 CCW Implementation .................................... 110

# Takeoff Performance & Instrumentation

9.1 Takeoff Performance ......................................... 115
   9.1.1 Takeoff Distance ..................................... 115
   9.1.2 Pilot Takeoff Technique & Takeoff Corrections ........ 116
   9.1.3 One-at-a-Time Sensitivity Analysis .................... 118
9.2 Instrumentation .................................................. 120

10 Flight Testing ...................................................... 124
  10.1 Aircraft Controls & Data Collection ....................... 125
    10.1.1 V-Tail Controls ........................................ 125
    10.1.2 Aircraft Flight Controls ............................... 128
    10.1.3 The Airfield .......................................... 129
  10.2 Ground Effect ............................................... 129
  10.3 Flight Data Analysis ....................................... 131
    10.3.1 Anaconda with NACA 0015 Wing Flight Data ........... 132
    10.3.2 UC²AV Flight Data .................................... 135
    10.3.3 Flight Controls ....................................... 135
    10.3.4 Preflight Data Analysis ............................... 136
    10.3.5 Flow uniformity ...................................... 137
    10.3.6 Momentum Coefficient of Blowing .................... 138
    10.3.7 Flight Data Analysis ................................ 139

11 Conclusions & Future Work ...................................... 142
  11.1 Conclusions ................................................ 142
  11.2 Future Work ............................................... 144
## List of Figures

1.1 The proposed comprehensive methodology. ........................................... 7
2.1 Comparison of the instantaneous spanwise vorticity field around the airfoil in the TE region: (a) low- and (b) high-blowing cases [13]. . 12
2.2 Lift coefficient over angle of attack for different slot heights [14]. . 13
2.3 Variation of incremental lift coefficient with time-averaged mass flow rate [18]. .................................................................................. 14
2.4 A-6/CCW flight demonstrator aircraft [20]. ........................................ 15
2.5 CFD simulation of TE boundary layer control and jet entrainment penetration around a Coandă surface [21]. ........................................ 16
2.6 Coandă surfaces and slot heights [23]. ................................................. 17
2.7 Left: FAST-MAC model in cruise configuration in the NTF. Right: 3-view drawing with pertinent dimensions of the FAST-MAC semi-span model in cruise configuration [26]. .............................................. 18
2.8 Dual radius flap design [28]. ................................................................. 19
2.9 CC demonstrator plenum chamber in wing tip [31]. ............................ 20
2.10 Deamon UAV (FLAVIIR Project) [33]. ............................................... 21
2.11 Model design (upper and lower chamber) [34]. ................................. 22
2.12 CEETA testbed demonstrator UAV [35]. ......................................... 23
3.1 Left: Definition of circulation. Right: Circulation around a lifting airfoil [50]. .................................................................................. 28
3.2 Effect of circulation on flow around an airfoil at an angle of incidence [48]. .................................................................................. 29
3.3 Contour geometry for Kelvin’s circulation theorem. ......................... 30
3.4 Tangential blowing over a Coandă surface [54]. ............................... 32
3.5 Newman experimental setup [55]. ....................................................... 33
4.1 Flow near a TE [48]. ........................................................................... 37
4.2 Removable Coandă surfaces with length-to-height ratios of (1:1), (2:1), (3:1), (4:1). ................................................................. 37
4.3 Circulation Control wing model. ........................................ 39
4.4 The CCWs and the Clark-Y (calibration wing). ..................... 41
4.5 CC Coandă performance [66]. ........................................... 42
4.6 Dual radius flap design methodology. ................................. 43
4.7 DRF10 and DRF45 dual radius flaps at various deflection angles. . 44
4.8 Left: Isometric view of the CAD design of the wing. Right: Wing model placed on the sting of the force balance in the test section with the endplates for 2D wind tunnel testing. .......................... 45
5.1 Internal plenum of the S8036 CCW: The lower part of the model is divided into four areas which have individual tubes, connected with the main flexible high-pressure tube that provides the air in the plenum. ............... 47
5.2 Left: $V_{jet}$ performance at the slot exit. Right: Average $V_{jet}$ behavior. .. 48
5.3 Left: Top view schematic of the plenum design in the NACA 2412, NACA 23015 and Clark-Y. Right: Clark-Y plenum design. ......................... 48
5.4 Left: Trimetric and side view of the CAD Design. Right: Simulation Results of the Design. .................................................... 50
5.5 Experimental setup for $V_{jet}$ measurements. ............................. 51
5.6 Experimental and simulation results representing the performance of the $V_{jet}$ at the slot across the span. .............................. 51
5.7 Left: CAD design of the plenum. Right: Simulation results representing the performance of the $V_{jet}$ at the slot across the span. .................... 53
5.8 Plenum CAD design: The vanes at the inlet for flow correction and the vanes in the diffuser to achieve uniformity are shown. Nine standoffs are placed to avoid slot deformation during blowing. ....................... 54
5.9 Flow uniformity performance across the span of the wing. .......... 55
6.1 Sketch of the wind tunnel. ................................................. 57
6.2 Illustration of the test section of the wind tunnel. The red box depicts the location of the wing model. ............................ 58
6.3 Full-bridge strain gauge circuit [72]. .................................... 59
6.4 Full-bridge strain gauge circuit [73]. .................................... 60
6.5 2-beam force balance. ..................................................... 61
6.6 Load cells used on the force balance. .................................. 62
6.7 Relation between the load applied and the sensor reading. .......... 63
6.8 Left: Calibration of the force balance before wind tunnel testing.
Right: Calibration and testing of the force balance applying weights on both (lift and drag) directions to evaluate the lift and drag coefficients. ................................................................. 64
6.9 Pitot probe calibration schematic. ......................................................... 66
6.10 Pitot calibration curves. ................................................................. 67
7.1 Conceptual calibration model of an experimental setup [71]. .............. 69
7.2 The effect of blowing on drag force for all CCWs with the most effective (2:1) Coanda surface at $M = 0.022$. .............................. 73
7.3 The effect of blowing on drag force for all CCWs with the most effective (2:1) Coanda surface at $M = 0.041$. .............................. 74
7.4 Histogram representation for the lift-to-drag ratio versus Angle of Attack. ................................................................................. 75
7.5 The incremental lift coefficient plotted against the moment coefficient ($C_\mu$) for all Coanda surfaces at $\alpha = 0^\circ$. .................... 77
7.6 The incremental lift coefficient plotted against the moment coefficient ($C_\mu$) for all Coanda surfaces at $\alpha = 2^\circ$. .................... 78
7.7 The incremental lift coefficient plotted against the moment coefficient ($C_\mu$) for all Coanda surfaces at $\alpha = 14^\circ$. ...................... 79
7.8 Effect of $C_\mu$ on measured drag coefficient at $\alpha = 0^\circ$. ............ 80
7.9 Effect of $C_\mu$ on measured drag coefficient at $\alpha = 14^\circ$. ............ 81
7.10 $C_\mu$ effect on Lift augmentation ratio for the (2:1) Coanda surface. .... 83
7.11 2-D testing arrangement with two endplates in the wind tunnel. ....... 85
7.12 0° deflection flap comparison at no blowing case. ......................... 86
7.13 Efficiency of the 0° deflection flap configurations at $-2^\circ$, $0^\circ$ and $2^\circ$ angle of attack respectively. .............................................. 87
7.14 Effect of $C_\mu$ on lift with the DRF10 flap (Left) and with the DRF45 flap (Right). ................................................................. 88
7.15 Effect of $C_\mu$ on drag with the DRF10 flap (Left) and with the DRF45 flap (Right). ................................................................. 88
7.16 The effect of blowing on the DRF10 flap at different angles of attack. 89
7.17 The effect of blowing on the DRF45 flap at different angles of attack. 90
7.18 Lift and drag performance for both flap configurations at no blowing case. .............................................................................. 91
7.19 Effect of blowing on lift coefficient for DRF10 (Left) and DRF45 (Right) at different angles of attack. ................................. 91
7.20 Effect of blowing on lift coefficient for DRF10 (Left) and DRF45 (Right) at different angles of attack. .................................................. 92
7.21 Effect of blowing on lift-to-drag ratio with the DRF10 at 0° deflection. 93
7.22 Comparison of DRF10 and DRF45 at 0° (Left) and 18° (Right) angle of attack. .............................................................. 93
7.23 Comparison of DRF10 and DRF45 at 0° (Left) and 18° (Right) angle of attack. .............................................................. 94
8.1 RMRC Anaconda UAV. .......................................................... 96
8.2 Left: The housing. Center: The impeller. Right: The motor mount. The weight of the ASU is 85 g without the motor. .............. 98
8.3 Left: CAD designs of junction configurations 1-4. Right: Velocity contour of the junctions at 50 m/s inlet velocity from CFD analysis. . 99
8.4 Left: Circulation Control system. Right: CAD design of the CC system. ............................................................... 100
8.5 Wing structure design using Solidworks. .................................. 101
8.6 Wing structure design on ANSYS Workbench. Configuration-III when 1g condition is applied (total deformation results). ............ 105
8.7 Left: The wing structure with the two carbon fiber rods and the ten ribs, equally spaced (100 cm). Right: The balsa leading- and trailing-edge parts and the balsa wood skin. ........................................... 108
8.8 Left: MonoKote tape for smooth surface finish. Right: The UAV with the NACA0015 wings. .................................................... 108
8.9 Wing structure components. .................................................... 111
8.10 Wing structure. .................................................................... 112
8.11 The CCW. .......................................................................... 112
8.12 The UC²AV. ........................................................................ 113
9.1 Takeoff performance expected behavior of the UC²AV compared to a conventional UAV. ......................................................... 115
9.2 The instrumentation that is on-board the UC²AV. .................... 121
9.3 Overview of the instrumentation system of the UAV. .............. 122
10.1 Bottom: Pilot input (roll, pitch, yaw and throttle). Top: The pitch and roll responc and the readings of the three pitot probe sensors. . 127
10.2 Top: The pitch angle recorded from the IMU. Bottom: The recorded data from the ultrasonic distance sensor during takeoff. ........ 127
10.3 Anaconda aircraft integrated with NACA0015 conventional wing flights controls data flowchart. ........................................... 128
10.4 Miniature Aero Sportsters: Remote Control (RC) field. 129
10.5 Anaconda with the conventional NACA 0015 wing integrated. 131
10.6 The UC\textsuperscript{2}AV flight controls data flowchart. 136
10.7 The UC\textsuperscript{2}AV’s CCW Solidworks design. 137
10.8 Experimental results representing the performance of the $V_{jet}$ at the slot across the span on each plenum for different RPM values. The deviation of the $V_{jet}$ from the average line (shown in black) at both RPM value is shown. 138
10.9 Top: The pitch and roll response and the readings of the three pitot probe sensors. Bottom: Pilot input (Roll, Pitch, Yaw and Throttle). 140
List of Tables

1.1 UAVs Classification according to the US DoD [1]. 2
2.1 Circulation Control - summary of existing experimental and computational research. 24
4.1 CCW chord lengths and wing area. 38
4.2 Baseline CC dual radius flap design parameters. 42
4.3 Dual radius flap design parameters. 43
6.1 Free stream velocity at various points in a cross section of the test section. 58
7.1 Maximum $\Delta C_L$ recorded for all CCWs in this experimental investigation. 71
7.2 Maximum $\Delta C_D$ recorded for all CCWs in this experimental investigation. 72
8.1 Anaconda RMRC geometric characteristics. 96
8.2 Input parameters of the wing structure. 102
8.3 Design parameters of the tested configurations. 104
8.4 Stress values at various $g$ Conditions for wing Configuration-I. 105
8.5 Stress values at various $g$ Conditions for wing configuration-II. 106
8.6 Stress values at various $g$ Conditions for wing Configuration-III. 106
8.7 Geometric parameters of the wing. 109
9.1 Comparison between the predicted takeoff distance the flight data. 119
9.2 Comparison between the predicted takeoff distance and the flight data. 120
9.3 Instrumentation/Sensor Specifications for UC$^2$AV. 123
10.1 Flight test data. 133
10.2 Theoretical and flight test data comparison. 134
10.3 CC Flight test data. 140
Nomenclature

\[ \begin{align*}
\alpha & \quad \text{Angle-of-attack (degrees)} \\
\theta_{sep} & \quad \text{Angle-of-separation (degrees)} \\
TE & \quad \text{Trailing edge} \\
P & \quad \text{Pressure (Pa)} \\
S & \quad \text{Wing surface area (m}^2\text{)} \\
c & \quad \text{Chord (mm)} \\
b & \quad \text{Wing span (mm)} \\
r & \quad \text{Radius (mm)} \\
r_1 & \quad \text{First Coand\u0103 radius (mm)} \\
r_2 & \quad \text{Second Coand\u0103 radius (mm)} \\
h & \quad \text{Slot height (mm)} \\
H & \quad \text{Wing distance from ground (mm)} \\
c_{ref} & \quad \text{Reference chord (150 mm)} \\
A & \quad \text{Cross-section area (m}^2\text{)} \\
\dot{m} & \quad \text{Mass flow rate (kg/s)} \\
\dot{m}_p & \quad \text{Mass flow rate per unit span} \\
C_\mu & \quad \text{Moment coefficient of blowing} \\
M & \quad \text{Mach number}
\end{align*} \]

xiv
\[ \rho \] Density of air (kg/m³)
\[ x \] Chordwise distance (mm)
\[ V_{jet} \] Velocity at the jet (m/s)
\[ V_{T.O.} \] Velocity at takeoff (m/s)
\[ V_{L.O.} \] Velocity at liftoff (m/s)
\[ V_\infty \] Free stream velocity (m/s)
\[ C_L \] Finite wing lift coefficient
\[ L \] Lift (N)
\[ L' \] Lift per unit span
\[ C_D \] Finite wing drag coefficient
\[ D \] Drag force (N)
\[ C_d \] Drag coefficient of the infinite wing (2-D wing)
\[ C_l \] Lift coefficient of the infinite wing (2-D wing)
\[ \Delta \] Incremental change
\[ \Delta C_L/C_\mu \] Lift augmentation ratio
\[ q_\infty \] Dynamic pressure (Pa)
\[ e \] Span efficiency factor
\[ a_o \] Lift slope \( dC_L/d_\alpha \) of the infinite wing (2-D airfoil)
\[ a \] Lift slope \( dC_L/d_\alpha \) of the finite wing (3-D wing)
\[ dc \] Differential length along a control surface (mm)
\[ n \] Load factor
\[ \mu \] Resistance coefficient
\[ K \] Induced drag parameter
\[ K_g \] Induced drag parameter at the ground
\[ \lambda \] Headwind percentage of ground speed
\begin{itemize}
\item $W$ Weight (kg)
\item $W_{T.O.}$ Takeoff weight (kg)
\item $\ell_{T.O.}$ Takeoff distance (m)
\item $\ell_{L.O.}$ Liftoff distance (m)
\item $\ell_{ground}$ Ground distance (m)
\item $\ell_{air}$ Air distance (m)
\item $\ell_{groundW}$ Ground distance with wind (m)
\item $\ell_{airW}$ Air distance with wind (m)
\item $\zeta$ Altitude measured from the ultrasonic sensor (m)
\item $Z$ Altitude measured from the barometric sensor (m)
\item $AR$ Aspect ratio
\end{itemize}
Chapter 1

Introduction

Unmanned aviation has expanded exponential growth over the last years, and civilian applications are expected to dominate the field in the near future. Unmanned Aerial Vehicles (UAVs) have specific advantages over manned aviation, but they also have mission limitations due to payload restrictions, power supply, etc. These limitations, affect considerably their utilization and flexibility allowing them to complete only specific missions. For missions requiring heavier payload, it is advantageous if the UAV can adapt and generate more lift than initially designed for, rendering it suitable for diverse missions.

According to the U.S. Department of Defense (DoD), UAVs are classified into five classes based on size, takeoff weight, operating altitude and airspeed, as shown in Table 1.1. Small-scale Class I (Maximum Gross Takeoff Weight (MGTW) <20 lbs) unmanned aircraft will be the first ones to be integrated into the National Airspace System (NAS). It may be argued that they offer a cost-effective alternative for new technology implementation and testing before full-scale flight tests demonstration.
Table 1.1: UAVs Classification according to the US DoD [1].

<table>
<thead>
<tr>
<th>Category</th>
<th>Size</th>
<th>Maximum Gross Takeoff Weight (MGTW) (lbs)</th>
<th>Normal Operating Altitude (ft)</th>
<th>Airspeed (knots)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Class I</td>
<td>Small</td>
<td>0-20</td>
<td>&lt;1,200 Above Ground Level</td>
<td>&lt; 100</td>
</tr>
<tr>
<td>Class II</td>
<td>Medium</td>
<td>21-55</td>
<td>&lt;3,500 Above Ground Level</td>
<td>&lt; 250</td>
</tr>
<tr>
<td>Class III</td>
<td>Large</td>
<td>&lt;1320</td>
<td>&lt;18,000 Mean Sea Level</td>
<td>&lt; 250</td>
</tr>
<tr>
<td>Class IV</td>
<td>Larger</td>
<td>&gt;1320</td>
<td>&lt;18,000 Mean Sea Level</td>
<td>Any airspeed</td>
</tr>
<tr>
<td>Class V</td>
<td>Largest</td>
<td>&gt;1320</td>
<td>&gt;18,000 Mean Sea Level</td>
<td>Any airspeed</td>
</tr>
</tbody>
</table>

In this research the focus is on small-scale Class I UAVs. Such platforms operate at low Reynolds numbers ($Re < 8 \times 10^5$); thus, low speed wind tunnel testing is considered sufficient for performance evaluation.

In general, when an active flow control system is developed issues related to: effectiveness, energy efficiency and ease of implementation need to be considered. The method to be used should provide enhanced aerodynamic performance (lift enhancement, drag reduction, stall margin increase, enhanced payload, runway reduction, etc.). Further, it should not require significant energy expenditure (minimum power penalties) and should not be difficult to implement [2]. Circulation Control (CC) is an active flow control method used to produce increased lift over the traditional systems (flaps, slats, etc.) that are currently in use. This method is applied to increase the aerodynamic circulation around a wing by blowing over a Coandă surface, which is a rounded or near rounded trailing edge (TE) of the wing, resulting in lift enhancement. CC keeps the boundary layer jet attached to the wing surface longer, compared to a conventional wing and, as a result, increases the lift generated on the wing surface.

This dissertation describes a detailed methodology for design, development and flight testing of an unmanned circulation control aerial vehicle (UC$^2$AV) capable of reducing the required runway distance during takeoff. More specifically, this research investigates the geometry characteristics of a Circulation Control Wing (CCW) and
because of extensive wind tunnel testing it provides insight into the geometrical effects of the dual radius flaps in CC. Additionally, the design and development of a complete CC system for small-scale UAVs is described in detail and flight testing is conducted to demonstrate the efficiency of CC on UC²AVs compared to conventional unmanned aircraft.

1.1 Motivation

As previously stated, CC is a promising technique in many industries already applied to wind turbines, rotorcraft airplane wings, underwater vehicles, among other applications [3, 4, 5, 6]. CC techniques in aerodynamics have been researched and developed over the years and many designs and configurations have been proposed focusing on increasing lift performance, thus, and replacing the conventional flap systems of aircraft.

CC is applied to increase the aerodynamic circulation around the wing by blowing high energy air over a Coandă surface, resulting in lift enhancement and increased useful payload. This problem is challenging, particularly when considering power, energy and payload limitations of small-scale fixed-wing UAVs, as it is difficult to generate significant blowing without imposing weight and/or power penalties. Literature review reveals that as of now there is no a generalized methodology for designing and testing CCWs for fixed-wing unmanned aircraft.

Five challenging and open research questions have motivated the research:

i Is it possible and feasible to apply CC on small-scale unmanned aircraft with limited power and weight penalties?
ii What type of air supply unit (ASU) and air delivery system (ADS) should be
developed and implemented on-board the UAV to efficiently achieve the required
mass flow for CC?

iii What needs to be done to overcome major roadblocks like unfavorable trade-offs
of mass flow, pitching moment and cruise drag in order to apply CC on UAVs?

iv What is the complexity of a wing structure and the weight penalties on small-scale
UAVs?

v How effective is CC during takeoff if only 1/3 of the wingspan is used for upper
surface blowing?

The proposed comprehensive methodology provides answers and solutions to the
above questions.

1.2 Problem Statement

Based on the literature review, many CC designs have been proposed and in-
vestigated since the 1970’s for conventional and military aircraft. However, CC
technology has only been applied on a few UAVs due to the complexity of the design
and the mass flow rate requirements. There are several challenges when applying
CC in small-scale aircraft and these include: source of air (typically bleed or by-
pass air from the engine or an addition of an auxiliary power unit); weight penalties
due to the internal air delivery system; additional power penalties due to the air
supply unit’s power consumption and cruise drag penalties due to Coandă surface
geometry. These challenges are used as a guide to design, develop and evaluate a
Circulation Control Wing (CCW) capable of achieving high lift augmentation ratios
at low blowing coefficients. Small-scale fixed-wing UAVs have restrictions associated with space and power, thus, CC becomes a challenging issue. Engine bleed air (as the main source for CC) is only an option for bigger scale UAVs where there are fewer weight and space restrictions. For small-scale aircraft a light-weight Air Supply Unit (ASU) that provides sufficient mass flow to a CCW needs to be considered. Also, the internal plenum design responsible for distributing the air at the slot exit, is mainly dictated by manufacturing constraints within the limited space available.

1.3 Method of Approach

The comprehensive methodology centers on the development of a new generation of fixed-wing UAVs endowed with improved aerodynamic efficiency (enhanced endurance), increased useful payload (fuel capacity, battery cells, on-board sensors) during cruise flight, delayed stall, and reduced runway during takeoff and landing. These advances can be achieved by using and implementing the concept of CC. The steps followed to establish the needed foundation are depicted in Figure 1.1.

The method of approach is composed of seven coupled phases: theoretical and mathematical analysis, design, simulation, 3-D printing prototyping, wind tunnel testing, wing implementation and integration, and flight testing. The theoretical analysis focuses on understanding the physics of the flow and on defining the design parameters of the geometry restrictions of the wing and the plenum. The design phase centers on: designs of Coandă surfaces based on wing geometry specifications; designing and modifying airfoils from well-known ones (NACA series, Clark-Y, etc.); plenum designs for flow uniformity; dual radius flap designs to delay flow separation and reduce cruise drag. The simulation phase focuses on Computational Fluid Dy-
namics (CFD) analysis and simulations, and on calculating lift and drag coefficients of the designed CCWs in a simulation environment. 3-D printing and prototyping focuses on the actual construction of the CCWs. Wind tunnel testing centers on experimental studies in a laboratory environment. One step before flight testing is the implementation of the qualified CCW and the integration on the UAV platform. Flight testing is the final phase, where design validation is performed.
1.4 Summary of Achievements & Contributions

The primary contribution of this work is the development and evaluation of a small-scale unmanned circulation control aerial vehicle (UC^2AV) capable of reducing...
by about 50% the runway distance during takeoff. To the best of the author’s knowledge, this is the first time CC has been demonstrated to provide tangible enhancement during flight testing of a small-scale UAV. The achievements and contributions are summarized as follows:

i  Determine the best configuration (high lift coefficients and low induced drag values during cruise flight) of the wing design and the curvature of the Coandă surface.

ii  Derive SolidWorks based designs and models, and perform simulations using ANSYS, XFLR5 (this step is essential to calculate lift, drag and moment coefficients).

iii  Design and build a plenum capable of providing highly uniform flow across the span of 400 mm.

iv  Design and build five CCW wind tunnel models.

v  Design and build numerous dual radius flaps.

vi  Conduct 2-D and 3-D wind tunnel testing/experiments to investigate the efficiency of CC with upper slot blowing.

vii  Integrate and install an Air Supply Unit (ASU) component on-board the UAV.

viii  Integrate and install a duct system on the UAV.

ix  Build the final CCW and integrate it on the UAV.

x  Build the UC²AV and integrate the required instrumentation.

xi  Conduct flight tests and reduce the runway takeoff distance by 50%.
Achieve lift coefficient enhancement up to 93% during takeoff compared to a non-blowing case.

Although CC on small UAVs has not received much attention, this research demonstrates that a sufficient lift increment may be achieved with blowing coefficients at the boundary layer regime. Lift enhancement on a fixed-wing UAV can be achieved during level flight and take-off with reasonably low blowing rates. It is shown that the UC²AV, which consists of a conventional fuselage integrated with a modified NACA 0015 CCW with dual radius flaps, an ADS (air delivery system) and an ASU (air supply unit) can achieve takeoff maneuvers with up to 50% less runway infrastructure.

1.5 Dissertation Outline

The remainder of this dissertation is organized as follows: Chapter 2 presents a literature review to provide background information on computational and experimental work done on CC for manned and unmanned aircraft. Chapter 3 establishes the theoretical background and the fundamentals of CC. Chapter 4 includes the wind tunnel model description while Chapter 5 details the Air Supply Unit (ASU) that is designed and built for CC. Chapter 6 presents details of the wind tunnel and instrumentation used for subsequent experiments. Chapter 7 discusses obtained results. Then, in Chapter 8 the overall Unmanned Circulation Control Vehicle (UC²AV) and its subsystems are presented. Chapter 9 deals with the theory of takeoff performance and the instrumentation that is required, and Chapter 10 presents the flight testing results. Finally, Chapter 11 summarizes the research performed throughout this dissertation and describes the work needed to further advance the technology.
Chapter 2

Literature Review

The first reported use of CC is from H.Haagedorn and P.Ruden in 1938, focusing on investigations of boundary layer control on a flap [7]. The first CCW design was patented in the 1960’s [8] and it was the first documented application of CC as a lift augmentation methodology on fixed-wing aircraft. Extensive theoretical, numerical and experimental research has been conducted since then, and published research is presented in the next two sections.

2.1 Numerical Analysis

Numerical investigations for varied slot heights at different flap deflection angles and Mach numbers as well as leading edge blowing have been investigated in the past to optimize the high-lift performance of different CCWs. Simulation tests allow researchers to get a better understanding of the physics of the flow on Coandă surfaces. Pfingsten et. al. [9] focus on the flow around CC airfoils. The flow solver used is based on a finite volume scheme and the turbulence model is the Spalart
Allmaras (SA) [10]. The simulation results are compared with a CC profile with a round TE that is tested in a wind tunnel. The comparison of the numerical and experimental results shows that the standard (SA) does not predict the position of the separation correctly. However, the Spalart Allmaras for rotation and/or curvature effects (SARC) model is able to predict the position of the detachment quite well. Simulation tests are also conducted for blowing over a flap and the capability of generating high-lift coefficients with a gapless high-lift device during cruise flight is shown. Nishino and Shariff [11] investigate the influence of the jet-nozzle-lip thickness on the overall airfoil performance. The CDP flow solver is used for the computational testing which is an incompressible Navier-Stokes code developed at the Center for Turbulence Research at Stanford University [12]. The results show that transitions due to turbulence are not related with the nozzle-lip thickness. On the other hand, the size of wake that is created behind the nozzle lip does depend on the thickness of the nozzle tip. The thicker the lip, the larger the decrease of the time-averaged jet velocity downstream. The authors suggest reducing the nozzle thickness up to the point that is feasible from a structural point of view because the momentum loss will be reduced and that will satisfy the objective of CC and lead to lift enhancement. Equation 2.1 can be used to calculate the momentum coefficient after the momentum loss ($C_{\mu}^*$), but it is not straightforward to estimate the momentum loss for different nozzle cases a priori.

$$C_{\mu}^* = \frac{\dot{m}_{jet} V_{jet} - [\text{momentum loss}]}{q_{\infty} S}$$  \hspace{1cm} (2.1)

Madavan and Rogers [13] focus on a direct numerical simulation of the flow around a CC airfoil for high ($C_{\mu} = 0.12$) and low ($C_{\mu} = 0.044$) momentum coefficients of
blowing. The airfoil has a thick non-cambered elliptical leading edge and a semi circular Coandă TE. Figure 2.1 shows the flow around the TE in case of high and low momentum coefficient of blowing.

Figure 2.1: Comparison of the instantaneous spanwise vorticity field around the airfoil in the TE region: (a) low- and (b) high-blowing cases [13].

Jensch et. al. [14] conduct numerical simulations using steady-state Reynolds-averaged Navier-Stokes for a two-dimensional airfoil using CC in order to increase the efficiency of the CC system. The results show that a second slot at the leading edge with additional blowing prevents the occurrence of a thin separation bubble near the leading edge. The results show, as Figure 2.2 depicts, that the variation of the slot height is useful to increase the efficiency of the CC airfoil.

A 2-D and 3-D lift and drag computational analysis on a CCW is conducted in [15] by Montanya and Marshall in order to investigate if Extreme Short Takeoff and Landing (ESTOL) vehicles can use this method to shorten the landing and takeoff distances. According to the author, two dimensional and dual radius CC airfoils are chosen because dual radius airfoils are the only simple systems that allow both low cruise drag and a large Coandă flap surface which provides high-lift values. The equations that are used to calculate the balanced field length (BFL) which was found in [16] and the landing distance equation which was found in [17] are shown below. The results reveal that the shortest BFL and landing distance are well within the
mission requirements that are set by NASA Ames. It is concluded that CC is a viable solution for high-lift applications and should be used in the future on ESTOL aircraft.

\[
BFL = \left( \frac{0.863}{1 + 2.3G} \right) \left( \frac{W}{S} \right) \frac{1}{\rho g C_{L_{climb}}} \left( 1 + \frac{1}{TAV/W - U + 2.7} \right) + \frac{655}{\sqrt{\frac{\rho}{\rho_s L}}} \right) \quad (2.2)
\]

\[
S_{LAND} = \frac{1.69W_L^2}{\rho S_{ref} C_{L_{max}} g [D + \mu_{brake}(W_L - L)]} \quad (2.3)
\]

Liu et. al. [18] conduct numerical simulations of steady and pulsed blowing of a CC airfoil. A 2-D and 3-D configuration is tested with a Mach number of 0.0836 and a Reynolds number of $3.95 \times 10^5$. The steady blowing 2-D case at zero angle of attack CC is applied with $C_{\mu}$ of 0.1657 resulting in almost 3 times higher lift coefficient. On
the other hand, the pulsed jet configuration gave larger increments in lift compared to the steady jet at a given time as Figure 2.3 depicts.

![Figure 2.3: Variation of incremental lift coefficient with time-averaged mass flow rate](image)

2.2 Experimental Work

2.2.1 CC on Full Scale Aircraft

In the late 1970’s [19] CC was applied by Robert J. Englar on the TE of an A-6 flight demonstrator fixed-wing aircraft, in order to achieve larger lift coefficient values during takeoff and landing (Figure 2.4). A system that combined a CCW with upper surface blowing was developed with a NACA 64A008.4 (modified) model. The 2-D wind tunnel tests show a lift coefficient of 6.5 and the three-dimensional case was found to be a factor of 2.2 greater than the non-blowing case. In continuation to this project, Nichols and Englar demonstrate that almost 100% of the actuators, flaps and other high-lift devices can be replaced by CC. Also, it is concluded that a reduced number of parts and reduced impact loads on the aircraft will increase reliability, maintainability, and aircraft lifespan.
Jones and Englar [21] conducted research on high-lift concepts and how that addresses drag reduction and mass flow reduction by using pulsed pneumatic blowing on a Coandă surface. Due to issues associated with engine bleed and blunt blown TE during the cruise flight, pneumatic control of aerodynamic high-lift techniques have not been frequently used on production aircraft [21]. Two supercritical types of 2-D airfoil designs are discussed in [21]. Dual-slot blowing modification is used in order to create a virtual TE to minimize cruise drag while at the same time maintaining an effective Coandă surface. The objective is to reduce the required blowing mass flow by using techniques such as the pulsed blowing and compare it with the steady blowing. The CFD simulation results show that the degree of jet turning around the Coandă surface can be related to flow separation from the surface, to the slot height, the surface radius, the jet velocity and even to the geometry of the Coandă surface. It is also shown that as the blowing level increases, the separation point moves around the Coandă surface towards the maximum of the airfoil (x/c =1) as Figure 2.5 shows.

After the pulsed blowing tests it is verified that a given lift value can be obtained at lower time-average mass flow rates than steady state flow but it also confirms previous research that claimed that for steady state blowing higher jet velocity ratios
Figure 2.5: CFD simulation of TE boundary layer control and jet entrainment penetration around a Coandă surface [21].

from smaller slots at constant $C_\mu$ produces better entrainment and lift augmentation [6], [22].

At NASA Langley Research Center (LaRC), a wind tunnel experiment is conducted on a six percent thick camber elliptical CC airfoil with both upper and lower blowing. Three elliptical TE surfaces are manufactured (Figure 2.6). Alexander et. al. [23] test three upper and lower slot heights for each Coandă surface (Figure 2.6). It is shown that by decreasing the slot height and increasing the Coandă surface the effectiveness increases at transonic cruise conditions and angle of attack equal to 3°. At low-speed conditions and at the same angle of attack, the effectiveness increases by decreasing both parameters. Also, TE blowing influenced the flow field upstream of the slot. Based on lift and momentum coefficient data, no appreciable Coandă surface nor slot height preference is found with dual slot blowing. However, dual slot blowing resulted in a reduction of the airfoil’s baseline drag at $M = 0.8$. 

16
Figure 2.6: Coandă surfaces and slot heights [23].

Dual Radius Flap

NASA LaRC is continuing research with the Fundamental Aeronautics Subsonic/Transonic Modular Active Control (FAST-MAC) model in the National Transonic Facility (NTF) focusing on viscous flow separation at full-scale Reynolds numbers [24, 25]. The FAST-MAC model, shown in Figure 2.7, has a modern super-critical wing and is designed to become an NTF standard for evaluating performance characteristics of integrated active flow control and propulsion systems. CFD simulations and wind tunnel testing at different moment coefficients of blowing for enhanced lift is conducted. Various slot heights are investigated and a new tailored spanwise blowing technique is demonstrated to reduce mass flow requirements. The results show that a slot height to chord ratio of 0.0022 is more beneficial than larger slot height and can have the same performance with a 30% lower momentum coefficient of blowing. The FAST-MAC is also tested at various flap deflections and low speed (M = 0.2) and high speed (M = 0.88) values to investigate the lift augmentation and the efficiency of drag reduction using CC.
Research has shown that CC flap systems offer significant payoffs in both performance and system complexity. CC flap systems augment aerodynamic forces by entraining and deflecting the airfoil flow field pneumatically, rather than solely by deflecting a mechanical surface [27, 28]. Englar’s dual-radius flap [27] configurations represent specially designed internally blown flaps using the Coandă effect at their curved leading edge, therefore, they are called Coandă flaps. Golden and Marshall [28] explored the design of various CC flap systems on a supercritical airfoil (NASA SC(2)-0414) (Figure 2.8) using 2-D CFD analysis and concluded that the largest lift augmentation was achieved with the shorter dual radius flap. However, large negative pitching moments are associated with such lift augmentation, along with large drag penalties resulting in the lowest L/D values of all flap configurations.

Jensch et al. [29, 30] conducted design sensitivity studies that led to the selection of particular flap configurations, where the most important design parameters are flap deflection angle, momentum coefficient, and blowing slot height. It is shown that flap angle and blowing momentum coefficient should increase for increased lift.
targets and good values for the flap length are found to be 0.25-0.30 of the airfoil chord.

### 2.2.2 CC on Unmanned Aerial Vehicles

CC is first used on an unmanned model aircraft scale (All Up Weight < 7 kg) at the University of Manchester [31]. The goal of this project is to use flapless flight control technologies on a UAV less than 7 kg. The Irvine Tutor 40 almost ready-to-fly aircraft shown in Figure 2.9 is used. A fully operational flight control system based on flow control and fluidic thrust vectoring technologies is developed. Those flow control techniques have not been successfully implemented as primary flight controls according to the author [31]. A pneumatic system design that provided roll control through asymmetric lift augmentation. That system worked by blowing a jet of air through a thin slot above a circular TE. Two options to provide air supply are considered. Compressed air bottle supply is one, but was soon ruled out due to weight restrictions. Finally, a lightweight modified turbocharger (able also to convert
the centrifugal air compressor into an axial type) gives mass flow rate values that measured to be 0.06 Kg/s at 3kPa, which are suitable values for both CC and control fluidic thrust vectoring on the aircraft.

Figure 2.9: CC demonstrator plenum chamber in wing tip [31].

A method to estimate the lift curves or changes of lift curves of a CCW at low speeds is proposed by Ran et. al.[32]. The method first estimates the jet speed out of the slot on information provided for the mass flow rate by using a valid assumption of the loss of some total pressure because of viscous friction from the plenun to the nozzle exit. After a series of calculations, the authors show that the jet speed can be calculated by the following equation:

\[ V_{jet} = \frac{\dot{m}_{jet}}{\rho_{jet} A_{jet}} = \frac{\dot{m}_{jet} R T_{duct}}{A_{jet} a P_{duct}} \left(1 + \frac{\gamma - 1}{2} M_{jet}^2 \right)^{\frac{1}{\gamma - 1}} \]  \hspace{1cm} (2.4)

Equation(2.4) has two solutions for \( M_{jet} \), one for subsonic jet speeds and one supersonic if the ratio \( a P_{duct} = P_\infty \) is greater than 1.89. An empirical method that estimates the lift coefficient of the wing and the change of the wing-lift coefficient caused by the CC is also proposed.

Buonanno and Cook [33] focus on flight dynamics on the Flapless Air Vehicle Integrated Industrial Research (FLAVIIR) project (Figure 2.10) which was a five year research program with the collaboration of ten British Universities. The main
The objective of this program is the development of technologies that can provide low cost flapless UAVs without the conventional control surfaces. *Demon* which is a delta wing tailless configuration and powered by a gas turbine engine is selected for this project. For CC, a flow control actuator capable of proportional bidirectional control is used. To design the model of the actuator a single upper slot was used to provide CC. According to the authors [33], the performance of CC depends on the momentum coefficient and thus the pneumatic system is characterized as a function of it. An analysis on the equations is given and the dependence on slot height is explained. It is also mentioned that the smaller slots provide higher jet velocities for a constant momentum coefficient. Finally, a detailed analysis regarding the dependence of the actuator control angle on the incremental lift is given.

![Figure 2.10: Deamon UAV (FLAVIIR Project) [33].](image)

Lance W. Traub and M. Biegner [34] from Embry-Riddle Aeronautical University conduct an experimental investigation in order to evaluate a self-contained CCW. A rapid prototyped S8036 airfoil with an aspect ratio of 3.34 is used for the wind tunnel tests at low Reynolds numbers. The wing design is divided in two chambers: the upper for ingested air and the lower for air ejection. The two impellers shown in Figure 2.11, which are driven by two outrunner brushless motors, draw air from the inlet vents that extend from the wing to each impeller’s eye.
The jet-momentum coefficients are found to be low, however it is shown that stall was delayed significantly. Significant increase in the minimum drag coefficient is observed due to the Coandă surface which was causing a reduction on the range and the endurance parameters.

N. R. Alley et al. [35] designed and developed the Cruise-Efficient Extremely Short Takeoff and Landing (ESTOL) Transport Aircraft testbed (CEETA) (Figure 2.12) that has a span of approximately 136 inches, length of 142 inches, and an overall height of 51 inches (Boeing 737 scale model UAV). It is using a zero-sweep zero-taper rectangular wing with a symmetrical (NACA 0017) airfoil shape. Moreover, the rectangular platform reduced the cost and complexity of the wing structure and tooling while significantly simplifying the integration of the CEETA systems. The testbed is powered by two turbine engines and the CEETA wing has leading-edge and TE blowing with dual-radius flaps and engines mounted in an over-the-wing configuration. A leading edge blowing with a maximum momentum coefficient of blowing of 0.0153 and a TE blowing with a maximum $C_\mu$ of 0.0660 is achieved. The auxiliary power unit APU design borrows from the RC aircraft industry by utilizing
off-the-shelf Electric Ducted Fan (EDF) units to act as axial compressors, which provide the pressure and mass flow needs for the CC implementation.

![Figure 2.12: CEETA testbed demonstrator UAV [35].](image)

2.3 Discussion

Literature review reveals that CC techniques still pose important challenges due to the unfavorable trade offs of mass flow, pitching moment and cruise drag, but these technologies have been around since the early 1970’s and have been successfully demonstrated not only in laboratory environments but also in flight vehicles. On the other hand, only a few UAVs have successfully flown using CC due to the design complexity and the mass flow requirements as Table 2.1 shows. These issues act as roadblocks to real aircraft applications and appear in every CC discussion. They are used as a guide to design, develop and evaluate a CCW capable of achieving high lift augmentation ratios at low blowing coefficients. This dissertation describes a CCW high lift concept that addresses the low mass flow requirement through the use of upper slot blowing. A comprehensive experimental methodology for design, development and testing of CCW-based UAVs with enhanced functionality, and abil-
Table 2.1: Circulation Control - summary of existing experimental and computational research.

<table>
<thead>
<tr>
<th>Aircraft</th>
<th>Author</th>
<th>Airfoil</th>
<th>Experimental</th>
<th>Computational</th>
<th>Momentum Coefficient</th>
<th>Upper/Lower Blowing</th>
<th>Results</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>R. J. Englar [19] 1979</td>
<td>NACA64A008.4</td>
<td>X</td>
<td>0.300</td>
<td>Upper</td>
<td>X</td>
<td>$C_L$ values 2.2 greater than the no-blowing</td>
</tr>
<tr>
<td></td>
<td>J. Abramson [30] 1979</td>
<td>15% thickness</td>
<td>X</td>
<td>0.220</td>
<td>Upper</td>
<td>Produced $C_L$ up to 4.64</td>
<td></td>
</tr>
<tr>
<td></td>
<td>J. K. Hardell et al. [31] 2005</td>
<td>20% thickness, 8.5° cambered elliptical airfoil</td>
<td>X</td>
<td>0.140</td>
<td>Upper</td>
<td>(2-D) Two blowing and (3-D) three blowing $C_L$ than single-slot blowing</td>
<td></td>
</tr>
<tr>
<td></td>
<td>G. S. Jones et al. [32] 2005</td>
<td>2-D GACC</td>
<td>X</td>
<td>0.120</td>
<td>Both</td>
<td>Pulsed blowing reduced the mass flow rate about 50%</td>
<td></td>
</tr>
<tr>
<td></td>
<td>M. G. Alexander et al. [33] 2005</td>
<td>15% thickness</td>
<td>X</td>
<td>0.220</td>
<td>Upper</td>
<td>Produced $C_L$ up to 4.63</td>
<td></td>
</tr>
<tr>
<td></td>
<td>J. K. Harvell et al. [34] 1985</td>
<td>20% thickness, 8.5% cambered elliptical airfoil</td>
<td>X</td>
<td>0.100</td>
<td>Upper</td>
<td>(2-D) Two blowing slots produced higher overall $C_L$ than single-slot blowing</td>
<td></td>
</tr>
<tr>
<td></td>
<td>G. S. Jones et al. [35] 2003</td>
<td>2-D GACC</td>
<td>X</td>
<td>0.120</td>
<td>Both</td>
<td>Pulsed blowing reduced the mass flow rate about 50%</td>
<td></td>
</tr>
<tr>
<td></td>
<td>M. G. Alexander et al. [36] 2005</td>
<td>6% thickness, slightly cambered elliptical</td>
<td>X</td>
<td>0.150</td>
<td>Both</td>
<td>Produced $C_L$ up to 8</td>
<td></td>
</tr>
<tr>
<td></td>
<td>R.C. Swanson et al. [37] 2006</td>
<td>Lockheed airfoil</td>
<td>X</td>
<td>0.100</td>
<td>Upper</td>
<td>Produced $C_L$ up to 4.48</td>
<td></td>
</tr>
<tr>
<td></td>
<td>J. B. Montanya et al. [38] 2007</td>
<td>NASA 64A-008.4</td>
<td>X</td>
<td>0.150</td>
<td>Upper</td>
<td>Produced $C_L$ up to 3.5 with flap deflection</td>
<td></td>
</tr>
<tr>
<td></td>
<td>G. S. Jones et al. [39] 2007</td>
<td>NASA 64A-008.4</td>
<td>X</td>
<td>0.150</td>
<td>Upper</td>
<td>Produced $C_L$ up to 8</td>
<td></td>
</tr>
<tr>
<td></td>
<td>J. K. Harvell et al. [40] 2008</td>
<td>2-D GACC airfoil</td>
<td>X</td>
<td>0.080</td>
<td>Upper</td>
<td>Produced $C_L$ up to 3.5</td>
<td></td>
</tr>
<tr>
<td></td>
<td>J. B. Montanya et al. [41] 2009</td>
<td>2-D 64A-008.4</td>
<td>X</td>
<td>0.100</td>
<td>Upper</td>
<td>Produced $C_L$ up to 3.5</td>
<td></td>
</tr>
<tr>
<td></td>
<td>R. J. Englar et al. [42] 2009</td>
<td>2-D 64A-008.4</td>
<td>X</td>
<td>0.100</td>
<td>Upper</td>
<td>Produced $C_L$ (2-D) up to 4.63</td>
<td></td>
</tr>
<tr>
<td></td>
<td>G. S. Jones et al. [43] 2010</td>
<td>Lockheed airfoil</td>
<td>X</td>
<td>0.150</td>
<td>Both</td>
<td>Pulsed blowing reduced the mass flow rate about 50%</td>
<td></td>
</tr>
<tr>
<td></td>
<td>C. Chen et al. [44] 2011</td>
<td>20% thickness, symmetric</td>
<td>X</td>
<td>0.100</td>
<td>Both</td>
<td>Determined the state of the boundary layer for $C_L$</td>
<td></td>
</tr>
<tr>
<td></td>
<td>R. B. Fischetti et al. [45] 2012</td>
<td>2-D GACC, symmetric</td>
<td>X</td>
<td>0.100</td>
<td>Both</td>
<td>Determined the state of the boundary layer for $C_L$</td>
<td></td>
</tr>
<tr>
<td>Unmanned</td>
<td>G. S. Jones et al. [46] 2013 &amp; W. E. Milholen et al. [47] 2014</td>
<td>Supercritical semi-span wing with flap</td>
<td>X</td>
<td>0.080</td>
<td>Upper</td>
<td>Lift performance improvement by deflection of the flap height</td>
<td></td>
</tr>
<tr>
<td></td>
<td>R. B. Fischetti et al. [48] 2014</td>
<td>Irvine Tutor 40 RC (semi-symmetrical)</td>
<td>X</td>
<td>0.100</td>
<td>Upper</td>
<td>CC technologies successfully integrated in the RC model</td>
<td></td>
</tr>
<tr>
<td></td>
<td>A. Kanistras et al. [49] 2015</td>
<td>Diamond Wing FLAVIIR</td>
<td>X</td>
<td>0.100</td>
<td>Upper</td>
<td>CC is successfully used for Roll Angle control</td>
<td></td>
</tr>
<tr>
<td></td>
<td>L. W. Ho et al. [50] 2016</td>
<td>S8036 airfoil</td>
<td>X</td>
<td>0.040</td>
<td>Upper</td>
<td>Lift performance improvement at low blowing coefficients (3-D) $\Delta C_L = 0.42$, $\alpha = 13^{o}$</td>
<td></td>
</tr>
<tr>
<td></td>
<td>K. Kanistras et al. [51] 2017</td>
<td>S8036NACA 2412NACA 0015NACA 23015</td>
<td>X</td>
<td>0.011</td>
<td>Upper</td>
<td>Comparative study of five airfoils Highest $\Delta C_L = 0.89$ recorded at $\alpha = 30^{o}$</td>
<td></td>
</tr>
<tr>
<td></td>
<td>K. Kanistras et al. [52] 2017</td>
<td>Clark-Y airfoil</td>
<td>X</td>
<td>0.011</td>
<td>Upper</td>
<td>Comparative study of five airfoils Highest $\Delta C_L = 0.89$ recorded at $\alpha = 30^{o}$</td>
<td></td>
</tr>
<tr>
<td></td>
<td>K. Kanistras et al. [53] 2018</td>
<td>S8036NACA 2412NACA 0015NACA 23015NACA 0015</td>
<td>X</td>
<td>0.011</td>
<td>Upper</td>
<td>Comparative study of five airfoils Highest $\Delta C_L = 0.89$ recorded at $\alpha = 30^{o}$</td>
<td></td>
</tr>
</tbody>
</table>
ity to execute complex missions with different payload requirements and on-board sensor suite flexibility is presented in detail.
Chapter 3

Fundamentals of Circulation Control

The main purpose of this chapter is to introduce the mathematical framework of the necessary fundamental principles and to formulate a set of equations that will be used to derive the aerodynamic parameters that will be used to evaluate the efficiency of CC. Terms such as vortex, vorticity, circulation and Coandă effect are defined and explained as they play an important role in CC. The Kutta-Joukowski, Kelvin’s and Helmholtz’s theorems and the importance of the aerodynamic coefficients are discussed. In addition, the Coandă effect and its application in aerodynamics is also explained in this chapter.

3.1 Vorticity & Circulation

Active flow control methods continue to be a promising research field that can enhance the aerodynamic performance of not only conventional aircraft but also
small-scale fixed-wing UAVs. By the end of the nineteenth century, the theory of ideal, or potential (irrotational velocity field) flow was well developed, but later it was recognized that, for practical applications in aerodynamics, much attention was required to successfully apply potential-flow theory [48]. Real flows have a tendency to separate from the surface of the body and especially on bluff bodies (airfoils with rounded or near-rounded trailing edges). Also, steady potential-flow around a body can produce no force irrespective of the body’s shape. This result is usually known as d’Alembert’s paradox [48]. There is no prospect of using potential-flow theory in its pure form to estimate the lift or drag of wings and, thereby, develop aerodynamic design methods. However, potential-flow can be adapted to provide a reasonable theoretical model for the flow around an airfoil that generates lift. It is assumed that a flow field is inviscid and irrotational everywhere, and possesses no vorticity, which translates to zero circulation condition in the flow, and hence no net force acting on the body or the flow. Potential-flow assumes the presence of a velocity potential in the flow, which helps determine the velocity at a point in the flow or at the surface and since there is no viscosity, there is no surface friction [49].

The generation of lift is always associated with circulation and vorticity, which are the two primary measures of rotation in a fluid. The total amount of vorticity passing through any plane region within a flow field is called circulation, denoted by $\Gamma$ in Figure 3.1. Circulation does not necessarily mean that the fluid elements are moving around an object in circles. As Figure 3.1 shows, circulation simply means that if the airfoil generates lift, then the integral of the equation that is shown in Figure 3.1 will be finite. Vorticity, however, is a vector field that gives a microscopic measure of the rotation at any point in the fluid. Potential-flow with the Kutta condition (a body with a sharp trailing edge, which is moving through a fluid will
create about itself a circulation of sufficient strength to hold the rear stagnation point at the trailing edge) can help determine the lift force produced, but still does not include the effects of viscosity found in all real flows. Thus, only drag produced due to pressure distribution can be calculated.

According to the Kutta-Joukowski theorem, a finite \( \Gamma \) is equal to the strength of the vortex, which is responsible for the generated lift. In other words, circulation theory of lift is an alternative way of thinking about the generation of lift on an aerodynamic body [50]. The momentum of the jet delays separation and moves the rear stagnation point, increasing circulation \( \Gamma \), which is defined as the line integral of the tangential velocity around a closed contour \( C \) (Figure 3.1).

![Figure 3.1: Left: Definition of circulation. Right: Circulation around a lifting airfoil [50].](image)

If in some way it is possible to use vortices to generate circulation, and thereby lift, for the flow around an airfoil, the results can be seen schematically in Figure 3.2. In Figure 3.2 (a), the pure non-circulatory potential-flow is around an airfoil at an angle of incidence and it can be seen, as circulation is added, the fore (SF) and aft (SA) stagnation points move downward. Figure 3.2 (d) shows the aft stagnation point (SA) to be located at the TE, which is the case that the Kutta condition is satisfied.
3.1.1 Kevin’s & Helmholtz’s Vortex Theorems

Vorticity is a vector field that is twice the angular velocity of a fluid particle. Flows in circular paths are called vortex flows and a vortex line is a curve in the fluid that is everywhere tangent to the local vorticity vector and is related to the vorticity vector the same way a streamline is related to a velocity vector. In a region of flow with nontrivial vorticity, the vortex lines drawn through each point of a closed curve constitute the surface of a vortex tube [51]. Lord Kelvin introduced the idea of circulation and proved the following theorem: *In an inviscid, barotropic flow with conservative body forces, the circulation around a closed curve moving with the fluid remains constant with time*, if the motion is observed from a non-rotating frame. The proof of Kelvin’s theorem can be found in [51]. Kelvin’s theorem implies that...
if four restrictions are satisfied, the irrotational flow will remain irrotational. The four restrictions are: i) there are no net viscous forces along C (Figure 3.3); ii) the body forces are conservative; iii) the fluid density must depend on pressure only (barotropic flow); iv) the frame of reference must be an inertial frame.

Figure 3.3: Contour geometry for Kelvin’s circulation theorem.

However, under the same four restrictions, Helmholtz proved the four theorems for vortex motion, which state:

- Vortex lines move with the fluid.
- The strength of a vortex tube (its circulation) is constant along its length.
- A vortex tube cannot end within the fluid. It must either end at a solid boundary or form a closed-loop (vortex ring or loop).
- The strength of a vortex tube remains constant in time.

### 3.1.2 Finite Wing Theory

2-D airfoil data could be used to predict the aerodynamic characteristics of 3-D wings, provided the aspect ratio is large and the assumptions of thin-airfoil theory are
met [48]. When a wing of any shape is accelerated from rest, the circulation around it, and therefore the lift (Kutta-Joukowski theorem, Equation (3.1)), is not produced instantaneously. Due to the sharpness of the TE and the high speeds that occur and the high local accelerations, the air is unable to turn around the TE and leaves the surface producing a vortex just above the TE. The stagnation point moves towards the TE, the circulation around the wing and the lift increases progressively as the stagnation point moves back [48]. From Helmholtz’s second theorem (Section 3.1.1) the strength of circulation is the sum of the strengths of the vortex filaments cut by the section plane. At any section the lift per unit span ($L'$) is given by Equation (3.1) and for a given flight speed and air density, circulation ($\Gamma$) is proportional to lift.

$$L' = \rho V \Gamma \quad (3.1)$$

The lift (3.2) and drag (3.3) coefficients are dimensionless coefficients that relate the lift and drag forces that are generated by a lifting body (fixed wing aircraft). The lift coefficient is a number that describes all the complex dependencies of shape, inclination, and flow conditions on lift. Equations (3.2) and (3.3) are used to calculate lift and drag forces that are applied on the wind tunnel models during testing. The relationship between the lift coefficient and the angle of attack and between the lift coefficient and drag coefficient is characteristic for each airfoil.

$$C_L = \frac{L}{q_\infty S} \quad (3.2)$$

$$C_D = \frac{D}{q_\infty S} \quad (3.3)$$
where $q_\infty$ is the dynamic pressure defined as:

$$q_\infty = \frac{1}{2} \rho V^2$$  \hspace{1cm} (3.4)

### 3.2 Coandă Effect

Three different phenomena are associated with the name Coandă [52]. The first, is the tendency of a fluid jet approaching a curved surface to remain attached to that surface. The effect is commonly seen in natural phenomena such as a stream of water falling onto the convex side of a spoon. The second, is the ability of a fluid jet to attach itself to a nearby surface and the third, which is used in aerodynamics, is the tendency of jet flows over convex curved surfaces to entrain ambient fluid and increase more rapidly than that of plane wall jets. The Coandă effect, in aerodynamics, is used to generate higher lift coefficients by blowing air close to the TE of the wing [53]. The jet that comes out of the slot, which is located at the TE of the wing, remains attached further along the curved surface of the wing and moves the separation point around the TE toward the lower surface of the wing resulting in lift augmentation (Figure 3.4). The Coandă effect was discovered accidentally by Henry Coandă in 1935 and since then, different uses of it have been investigated in aeronautics [15].

![Tangential blowing over a Coandă surface](image)

Figure 3.4: Tangential blowing over a Coandă surface [54].
3.2.1 2-D Analysis

Newman [55] investigated a 2-D, incompressible, turbulent jet flowing around a circular cylinder (Figure 3.5) and showed that the *Coandă* adhesion effect is a direct consequence of the balance of the forces applied on the fluid. As the jet exits the slot, the contact pressure with the curved wall is lower than ambient pressure because of the presence of viscous drag phenomena generated by the interaction of the fluid and the curved wall.

![Figure 3.5: Newman experimental setup [55].](image)

The *Coandă* effect, as Figure 3.5 shows, can be described by parameters such as the angle of separation ($\theta_{sep}$), the slot width (b), the radius of curvature (a), the Reynolds number (Re) and pressure differential ($p_s - p_\infty$) where $p_s$ is the supply pressure.

The angle of separation, Equation (3.5), which describes the flow along a cylinder is proposed by Newman [55] and is a function of the pressure differential, geometrical and fluid properties.
\[ \theta_{sep} = f \left[ \left( \frac{(p_0 - p_\infty) \cdot b \cdot a}{\rho \cdot \nu^2} \right)^{0.5} \right] \quad (3.5) \]

Experimental analysis was conducted by Newman [55] to investigate the angle of separation for 2-D real fluids at Reynolds numbers greater than \( 4 \times 10^4 \) and small slot width to radius ratios, \( b/a \).

Equation (3.6) is obtained experimentally from the detachment angle for a non-compressible fluid. A detachment angle is relatively constant near 240° downstream and that result was later confirmed by other researchers [56, 57].

\[ \theta_{sep} = 245 - 391 \cdot \frac{b}{a} \cdot \left( \frac{b}{a} \right) \quad (3.6) \]

### 3.2.2 Coandă Jet Circulation Control

When the Coandă effect is used as a flow control technique (Coandă jet) for enhanced aerodynamic performance (lift enhancement) the mass flow rate \( \dot{m} \) and the momentum coefficient of blowing \( C_\mu \) are the parameters that need to investigated other than the wing geometry. The momentum coefficient \( C_\mu \), which is the ratio of jet and freestream momentum, is a critical parameter in understanding the efficiency of blowing in CC, the actual momentum coefficient at the jet is given by Equation (3.7).

\[ C_\mu = \frac{Thrust}{q_\infty S} = \frac{\dot{m}_{jet} V_{jet}}{q_\infty S} \quad (3.7) \]

For airfoil flapless surface blowing with a circular or near circular TE, the TE jet can be directed downward, as a jetflap. The force on this body is given by Equation (3.8).
\[ F = - \int_C = Pndc - \dot{m}_p(V_{jet} - V_\infty) \] (3.8)

The total force on the body, which is the modified Kutta-Joukowski theorem for an airfoil with additional blowing is given by Equation (3.9). The addition of surface blowing at the TE modifies the initial circulation (generated by the airfoil), and produces a net thrust [58].

\[ F = (\rho_\infty V_\infty \Gamma_{mod})\mathbf{k} + \rho_\infty V_\infty (V_{jet}ysin\delta)\mathbf{k} - \dot{m}_p(V_{jet} - V_\infty)\mathbf{i} \] (3.9)

Another important parameter used in the analysis of CC is the lift augmentation ratio ($\Delta C_L/C_\mu$), which is the ratio of the change in lift coefficient to the momentum coefficient. In theory, good circulation airfoils should achieve augmentation ratios between 50 to 70, when jet flaps attain augmentation ratios of approximately 14 [23]. There has been significant research done [21, 27, 41] on the performance of CC-airfoils and Englar tested a 15% CC ellipse with a rounded trailing edge and obtained a section lift coefficient of 4.3 for a $C_\mu$ of 0.2. Depending on the dimensions of the airfoil and other parameters (angle of attack and $C_\mu$), lift augmentation ratios higher than 80 can be achieved. Typically the highest lift augmentation ratios occur at momentum coefficients less than 0.03, where super-circulation is not yet achieved [59].
Chapter 4

Model Description

Five semi-span CCWs with zero leading and trailing edge sweep and no winglets are chosen to investigate the effect of blowing on different configurations. Details pertaining to the airfoil shape, slot characteristics and Coandă surface geometry are presented next.

4.1 Coandă Surfaces

As Known, “A body with a sharp TE which is moving through a fluid will create about itself a circulation of sufficient strength to hold the rear stagnation point at the TE.” - this is the Kutta condition [60].

Given an airfoil with a sharp TE, the Kutta condition refers to the flow pattern in which fluid approaches the corner from both directions, meets at the corner and then flows away from the body as Figure 4.1 shows. However, in CC, the wing and flap geometry need to be investigated. At first, the airfoil shape and the Coandă surface (TE radius) that gives the maximum lift augmentation need to be investigated. The
drag penalties are expected to be high due to the blunt (round) TE but at this point, focus is given to lift augmentation.

Figure 4.1: Flow near a TE [48].

The experimental framework is inspired by a wind tunnel experiment conducted on a 6% thick camber elliptical CC airfoil with both upper and lower blowing at the NASA Langley Research Center [23]. However, no influence on lift with lower slot blowing subsonically was reported, thus, it was decided to apply upper blowing and test Coandă surfaces with bigger length-to-height ratios in order to investigate the influence of the curvature on upper blowing cases. Four Coandă surfaces are designed and built with length-to-height ratios of (1:1), (2:1), (3:1), (4:1), see Figure 4.2.

Figure 4.2: Removable Coandă surfaces with length-to-height ratios of (1:1), (2:1), (3:1), (4:1).

The major and minor axes of each Coandă surface are positioned in a way such that the slot is fixed at $x_{/crf} = 0.8644$ and the slot height is fixed and equal to $h = 0.7\text{mm}$. The full chord length of the model with the Cs (Coandă surfaces) attached and the total wing area are shown in Table 4.1, which also shows the radius
Table 4.1: CCW chord lengths and wing area.

<table>
<thead>
<tr>
<th>Cs</th>
<th>c (m)</th>
<th>S (m²)</th>
<th>r (mm)</th>
<th>r/c</th>
<th>h/c</th>
<th>h/r</th>
</tr>
</thead>
<tbody>
<tr>
<td>1 : 1</td>
<td>0.1325</td>
<td>0.0397</td>
<td>2.8</td>
<td>0.0211</td>
<td>0.0053</td>
<td>0.2500</td>
</tr>
<tr>
<td>2 : 1</td>
<td>0.1353</td>
<td>0.0406</td>
<td>5.6</td>
<td>0.0414</td>
<td>0.0052</td>
<td>0.1250</td>
</tr>
<tr>
<td>3 : 1</td>
<td>0.1381</td>
<td>0.0414</td>
<td>8.4</td>
<td>0.0610</td>
<td>0.0051</td>
<td>0.0830</td>
</tr>
<tr>
<td>4 : 1</td>
<td>0.1409</td>
<td>0.0422</td>
<td>11.2</td>
<td>0.0795</td>
<td>0.0050</td>
<td>0.0625</td>
</tr>
</tbody>
</table>

of curvature of the Coandă surfaces and the radius-to-chord length, the slot height-to-chord length and the slot height-to-radius ratios, respectively.

4.1.1 Airfoil Shapes

Symmetrical and non-symmetrical airfoil shapes are chosen. All airfoils are commonly used on Remote Control (RC) fixed-wing aircraft models.

S8036 airfoil

The S8036 is chosen because it exhibits good behavior at low Reynolds numbers and has similar thickness to the NACA0015 and Clark-Y airfoils. The S8036 airfoil has a thickness of 16% and a gentle stall [61]. The CCW (Figure 4.3) is designed with an aspect ratio of $AR = 2$. The chord length of the wing before the design modification is $c_{\text{ref}} = 150\text{mm}$ and the span is $b = 300\text{ mm}$. After the wing modification (the chord reduction and the removable Coandă surfaces) the aspect ratio increases to $AR = 2.3$. 
NACA0015 airfoil

The NACA0015 airfoil is selected because it is well-studied [62, 63, 64] in both 2-D and 3-D and it has approximately the same thickness as the other airfoils that were tested but with a zero camber. It is a symmetrical airfoil with a 15% thickness. This airfoil belongs to the 4-digit NACA series which do not usually have applications on commercial aircraft due to the low maximum lift coefficient. However, it shows good stall characteristics. The thinner NACA airfoils are popular for V-tails, horizontal stabilizers, fins and rudders and the thicker NACA0015 is a popular wing airfoil for aerobatic and sport aircraft.

NACA2412 airfoil

The NACA2412 is selected because it is a semi-symmetrical airfoil, which has a 2% camber and its point of maximum camber is located at its 40% chord point. Its maximum thickness is 12% and it is also well-studied and commonly used in RC
aircraft. It is stable with a high stall angle and has similarities with the other chosen airfoils.

**NACA23015 airfoil**

The NACA23015 is chosen because it is well-studied and the most commonly used in RC aircraft and has been a popular choice for general-aviation applications for many years [65]. It has similarities with the other airfoils and it performs well at low Reynolds numbers. It is a 5-digit airfoil, which has a maximum thickness of 15%. It shows a higher $C_{l_{max}}$ relative to NACA0015, but this advantage is lost at high Mach numbers.

**Clark-Y airfoil**

The Clark-Y (smoothed) is not only widely used in RC airplanes but also in general purpose aircraft designs. Due to its high lift-to-drag ratios and high stall angle it is popular in model aircraft. This airfoil has a thickness of 11.7% and is flat on the lower surface from 30% of the chord back. CCW configurations to achieve high lift coefficients with low drag forces have been investigated on a Clark-Y airfoil [45], where a 3D wing with an aspect ratio of AR = 2 is built and tested in a low speed wind tunnel. Similar tests are repeated here, but at lower moment coefficients of blowing and with a different testing apparatus. A non-modified Clark-Y wing is used as a calibration wing with an aspect ratio of AR = 2 in order to validate the force balance sensor. The five CCWs and the Clark-Y (calibration wing) are presented in Figure 4.4.
The results, which can be found in Chapter 7, indicate that upper slot blowing shows sufficient lift enhancement for all tested configurations and the CCW with the NACA 0015 symmetrical airfoil configuration with the (2:1) Coandă surface, gives the highest lift enhancement.

### 4.1.2 Dual Radius Flap Geometry

In an attempt to improve the aircraft performance during cruise flight and provide insight into the aerodynamic characteristics of the geometric parameters of the dual radius CC flaps, two dual radius flaps were developed by varying specific flap parameters. The design parameters that define the efficacy of CC are known to be: the slot height, the slot location, the 1st Coandă radius ($r_1$) and the 2nd Coandă radius ($r_2$) along with the flap design [21, 66, 6, 19, 22, 27, 39, 41]. The slot height, the slot location and the 1st Coandă radius ($r_1$) are determined from previous research [45, 46] based on the region of most effective Coandă operation, which is represented by the yellow region [66] in Figure 4.5. The chosen baseline parameters are shown in Table 4.2.
The Coandă radius \( r_1 \) of the dual radius flap is a constraint of primary importance since it defines the slot location. The upper and lower surface of the airfoil (Figure 4.6) are intersected by a line, which is constrained in dimension equal to the summation of the \( r_1 \) and the slot height (\( h \)). The slot is placed on the intersection of the upper and lower surface of the airfoil. The upper intersection of the airfoil is, then, used to build a tangent to the upper surface at the slot. A projection parallel to the tangent line is also constrained in a way that it passes through the slot. The projection is set tangential to \( r_1 \) at the slot exit as Figure 4.6 (Step 4) depicts. The projection set to be tangential to the slot and the center of the radius \( r_2 \) is constrained to lie on the normal to the projection. Next, a line from the lower surface parallel to the chord is designed and intersects with the secondary radius, closing the contour of the flap. Table 4.3 shows the dual radius flap parameters of the DRF10.
(Dual Radius Flap) and DRF45 designed flaps. Figure 4.6 shows the dual radius flap design process.

Table 4.3: Dual radius flap design parameters.

<table>
<thead>
<tr>
<th>Configuration</th>
<th>( \frac{c'}{c} )</th>
<th>( \frac{c_f}{c'} )</th>
<th>( r_1 )</th>
<th>( r_2/r_1 )</th>
</tr>
</thead>
<tbody>
<tr>
<td>DRF10</td>
<td>1.0003</td>
<td>0.1303</td>
<td>9</td>
<td>10</td>
</tr>
<tr>
<td>DRF45</td>
<td>1.0643</td>
<td>0.2026</td>
<td>9</td>
<td>45</td>
</tr>
</tbody>
</table>

Figure 4.6: Dual radius flap design methodology.

Figure 4.7 shows the 3-D printed dual radius flaps at all deflection angles that are designed, built and tested. The design of the flaps allows easy installation on the 3-D printed wing before the wind tunnel test is conducted.
4.1.3 CCW Wind Tunnel Model

Since the results from the Coandă surface investigation indicate that the CCW (NACA 0015 the symmetrical airfoil configuration) with the (2:1) Coandă surface, gives the highest lift enhancement, it is decided to proceed to wind tunnel testing using this configuration.

A modified NACA 0015 airfoil shape CCW shown in Figure 8.8 with zero leading and TE sweep and no winglets is designed and built. The selection of the airfoil profile NACA 0015, is largely driven by the good CCW characteristics for upper slot TE blowing cases [46, 47]. It gives the highest lift enhancement compared to other airfoils and high augmentation ratios with the 2:1 Coandă surface configuration. The wing model is rapid prototyped out of acrylonitrile-butadiene-styrene (ABS) plastic and the surface is covered with tape to give a smoother finish. A tangential blowing slot is located at the 88% chord location on the upper TE surface, and is directed across the span of the wing. The span is $b = 400$ mm with a slot length of 390 mm and a measured average height of $h = 0.4$ mm. The wing is mounted vertically on the external force balance (Figure 4.8) and has an aspect ratio of 1.7.
The results, which are summarized in Chapter 7, show that the smaller Coandă radius ratio \((r_2/r_1)\) flap, which is tested at 0°, 30° and 60° deflection, is found to be the most efficient at CC blowing with a maximum incremental lift coefficient \((\Delta C_l)\) of 0.89 at 30° flap deflection. The same configuration gives high lift-to-drag ratios at 0° deflection and at 0° angle of attack and it performs better in terms of high-lift enhancement at takeoff scenarios where the deflection of the flap non-zero. The plenum geometry and experimental evaluation is described next.
Chapter 5

Plenum Design

Although CC has been widely studied, most research has concentrated on the design and analysis of the TE and the wing design itself, and less on the design of a plenum that distributes the air evenly across the span. CCWs, which are considered to be the most effective alternative to conventional high-lift systems, are designed to increase the lifting force of an aircraft mainly during take-off and landing when large lifting forces at low speeds are required. An important factor for the efficiency of CC is the flow uniformity at the slot [67]. The approach required to achieve uniform flow for CCWs is explained in detail, and a design that can distribute air evenly across the span is presented.

Plenum designs have been investigated in the past, and different techniques to achieve flow uniformity at the slot-exit along the span have been tested [23, 67]. This chapter presents the methodology followed to achieve a plenum design capable of distributing the flow equally at the slot of a CCW. The SolidWorks designs and the ANSYS CFX results are presented first. The chapter concludes with the design configurations that achieved flow uniformity.
5.1 Plenum Geometry

5.1.1 Preliminary Plenum Design

At first, the plenum consisted of internal tubes to distribute the air at the slot of the CCW. The CCWs consist of two parts: the lower section with the ribs attached, and the upper section, which is the cover of the wing. The lower section is divided into four areas each of which has a separate tube connected to the main air supply tube providing the air in the plenum. Two different plenum designs were tested in order to investigate the air distribution along the span, and the effect on aerodynamic forces on a CCW. Figure 5.1 shows the plenum design that was used on the S8036 and NACA0015 CCWs.

Figure 5.1: Internal plenum of the S8036 CCW: The lower part of the model is divided into four areas which have individual tubes, connected with the main flexible high-pressure tube that provides the air in the plenum.

The performance of the plenum design that is used for the NACA 2412, NACA 23015 and Clark-Y CCWs is shown in Figure 5.2. The velocity at the jet, $V_{\text{jet}}$, is measured using a hot-wire anemometer placed in different positions along the slot exit. The pattern shown in Figure 5.2 is the same for all mass flow rates. As the
mass flow rate increases, the average jet velocity increases linearly. Figure 5.3 depicts a schematic of the plenum design and a picture of the plenum design of the Clark-Y CCW. The pattern shows that more air is coming out of the jet at the middle of the span which is expected due to the position of the tubes inside the plenum. However, no influence of the performance of the velocity at the jet on the aerodynamic forces is noticed.

![Graph showing V_{jet} performance at the slot exit and average V_{jet} behavior.](image)

Figure 5.2: Left: $V_{jet}$ performance at the slot exit. Right: Average $V_{jet}$ behavior.

![Schematic of the plenum design in NACA 2412, NACA 23015 and Clark-Y.](image)

Figure 5.3: Left: Top view schematic of the plenum design in the NACA 2412, NACA 23015 and Clark-Y. Right: Clark-Y plenum design.

The momentum coefficients of blowing ($C_\mu$) are found to be low compared to those reported in the literature obtained using compressed air systems. This is likely due to the complexity of the plenum and the lack of space inside the printed CCWs.
The limited space does not allow for bigger high-pressure pipes and in combination
with the limited compressed air supply it is unlikely to generate higher moment
coefficients. Since flow uniformity is not observed in this design, focus is given on
diffuser-based designs.

5.1.2 Diffuser-Based Plenum Design

After reviewing previous studies in plenum designs for CCWs [24, 67, 68, 69] and
due to space limitations inside the wing area, it was decided to use a diffuser design
in order to distribute the air across the span. The objective of a diffuser, which is
basically an expanding duct, is to recover static pressure from a fluid (in this case air)
stream while reducing the flow velocity [70]. The challenges to use a diffuser-based
design are: (i) the speed of air needs to be accelerated and not decelerated inside
the plenum, since the $V_{jet}$ must be high, and (ii) the walls of the diffuser need to
be designed such that they do not reduce the diffuser’s performance. According to
[70], a wide angle diffuser can improve its performance if vanes are installed. The
vanes divide the diffuser into a series of sub-diffusing passages with area ratios and
divergence angles smaller than the initial diffuser. That way, stall is avoided and
each passage can operate at near optimum pressure recovery.

Details of the CAD design geometry and the Computational Fluid Dynamics
(CFD) analysis along with the results for all tested designs are presented here. The
plenum designs that are tested are part of a NACA 0015 CCW. The inlet and outlet
gameometry characteristics are kept constant. For the inlet, the inner diameter tube is
kept to $d_{inner} = 10$ mm and the slot-exit is $h = 1$ mm height. The span of the plenum
is $S = 150$ mm.
1st Generation of Diffuser-Based Plenum Design

Several designs were considered and simulations were conducted to identify the design that can provide uniform flow with minimum losses. The CAD designs and the simulation results of all the designs tested that have been considered for flow uniformity, can be found in Appendix B. The results of the qualified plenum design are presented here. SolidWorks is used for the geometric design of all the plenum designs before the CFD analysis and Figure 5.4 shows the design.

![CAD Design and Simulation Results]

**Figure 5.4:** Left: Trimetric and side view of the CAD Design. Right: Simulation Results of the Design.

The plenum is simulated for four different inlet velocities. Those different velocities correspond to four different RPM values that an Air Supply Unit (ASU) achieves. A centrifugal compressor (ASU) is used to supply air to the plenum and the inlet velocities that correspond to 7000, 12500, 17500 and 21500 RPMs are 17.2 m/s, 31.1 m/s, 41.7 m/s and 53.6 m/s respectively.

Since the CFD results show flow uniformity across the span, the next step was to 3-D print the design and experimentally test it. As Figure 5.5 demonstrates, the setup that is used to measure the velocities at the jet ($V_{jet}$) across the span uses a pitot tube (which can freely be moved across the span while retaining the same
height and distance from the slot), for different inlet velocities. Measurements are recorded at 14 points spaced evenly across the span.

Figure 5.5: Experimental setup for $V_{\text{jet}}$ measurements.

![Figure 5.5: Experimental setup for $V_{\text{jet}}$ measurements.](image)

![Figure 5.6: Experimental and simulation results representing the performance of the $V_{\text{jet}}$ at the slot across the span.](image)

Figure 5.6: Experimental and simulation results representing the performance of the $V_{\text{jet}}$ at the slot across the span.

Figure 5.6 shows the comparison between the computational and the experimental results of flow uniformity across the span for different input velocities. The second plot shows the linear relation between the average $V_{\text{jet}}$ and the RPMs at which the ASU operates.
2\textsuperscript{nd} Generation of Diffuser-Based Plenum Design

Since the first diffuser based design performs well and provides flow uniformity across the span, further optimization of the design (minimize losses and stall conditions that occur at high Reynolds numbers) and integrating it on the wing for wind tunnel testing is followed.

Lift is a mechanical force and it is generated by the interaction and contact of the aircraft’s body and mainly the wings, with air. Thus, to take advantage of the total area of the wing, a slot along the total span of the wing is required for lift enhancement. Since experimental results show a good match with the simulation results, it was decided to scale up the plenum design and determine if a larger scale of that plenum design can still provide flow uniformity across the span.

The next design that is tested is scaled up to a span of 400 mm. The air is controlled by an air pressure regulator delivering a maximum of 400 KPa. The results show uniformity with small variations from the mean (black line) as Figure 5.7 depicts. The slot height is 0.4 mm and the Coandă surface has a radius of 3.6 mm. The results show high flow uniformity performance but uniformity is reduced at high speeds. That effect is caused mainly for two reasons: i) stall effects are introduced due to the vanes and; ii) due to the fact that the vanes are curved and Coandă effects are introduced.
Figure 5.7: Left: CAD design of the plenum. Right: Simulation results representing the performance of the $V_{\text{jet}}$ at the slot across the span.

### 3rd Generation of Diffuser-Based Plenum Design

A vaned straight-walled wide angle diffuser with a nozzle at the exit to provide flow uniformity across the span of the wing is used. The diffuser is divided using nine equally distanced vanes as Figure 5.8 demonstrates. The skin of the wing above the slot has a thickness of 0.35 mm and without any internal support, the slot will deform and increase during blowing. To prevent this, nine equally distanced stationary aerodynamic standoffs are designed into the aft plenum along the span to maintain a known slot height. Before the air is driven through the diffuser, it passes through an elbow tube fitting, which results in a non-uniform flow across the outlet of the tube. To correct the direction of the flow, three vanes are placed inside the tube and Ansys Fluent CFD analysis is conducted to find the exact position that the vanes should be placed. The design that showed the best result is then 3-D printed and attached to the plenum as Figure 5.8 illustrates.
Figure 5.8: Plenum CAD design: The vanes at the inlet for flow correction and the vanes in the diffuser to achieve uniformity are shown. Nine standoffs are placed to avoid slot deformation during blowing.

For flow uniformity testing, the wing is placed inside the wind tunnel and an experimental setup using a stepper motor and a calibrated pitot probe is used to measure the velocities at the jet ($V_{jet}$) across the span (the pitot tube can move freely with equal increments across the span while retaining the same height and distance from the slot). Measurements are recorded at 20 points and the process is repeated 6 times for each inlet velocity. The average of the 6 runs for each inlet velocity along with the mean velocity (lines in black) at each case are shown in Figure 5.9. Flow uniformity is tested at 5 different inlet pressures 25kPa, 50kPa, 75kPa, 100kPa and 200kPa. The maximum inlet pressure to the air system that is used for testing is 110kPa but it is decided to test the efficiency of uniformity at higher velocities as well. The results showed that even at high velocities the plenum responds well and despite minimum losses, flow uniformity is achieved.
Figure 5.9: Flow uniformity performance across the span of the wing.

The results (Figure 5.9) show that flow uniformity is achieved at all tested cases. The last stage of optimization is conducted when the Air Delivery System (ADS) and the ASU is implemented and tested as the CCW system. At this stage, it was decided to proceed with this plenum design since the results are sufficient.
Chapter 6

Wind Tunnel & Instrumentation

Experimental information useful for solving aerodynamic problems can be obtained in a number of ways but in this chapter, the use of low-speed wind tunnels is considered. Wind tunnels are often the most timely, economical and accurate means for conducting aerodynamic research and obtaining aerodynamic data to support design decisions because they can provide large amounts of reliable data [71]. The measurement and control applications for wind tunnel testing typically include the wind speed measurement and model balance and/or control with respect to the wind in the tunnel. This chapter presents the implementation aspects of a reliable wind tunnel and the required instrumentation. The instrumentation that is described in this chapter (force balance and pitot probe), is designed, built and calibrated in University of Denver facilities.
6.1 Wind Tunnel

The wind tunnel used in these experiments (Figure 6.1) is a fan-driven, open-loop circuit, continuous-flow tunnel with a circular test section of 35.5 in (0.9 m) diameter. The tunnel can operate at speeds up to 14 m/sec (equivalent to a Mach number of 0.04).

Figure 6.1: Sketch of the wind tunnel.

The calibration of the wind tunnel is an important initial step in the process. The wind tunnel is initially powered and allowed to reach operating conditions. The atmospheric pressure, air density and temperature are measured and recorded. Then the airspeed in the wind tunnel is increased and measurements at various points close to the test section are recorded using hot wire anemometry. The measurements are repeated 3 times for each point to minimize the error. Table 6.1 shows the free stream velocity data recorded at the cross section of the test section. The hot wire is placed in the test section of the wind tunnel and the data are recorded with increments of 2 in along the x and y axes, as Figure 6.2 demonstrates.
Figure 6.2: Illustration of the test section of the wind tunnel. The red box depicts the location of the wing model.

Table 6.1: Free stream velocity at various points in a cross section of the test section.

<table>
<thead>
<tr>
<th>$x,(\text{in})$</th>
<th>-12</th>
<th>-10</th>
<th>-8</th>
<th>-6</th>
<th>-4</th>
<th>-2</th>
<th>0</th>
<th>2</th>
<th>4</th>
<th>6</th>
<th>8</th>
<th>10</th>
<th>12</th>
</tr>
</thead>
<tbody>
<tr>
<td>$V_{\infty,x},(\text{m/s})$</td>
<td>7.8</td>
<td>7.7</td>
<td>7.7</td>
<td>7.7</td>
<td>7.6</td>
<td>7.7</td>
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<td>7.7</td>
<td>7.8</td>
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<td>$y,(\text{in})$</td>
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<td>-6</td>
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<td>2</td>
<td>4</td>
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<td>8</td>
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</tr>
<tr>
<td>$V_{\infty,y},(\text{m/s})$</td>
<td>7.8</td>
<td>7.7</td>
<td>7.7</td>
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<td>7.6</td>
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<td>7.8</td>
<td>7.7</td>
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<td>7.7</td>
<td>7.7</td>
<td>7.7</td>
<td>7.8</td>
</tr>
</tbody>
</table>

6.2 Force Balance

Measurement of steady and fluctuating forces acting on a body (airfoil) in a flow is one of the main tasks in wind tunnel experiments. In aerodynamic testing, strain gauge balances are typically used for this task as however, balances based on piezoelectric multicomponent force transducers are a recommended alternative solution. This section provides information about the design, calibration and implementation of the force balance that is used in wind tunnel testing.
6.2.1 The Concept

In the wind tunnel, measurements of the aerodynamic loads acting on the test model are made using an internal/external strain-gauge balance. Balances can be designed to measure up to six components of the loads. The loads are a combination of the aerodynamic loads (Lift, Drag, etc.), model weight, and a portion of the weight of the balance itself. A balance measures the loads by using strain-gauges, arranged in a Wheatstone. For the purpose of this experimental work, a full Wheatstone bridge setup (Figure 6.3) is used as this configuration is more sensitive compared to half or quarter bridge setup and because the relationship between strain and voltage is linear while the others are not.

![Full-bridge strain gauge circuit](image)

Figure 6.3: Full-bridge strain gauge circuit [72].

With a full-bridge, the output voltage is directly proportional to applied force, with no approximation (provided that the change in resistance caused by the applied force is equal for all four strain gauges) and this is the main reason why the installation of the gauges is important [73]. Figure 6.4 depicts the full bridge setup that is used on the beam which is part of the force balance design.
The output voltage of a balanced bridge changes as a function of the strain at the bridge location produced by the applied loads. In order to convert the output voltage into a load, the balance must be calibrated. The balance is calibrated by applying known loads to the balance and recording the output of the various bridges. A numerical relationship, or calibration matrix, is then determined between the applied loads and the voltages (readings) from the balance bridges. The calibration of the balance is extremely important in the use of a strain-gauge balance. The measured loads can never be more accurate than the accuracy of the calibration.

6.2.2 Setup

For the purpose of accurately measuring the forces acting on the airfoil, a two-component force balance was built. It has 2-degrees of freedom with two measurement channels capable of simultaneously measuring lift and drag forces acting on the wing. Due to the wing geometry and design restrictions (plenum design and compressed air supply) an external force balance configuration was chosen. A stepper motor is mounted to turn a rotating plate, thereby allowing measurement at different angles of attack, shown in Figure 6.5. Two load cells are placed on the bottom of the square support beam that holds the wing, creating two full-Wheatstone bridge
configurations for lift and drag measurements. A motor driver controls and rotates a stepper motor with 1.0125° per step allowing for measurements with an increment of 1° and an error of approximately 1%.

![Schematic of the force balance](image1.png)
![Picture of the force balance](image2.png)

(a) Schematic of the force balance  (b) Picture of the force balance

Figure 6.5: 2-beam force balance.

### 6.2.3 Calibration

Strain-gauges are frequently used on aircraft component testing where tiny strain-gauge strips are glued to structural members, linkages and other critical airframe components to measure stress [73]. A strain-gauge is a long length of conductor arranged in a zigzag pattern on a membrane and when it is stretched, its resistance increases; the sensors are therefore mounted in the same direction as the strain. Strain measurements must be made in the absence of electric and/or magnetic fields otherwise the noise can lead to inaccurate results and incorrect interpretation of the strain signals. In order to avoid noise issues and inaccurate data, it was decided to purchase calibrated load cells and support the two load cells on the bottom of
the support square tube as Figure 6.6 depicts so strain in two directions can be measured. Even if the strain gauges on the load cells are individually pre-calibrated, a calibration method needs to be applied in order to decouple the forces applied on the load cells. Both cells are glued together measuring forces applied on the supporting tube in two different directions. However the system is coupled and both sensors measure forces applied in the other axis as well. Thus, a calibration process needs to be conducted to calculate the forces applied in each direction and decouple the system. This coupling effect is critical in determining the relation between lift and drag.

![Load Cell](image)

Figure 6.6: Load cells used on the force balance.

For force balances in general (whether internal or external), the calibrating variables are the device loads in the balance axis system and the sensor outputs are the bridge outputs. For the purpose of this research, the loads are the aerodynamic lift and drag applied on the wing and transferred to the supporting tube while the outputs are recorded from the load cells that are placed on the tube. The experimental design for the calibration loadings is specified and detailed because the choice of calibration loadings can significantly influence the calibration results. Ideally the calibration process should be performed prior to each major test. This is essential to ensure the validity of the recorded data; however, the process can be tedious and
time consuming. In a first, preliminary, different loads were applied 14 inches from
the fixed point of the supporting tube (considering a 12 inch semi-span model).

Since the relation between the load applied on the support tube and the sensor’s
reading is linear as Figure 6.7 depicts, force decoupling is easier. Loads are applied
on two of the four sides of the square tube, (where lift and drag forces are applied on
the tube) and then two linear equations are used to calculate the coefficients of each
equation. The coefficients are imported into a LabView program that was designed
in order to record the lift and drag coefficients. Data are collected using two digital
multimeters in conjunction with LabVIEW acquisition code.

Figure 6.7: Relation between the load applied and the sensor reading.

Afterwards, and since it was confirmed that the load cells were placed properly
and the relation was linear, it was decided to place the force balance in the wind
tunnel’s testing area and recalibrate the sensor in an upright orientation (Figure 6.8).
The results show high repeatability and the sensor was determined to be ready for
testing. It needs to be highlighted that the force balance calibration is repeated
before wind tunnel testing is conducted.
Figure 6.8: Left: Calibration of the force balance before wind tunnel testing. Right: Calibration and testing of the force balance applying weights on both (lift and drag) directions to evaluate the lift and drag coefficients.

**Thrust Removal Procedure**

Air is provided via a Curtis air compressor and is controlled by an air pressure regulator delivering up to 80 psi (stationary compressor’s maximum pressure). Air is supplied to the wing via a high-pressure flex line connected to a flexible tube with a 9.5 mm (0.375 in) inner diameter. The air at the slot is considered to be incompressible and thus air density is assumed to be constant at the jet. The momentum coefficient of blowing $C_\mu$ is the most critical parameter in understanding the efficiency of blowing in CC. The mass flow rate $\dot{m}_j$ is calculated using Eq. (3.7). The velocity at the jet ($V_{jet}$) is measured by a pitot tube which is placed on the edge of slot jet of the CC wing. The momentum coefficient ($C_\mu$) values are in the range of 0 to 0.3.

In order to remove the effects of the blowing slot static thrust from the wind-on force balance measurements, the static thrust at wind-off conditions must first be measured or calculated. However, it is a complicated procedure to isolate the pure static thrust effects due to the blowing slot. Since the tubing is located inside the
sting, which is part of the force balance that supports the wing model, the force balance sensor cannot distinguish between the static thrust effect and the aerodynamic Coanda effect of the wall-bounded jet, so it measures the combined effect. Therefore, the static thrust at wind-off condition (due to slot blowing), is recorded and subtracted and imported on LabView as offset. For all different momentum coefficients that are applied in this experiment, the same procedure is followed at all angles of attack. Repeating this procedure in all cases ensures that the additional trust caused by the blowing slot is not added to the collected data. It is noted that in a real flight CC will contribute to the overall generated thrust; however, in this study, the implemented thrust removal method underestimates the generated thrust caused by CC. Therefore, it is expected that the efficiency of CC will be better in a real flight.

6.3 Pitot Probe

To measure the velocity at the slot \( V_{\text{jet}} \) with high accuracy, a pitot tube is built and calibrated. To minimize the error caused by jet blockage, the probe of the pitot tube needs to be equal or smaller to the slot height \( h=0.4 \text{ mm} \). A conventional pitot tube is selected and the inner probe diameter is modified for that purpose. The probe length is extended and the inner/outer diameter is gradually reduced, using brass tubing. The outer diameter of the probe is then measured to be 0.5 mm with an inner diameter of 0.3 mm.

The pitot is calibrated using a Flotek 360 wind tunnel with a 6" × 6" × 18" testing section of a maximum velocity of 27 m/s. The pitot tube is placed inside the wind tunnel section next to the wind tunnel’s conventional pitot tube and a
manometer is used to record the dynamic pressure from the conventional pitot tube. The pitot is connected through a Freescale Semiconductor’s MPXV7002 transducer to an Arduino-Uno board. The transducer outputs an analog value proportional to the difference between total and static pressure. The pitot principle of functioning is represented in Figure 6.9. The pitot probe collects the total and static pressures through silicon tubes which are carried to the transducer. The pressure head that is measured from the wind tunnel testing is plotted against the analog output of the transducer.

![Figure 6.9: Pitot probe calibration schematic.](image)

The function relating the analog output with the pressure difference \( dP \) is assumed linear with a maximum error of about 0.2 m/s (1.15%) and the coefficients \( c_1 \) and \( c_2 \) are obtained through the wind tunnel calibration, see Figure 6.10. As a consequence, the relationship between pressure difference and free stream velocity is quadratic. In this case the linear law is defined by Equation (6.1)

\[
dP = 26.381 \times Av - 13552 \tag{6.1}
\]
Figure 6.10: Pitot calibration curves.
Chapter 7

Wind Tunnel Results & Discussion

Low Reynolds numbers (Re < 2.2 × 10^5) wind tunnel tests are conducted and the results are presented in this chapter. Based on the wind tunnel specifications and the focus on small-scale platforms, 2-D and 3-D wind tunnel tests are conducted based on the assumption that most of the platforms at that scale operate at the same Reynolds numbers. The platform, which is described in Section 8.1, is chosen because on the fact that the average speed that operates is the speed that the wind tunnel wing models are tested.

The experimental results of all CCWs are presented, compared and discussed. The effect of blowing on lift and drag coefficients is presented along with the improved lift-to-drag ratios. Lift augmentation ratio results are shown and a comparison between all Coanda surfaces tested show that CC works at all configurations and with upper slot blowing, lift enhancement is achieved. The qualified wing configuration (NACA 0015 CCW) is tested (2-D wind tunnel test) with two dual radius flap configurations and the results are presented next.
Before moving on to the analysis of the results, it is important to understand the level of uncertainty of the measurements and the parameters that may not be controllable during a wind tunnel test. A block diagram representation of a wind tunnel experiment is shown in Figure 7.1. The inputs can be parameters that define the experiment as: angle of attack of the wing, roll, pitch, yaw, etc. and in general the initial conditions that are set. Elements that are controlled and defined by the researcher are the: model size, tunnel size, model material and time of the experiment. The elements of the output vector are the parameters that need to be investigated and are defined by the experiment and are known a-priori. Those elements can be forces, moments components as indicated by the balance, pressure readings, video image or smoke visualization methods, etc. At last, uncontrollable factors include variables as turbulence level of incoming stream, temperature, relative humidity, model deformation etc. Even though in principle a parameter should be controllable, it may be uncontrollable. Barlow et al. [71] states that the scope of the parameters that are controllable depends on the resources available to the experiment planner.

Figure 7.1: Conceptual model of an experimental setup [71].
In a wind tunnel, the flow conditions are not exactly the same as in an unbounded air stream (*free air*) and the distances of some or all of the stream boundaries from the model under investigation are usually less than in an actual operation and this needs to be evaluated before wind tunnel testing. In both 2-D and 3-D wind tunnel testing, tests need to be conducted and data need to be corrected for: Buoyancy, solid and wake blockage and streamline curvature. An accurate description of wind tunnel data and flow field phenomena is possible, if wall interference phenomena are understood and wall interference corrections are applied to the data. However, that requires the appropriate instrumentation. Within the limitations of the facilities immediately available, it was decided not to proceed to 2-D or 3-D data corrections. The models met the solid blockage conditions.

### 7.1 No-Blowing & Blowing Case Comparison

The primary results of this 3-D wind tunnel test are the measured effects of blowing on lift and drag coefficients at different Reynolds numbers and angles of attack. No end-plates are used in this experimental testing. The Force balance is used (Section 6.2) and thrust removal procedure is not applied. No corrections were applied to the data to account for tunnel flow angularity or wall interference. The angle of attack ranges between 0° degrees and the stall angle in order to cover tests for level flight, take-off and stall conditions on a UAV.

#### 7.1.1 Effect of blowing on lift coefficient $C_L$

From all the Coanda surfaces tested, it is found that the (2:1) surface is the most effective one: in all tested wings, the configuration with the (2:1) Coanda surface
gives the best lift enhancement along with low drag coefficients. The effect of blowing is shown for all CCWs at the lowest free stream velocity that is tested, which is equal to $V_\infty = 7.6\text{m/sec}$, and gives a ratio of $V_{jet[Avg]}/V_\infty = 1.04$. The increasing moment coefficient $C_\mu$ creates positive lift increments. The highest value of incremental lift coefficient $\Delta C_L = 0.18$ is measured with the NACA 0015 CCW at the maximum blowing rate and at the angle of attack of $\alpha = 18^\circ$. Results indicate that the blowing technique works as expected, and provides lift enhancement at all CCWs and all angles of attack. On the other hand, for the same blowing coefficients at the highest free stream velocities tested, the effect of blowing on the lift coefficients for the most effective Coanda surface (2:1) is smaller at all times. This behavior is expected since the ratio of the average jet velocity at the slot $V_{jet[Avg]}$ and the free stream velocity $V_\infty$ is reduced and equal to $V_{jet[Avg]}/V_\infty = 0.57$. Surprisingly, the NACA 0015 seemed not to be affected by the velocity ratio change as much as the other wings and gave the highest $\Delta C_L = 0.13$ at $16^\circ$.

Table 7.1 shows the highest values of $\Delta C_L$ that are recorded at both the smallest and highest $V_\infty$ that are tested for all five CCWs.

Table 7.1: Maximum $\Delta C_L$ recorded for all CCWs in this experimental investigation.

<table>
<thead>
<tr>
<th>CCW</th>
<th>Angle of Attack</th>
<th>$\Delta C_L$ max</th>
<th>Angle of Attack</th>
<th>$\Delta C_L$ max</th>
</tr>
</thead>
<tbody>
<tr>
<td>S8036</td>
<td>4°</td>
<td>0.15</td>
<td>16°</td>
<td>0.07</td>
</tr>
<tr>
<td>NACA 2412</td>
<td>2°</td>
<td>0.12</td>
<td>2°</td>
<td>0.06</td>
</tr>
<tr>
<td>NACA 23015</td>
<td>18°</td>
<td>0.17</td>
<td>16°</td>
<td>0.08</td>
</tr>
<tr>
<td>NACA 0015</td>
<td>18°</td>
<td>0.18</td>
<td>16°</td>
<td>0.13</td>
</tr>
<tr>
<td>CLARK - Y</td>
<td>14°</td>
<td>0.13</td>
<td>14°</td>
<td>0.07</td>
</tr>
</tbody>
</table>
Table 7.2: Maximum $\Delta C_D$ recorded for all CCWs in this experimental investigation.

<table>
<thead>
<tr>
<th>CCW</th>
<th>Angle of Attack</th>
<th>$\Delta C_D \text{ max}$</th>
<th>Angle of Attack</th>
<th>$\Delta C_D \text{ max}$</th>
</tr>
</thead>
<tbody>
<tr>
<td>S8036</td>
<td>20°</td>
<td>0.018</td>
<td>20°</td>
<td>0.007</td>
</tr>
<tr>
<td>NACA 2412</td>
<td>18°</td>
<td>0.027</td>
<td>18°</td>
<td>0.010</td>
</tr>
<tr>
<td>NACA 23015</td>
<td>18°</td>
<td>0.021</td>
<td>18°</td>
<td>0.010</td>
</tr>
<tr>
<td>NACA 0015</td>
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<td>16°</td>
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<td>0.024</td>
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<td>0.009</td>
</tr>
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</table>

**Effect of blowing on drag coefficient $C_D$**

In Figure 7.2 the effect of the (2:1) Coanda surface on drag coefficient is depicted. It is observed that as the moment coefficient and the angle of attack increase, the drag coefficient increases slightly as well. However, across all Coanda surfaces tested, a significant increase in drag is not observed. Also of interest is that the drag coefficient at high free stream velocity shows the same behavior as the lift coefficient (Figure 7.3). The velocity ratio is an important factor that influences the effect of blowing during flight. It is observed that the higher the ratio the higher the effect of blowing on any tested configuration.

As Table 7.2 indicates, the S8036 CCW configuration shows less effect of blowing with respect to the drag coefficient. It also needs to be stated that NACA 0015 CCW has the lowest values of incremental drag coefficient at low free stream velocities along with S8036 CCW but at the same time shows the highest effect of blowing on the lift coefficient.

**7.1.2 Lift-to-Drag Ratio**

To get a better understanding of the effect of blowing on lift and drag at each angle of attack, a histogram representation of the lift-to-drag ratio against the angle
Figure 7.2: The effect of blowing on drag force for all CCWs with the most effective (2:1) Coanda surface at $M = 0.022$. 
Figure 7.3: The effect of blowing on drag force of all CCWs with the most effective (2:1) Coanda surface at $M = 0.041$. 
of attack is shown in Figure 7.1.2. It is observed that S8036 and NACA0015 exhibit improved behavior in blowing than the other CCWs, which show similar behavior but the influence of blowing is significantly smaller. However, NACA 23015 CCW shows higher influence at lower angles of attack. At zero and close to zero angles of attack the ratio is close to an average of 20. All plots generated from data collected with the most effective Coanda surface (2:1) and this is an important outcome of this investigation.

Figure 7.4: Histogram representation for the lift-to-drag ratio versus Angle of Attack.
Comparison of incremental lift ($\Delta C_L$) at three Angles of Attack

In this section, results are shown for three different angles of attack: $0^\circ$, $2^\circ$ and $14^\circ$. Those angles of attack represent take-off and cruise flight and they are chosen in order to investigate the effect of blowing on zero, close to zero and high angles of attack. This comparison reveals that not only does blowing enhance lift at all angles but also that in most cases the (2:1) Coanda surface gives the highest lift enhancement. Figure 7.5 shows a comparison of all wing configurations (Coanda surfaces and airfoils) where the behavior of lift coefficients is shown as the moment coefficient of blowing increases.

Results also demonstrate conclusively that with a low blowing rate a high lift augmentation ratio can be achieved at both zero and non-zero angles of attack. The results (Figure 7.6) show that lift enhancement can be achieved at non zero angles and in fact the blowing influence is higher than with zero angles of attack.

Those results in combination with the fact that the effect on drag is small even at high angles of attack and higher blowing rates are important because they indicate that CC can be achieved even at low blowing rates.

Figures 7.8 and 7.9 present the effect of $C_\mu$ on $C_D$ for $\alpha = 0^\circ$ and $\alpha = 14^\circ$ for all Coanda surfaces at the minimum and highest free stream velocities tested. At
Figure 7.5: The incremental lift coefficient plotted against the moment coefficient ($C_\mu$) for all Coanda surfaces at $\alpha = 0^\circ$. 
Figure 7.6: The incremental lift coefficient plotted against the moment coefficient ($C_\mu$) for all Coanda surfaces at $\alpha = 2^\circ$.  

78
Figure 7.7: The incremental lift coefficient plotted against the moment coefficient ($C_\mu$) for all Coanda surfaces at $\alpha = 14^\circ$. 

79
Figure 7.8: Effect of $C_\mu$ on measured drag coefficient at $\alpha = 0^\circ$. 

80
Figure 7.9: Effect of $C_{\mu}$ on measured drag coefficient at $\alpha = 14^\circ$. 
$0^\circ$ incidence, increasing moment coefficient of blowing shows a systematic reduction in $C_D$ in most of the cases at (2:1) Coanda surfaces. Also the (2:1) Coanda surface shows the least increase on drag during blowing. As the angle of attack increases and the blowing rate increases as well, an increase of drag is noticeable which cannot be explained as a $C_{D_{\text{min}}}$ responsible for supercirculation after the end of the boundary control is not expected at these $C_\mu$ values. The $C_\mu$ magnitude that is used is much less than the magnitude that causes supercirculation (0.025 - 0.04) according to literature [39, 34].
Figure 7.10: $C_\mu$ effect on Lift augmentation ratio for the (2:1) Coanda surface.

Figure 7.10 examines the behavior of the lift augmentation ratio as a function of $C_\mu$. The results demonstrate a decrease in lift augmentation ratio as the moment coefficient of blowing increases. The data follow the same trend as in the experimental investigation at NASA Langley Research Center [23], which is expected since the velocity ratio, which is one of the main factors of lift enhancement, decreases as well.
The highest lift augmentation ratio is reported on the S8036 CCW configuration with the (2:1) Coanda surface and is equal to 61 at $\alpha = 2^\circ$ angle of attack. In all cases the data show the same behavior but in some cases (Clark-Y and NACA 23015) the (2:1) does not give the highest augmentation ratio. This study suggests that before a practical implementation is considered, further wind tunnel testing should be conducted in order to investigate the effect of the (2:1) Coanda surface in different design constraints (radius, position, slot height). Circulation control on small UAVs can be applied if flow uniformity at the slot is achieved and there is sufficient space for the design of the plenum and an energy efficient air supply on-board. The current results show that lift enhancement can be achieved even at low blowing rates with less drag increase. The achieved lift enhancement would be insufficient at those blowing rates on a small UAV, however future research will focus on those configurations that are efficient at low blowing rates and further investigation will be conducted at higher rates.

7.2 NACA 0015 CCW Configuration

7.2.1 Wind Tunnel Test Conditions

The wind tunnel testing is conducted at a freestream Mach number of 0.03. The Reynolds number range based on the mean chord and freestream Mach numbers is 130,000 to 150,000. The angle-of-attack varies from $0^\circ$ to $21^\circ$ where stall occurred in most of the cases. All wind tunnel tests that are conducted here are 2-D and the experimental setup is presented in Figure 7.11.
Figure 7.11: 2-D testing arrangement with two endplates in the wind tunnel.

The Force balance is re-calibrated (Section 6.2.3) and used to measure the aerodynamics forces that are applied. The performance of the CCW model is best described as a function of the momentum coefficient, $C_\mu$, thus the tests are conducted by setting five chosen $C_\mu$ values in order to investigate the effect of CC blowing in both the boundary layer control and supercirculation regime. During testing, the thrust removal procedure is applied (Chapter 6, Subsection 6.2.3) and the data presented below does not include the thrust from the blowing slot. No corrections were applied to the data to account for tunnel flow angularity or wall interference. During back-to-back repeats the uncertainty, based on $\sigma_N/N^{0.5}$, of $C_l$ is within the band of $\pm 0.0069$ and $C_d$ of $\pm 0.0027$, $\sigma_N$ being the standard deviation and N the number of runs the experiment is repeated. The pitot probe (Section 6.3) is used to set the momentum coefficient before each testing is conducted. Flow uniformity is also tested before every set of experiments.

85
7.2.2 Cruise Flight Performance, 0° Flap Deflection

To examine the effect of blowing at cruise flight conditions, a wind tunnel test is performed on the modified NACA 0015 with both DRF10 and DRF45 flap designs at various (from −6° to +6°) angles-of-attack around 0° angle of attack. To compensate for the pitching moment effect that is expected during blowing, it is decided to investigate the effect of blowing in a bigger range of angles of attack around 0°.

![Figure 7.12: 0° deflection flap comparison at no blowing case.](image)

Before examining the effect of blowing, both DRF10 and DRF45 flap configurations are tested at 0° flap deflection and no blowing condition to investigate the effect of the dual radius flap with a large and a small Coanda radius throughout the exit of the flap on the L/D ratio. From the results obtained (Figure 7.12), it is shown that the longer flap with the bigger Coanda radius ratio \((r_2/r_1)\) shows a higher L/D ratio. However, the DRF10, is found to be more efficient at different blowing rates and showed a higher lift augmentation at all angles of attack. A comparison between the two configurations at −2°, 0° and 2° angles of attack is shown in Figure 7.13.
Figure 7.13: Efficiency of the $0^\circ$ deflection flap configurations at $-2^\circ$, $0^\circ$ and $2^\circ$ angle of attack respectively.

From the plots of the lift curves it is observed that the CC efficacy is higher at the flap with a smaller $r_2/r_1$ ratio. Figure 7.14 shows the effect of blowing at the smaller and larger $C_\mu$ values tested on the DRF10 flap (left) and DRF45 flap (right). Although, it is found that lift enhancement is higher on the smaller flap, the effect of blowing on drag shows different characteristics. Upper slot TE blowing at low rates gives in both flap configurations the ability to manage well lift-to-drag ratios by generating reduced drag as shown in Figure 7.15. The drag reduction is higher at angles of attack around $0^\circ$. It is also shown that at low $C_\mu$, the drop on drag is higher when the DRF45 is used. However, using the same flap at high blowing rates
the drag increases and reaches values higher than the no blowing case for positive angles of attack.

Figure 7.14: Effect of $C_\mu$ on lift with the DRF10 flap (Left) and with the DRF45 flap (Right).

Figure 7.15: Effect of $C_\mu$ on drag with the DRF10 flap (Left) and with the DRF45 flap (Right).

7.2.3 High-Lift Takeoff Performance, 30° Flap Deflection

Both flaps are tested under the same conditions and showed a good CC performance. CC is applied in both configurations and the $C_{l_{\text{max}}}$ achieved is 2.47. The incremental lift coefficient ($\Delta C_l$) that is measured is 0.89 at 2° angle of attack us-
ing the smaller flap (DRF10). The wing configuration with the DRF45 flap showed higher $C_l$ and $C_d$ values at the no blowing case, however, the wing configuration with the smaller flap achieves the same and higher lift coefficient values during blowing. From the analysis of the results, it is well understood that the lift enhancement efficiency is better with the smaller flap, since it achieves the same $C_l$ values at lower $C_d$ values. However, the maximum lift coefficient achieved is the same for both flaps.

![Figure 7.16: The effect of blowing on the DRF10 flap at different angles of attack.](image)

The effect of blowing can be seen in Figures 7.16 and 7.17, where the lift curves and the drag polars are plotted. The drag polars for different blowing cases, show that the lift-to-drag ratios are increased with blowing. For low blowing rates, and before super circulation ($C_\mu \leq 0.1$) lift is enhanced with small increase of the drag. For higher blowing rates ($C_\mu > 0.1$), lift enhancement is achieved with higher drag penalties.
Figure 7.17: The effect of blowing on the DRF45 flap at different angles of attack.

7.2.4 High-Lift Takeoff Performance, 60° Flap Deflection

The 60° Flap deflection configurations performed poorly in terms of lift augmentation compared to the 30° flap deflection with CC blowing. The behavior of the DRF10 and the DRF45 (Figure 7.18) followed the same trend as in the 30° deflection flaps but the lift enhancement found to be lower.

The thickness ratio on the airfoil has a direct effect on maximum lift, drag and stall characteristics. On un-blown CC wings the thickness ratio primarily affects the maximum lift and stall characteristics by its effect on the nose shape[50]. For a symmetrical airfoil of a low aspect ratio, no sweep and a large nose radius, a higher stall angle and a greater maximum lift coefficient is expected. However, without blowing or active flow control the drag increases with thickness due to increased separation and the stall angle is lower due to pitching moments generated from TE blowing. Increasing $C_\mu$ does not affect the stall angle in 0° and 30° configurations but the effect is notable in 60° deflection configurations since it stalls earlier up to 2° degrees at $C_\mu \geq 0.2$ (Figure 7.19).
Figure 7.18: Lift and drag performance for both flap configurations at no blowing case.

Figure 7.19: Effect of blowing on lift coefficient for DRF10 (Left) and DRF45 (Right) at different angles of attack.

From Figure 7.20 can be seen that after super circulation ($C_\mu > 0.1$) lift and drag are increased resulting in a shift to the right of the lift-to-drag curve. It is shown that the smaller flap (DRF10) achieves higher lift enhancement with lower drag penalties. However the drag penalties are high at super circulation regime. It needs to be noted that the effect of CC blowing is seen at all angles of attack and at all blowing cases.
Figure 7.20: Effect of blowing on lift coefficient for DRF10 (Left) and DRF45 (Right) at different angles of attack.

7.2.5 Further Data Analysis

To characterize the lift performance of the dual blown configurations and summarize the results of all tested cases, a direct comparison and further analysis is performed. Analysis investigates the design sensitivities and the flap design complexity of the development of an efficient CCW for small scale UAVs. Wind tunnel tests are conducted to achieve sufficient lift enhancement at low blowing rates enhanced lift-to-drag ratios at cruise flight. Up to 11 times enhanced lift-to-drag ratio is achieved with the DRF10 0° drag deflection flap at 0° angle of attack at the maximum blowing rate. The effect is observed at all tested angles and the results are presented in Figure 7.21.
As expected, the results showed that the upper blowing performance remains proportional to lift. The higher the blowing, the higher the lift enhancement. However, the lift augmentation does not increase as much since drag penalties increase with blowing as well. Comparing DRF10 with DRF45 at both flap deflections it is observed that with DRF10 configuration higher lift enhancement is achieved in all tested cases. Figure 7.22 shows the comparison of DRF10 and DRF45 at 30° flap deflection of 0° (Left) and 18° (Right) angle of attack. The difference in lift enhancement is not much and the maximum incremental lift coefficient ($\Delta C_l$) is found to be 0.884 by using the DRF10 configuration at 2° angle of attack.
At 60° flap deflection, the measured lift enhancement is lower compared to the same flap configurations at 30° deflection. As Figure 7.23 depicts, the DRF10 performs again better compared to the DRF45 at all angles and all blowing rates, and that can be seen from the two representative angles that are presented here.

Figure 7.23: Comparison of DRF10 and DRF45 at 0° (Left) and 180° (Right) angle of attack.

The overall results of the present Coanda dual radius flap investigation are promising. The results show that the DRF10 flap that is tested at 0°, 30° and 60° deflection is found to be the most CC efficient flap from the ones that are tested. The performance is better in all cases and in all blowing rates. It gives high lift-to-drag ratios at 0° deflection and at 0° angle of attack and it performs better in terms of high-lift enhancement at high-lift takeoff testing scenarios where the deflection of the flap is high.
Chapter 8

UC\textsuperscript{2}AV: Unmanned Circulation Control Aerial Vehicle

8.1 The Platform

In order to apply CC to a small-scale fixed-wing UAV, space limitations and weight restrictions must be considered. A suitable CC system\textsuperscript{1} consisting of the Air Supply Unit (ASU), the Air Delivery System (ADS) and the CCW needs to be designed to meet set restrictions. Since the ASU and part of the ADS are located inside the fuselage and close to the center of gravity of the aircraft, a UAV with sufficient fuselage space needs to be considered. High payload capability is also a requirement as the CC system adds weight on-board the UAV that needs to be compensated for. In addition, wingspan, wing loading and the chord length must also be considered. The Anaconda RMRC (Figure 8.1), which is an inverted V-tail twin-boom type aircraft, has a wingspan of 2 m and sufficient fuselage space (Table

\textsuperscript{1}The word system is interchangeable and refers either to the air supply system (ASU), the air delivery system (ADS) and to the CC system.
Table 8.1: Anaconda RMRC geometric characteristics.

<table>
<thead>
<tr>
<th>Fuselage</th>
<th></th>
<th></th>
</tr>
</thead>
<tbody>
<tr>
<td>Length</td>
<td>L</td>
<td>0.8 m</td>
</tr>
<tr>
<td>Max. Height</td>
<td>$H_{\text{max}}$</td>
<td>0.11 m</td>
</tr>
<tr>
<td>Max. Width</td>
<td>$W_{\text{max}}$</td>
<td>0.16 m</td>
</tr>
</tbody>
</table>

<table>
<thead>
<tr>
<th>Propeller</th>
<th></th>
<th></th>
</tr>
</thead>
<tbody>
<tr>
<td>Diameter</td>
<td>D</td>
<td>15 in</td>
</tr>
<tr>
<td>Pitch</td>
<td>P</td>
<td>4 in</td>
</tr>
<tr>
<td>Number of Blades</td>
<td></td>
<td>2</td>
</tr>
</tbody>
</table>

8.1. The aircraft also has an average wing chord of 238 mm with slats, flaps and ailerons. It has a payload of about 1.5 kg and an average speed of 10 m/s. The Anaconda is chosen as the candidate testbed since it meets all design requirements (Figure 8.1).

![Figure 8.1: RMRC Anaconda UAV.](image)
8.2 Air Supply Unit & Air Delivery System

8.2.1 Air Supply Unit

Compressed air-based technology required for CC has difficulties when implemented on-board small-scale fixed-wing UAV platforms due to payload, space and power constraints. Hence, small and efficient modules that may be integrated on those platforms are investigated. To tackle this problem, various air supply designs, which can provide the required mass flow rate for a CC-based flight are studied. Multiple centrifugal compressor configurations are designed and a Computational Fluid Dynamics (CFD) analysis is conducted to validate performance [74, 75]. Tested configurations are chosen based on: the efficiency of the compressor with respect to the power consumption; a trade-off between weight (payload) and air supply efficiency; the controllability of the jet stream velocity.

An air compressor is a mechanical device capable of transferring energy to air so it can be delivered in large quantities at higher pressure [76]. Variable displacement compressors are of two kinds: axial and centrifugal compressors. Axial compressors are suitable for large flow applications at low pressure, while centrifugal compressors are more commonly used for medium flow and higher pressure applications. Performance characteristics of an air supply system to power CCWs are defined by two parameters: the mass flow rate and/or the moment coefficient of blowing ($C_\mu$); and the pressure-ratio at which the air can be supplied to the trailing edge of the CCW [45]. Centrifugal compressors can be used for high energy air applications. If CC is applied for high-lift augmentation purposes, high velocities at the slot are required and centrifugal compressors can be designed and optimized to reach the performance that is required. Various compressor designs are investigated and it is concluded that
a forward impeller centrifugal compressor can meet the mass flow requirements with lower losses compared to the other configurations that are tested [74, 75]. The design methodology to improve the efficiency of the centrifugal compressor that is followed is presented in [74]. A CFD analysis is also conducted to investigate and optimize the efficiency of the design before manufacture. The qualified ASU design can be seen in Figure 8.2.

Figure 8.2: Left: The housing. Center: The impeller. Right: The motor mount. The weight of the ASU is 85 g without the motor.

### 8.2.2 Air Delivery System

The ADS is a high pressure air system that delivers a continuous supply of air to the slot that is located at the TE of the CCW. The system is designed to provide a mass flow of 0.03 kg/s and share it equally between two flow paths. The ADS consists of: i) the inlet ducting passage; ii) the internal ducting that connects the ASU with the plenum; and iii) the plenum, which is a vaned straight-walled wide angle diffuser and is responsible for delivering the required mass flow uniformly along the span. Losses in a piping system are typically categorized as major and minor losses [24, 25]. The ADS is a piping system that consists of pipe inlets and outlets, fittings and bends, expansions and contractions, minor losses should be considered
since they account for more head loss than the pipes themselves. A complete CFD analysis and experimental test is conducted on four junction designs (Figure 8.3). It is found that junction-4 is the most efficient design (minimized losses) compared to the other tested designs.

![Figure 8.3: Left: CAD designs of junction configurations 1-4. Right: Velocity contour of the junctions at 50 m/s inlet velocity from CFD analysis.](image)

### 8.3 Circulation Control System

To overcome CC challenges (mass flow requirements, source of air, power requirements, etc.) and to apply upper slot TE active flow blowing on small-scale UAVs, focus is on the CC system that provides the mass flow and delivers it at the slot. Since CC is used for lift enhancement, the system must be: light-weight; capable of reducing the air losses; and delivering/distributing the flow uniformly across the span. The overall system including the motor that runs the ASU, adds 0.65 kg (1.4 lbs) on-board and is able to provide the mass flow required for sufficient lift enhancement and takeoff runway reduction. The two plenum designs can distribute the air uniformly across the span and the 3-D printed tubing reduces the overall weight. The CC system consists of: the ASU, which defines the CC efficiency by providing the required mass flow; the ADS that is responsible to deliver the mass flow with
minimum losses at the slot; the CCW with the dual radius flap that is designed with the required geometry for efficient active flow blowing.

The qualified ASU design with the ADS and the CCW design are shown in Figure 8.4 (left) and a CAD design of the system is presented in Figure 8.4 (Right). Details of the system can be found in [75].

Figure 8.4: Left: Circulation Control system. Right: CAD design of the CC system.

8.4 Circulation Control Wing

8.4.1 CCW Structural Analysis

In an aircraft wing structure ribs and support rods or spars are provided to support and give rigidity to the wing section. The wing structure, which is already integrated with the ADS and the plenum design, needs to be strong and light-weight. Focus is given on the strength of the ribs of the wing structure and Finite Element Analysis (FEA) using ANSYS Workbench is used, to investigate which of the three considered configurations meets the design requirements.

The wing structure (Figure 8.5) is modeled in Solidworks. It consists of twenty ribs in transverse direction and two rods in longitudinal direction. Two of the ribs
have two mounting points each, which are screwed to the fuselage and they are considered as the fixed points in this analysis. Support and additional strength is given by the plenum and the dual radius flaps, which are also part of the wing model on this structural analysis. The material that is used, the yield strength, the specifications of the wing structure and the configurations that are tested, are presented in Tables 8.2 and 8.3.

Figure 8.5: Wing structure design using Solidworks.
Table 8.2: Input parameters of the wing structure.

<table>
<thead>
<tr>
<th>Wing Structure: Input Parameters</th>
</tr>
</thead>
<tbody>
<tr>
<td>Root Chord</td>
</tr>
<tr>
<td>Tip Chord</td>
</tr>
<tr>
<td>Span Length</td>
</tr>
<tr>
<td>Exposed Semi Span</td>
</tr>
<tr>
<td>Aircraft Weight</td>
</tr>
<tr>
<td>Lift Load</td>
</tr>
<tr>
<td>Safety Factor</td>
</tr>
<tr>
<td>2 Rods</td>
</tr>
<tr>
<td>20 Ribs</td>
</tr>
<tr>
<td>Leading Edge/ Ailerons</td>
</tr>
<tr>
<td>Flaps</td>
</tr>
</tbody>
</table>

**Loads Acting Over the Wing Structure**

Lift load is considered as important criterion while designing an aircraft. Fuselage and wings are the two regions where lift load is acting on an aircraft but since the exact percentage of lift load that is acting on the fuselage cannot be evaluated at this point, it is assumed that the total lift load is acting on the wings. The maximum load acts nearer the wing roots and for simplicity and symmetry in FEA tests a semi-span model is usually used. In this FEA simulation, the total wingspan is used since the wing is built in one piece due to the complexity of the ADS that is integrated in the wing structure. Also, the support rods are connecting the two semi-span wings, adding more support to the structure. The four fixed points, which are located on the middle ribs of the wing model, are the mounting points of the wing to the fuselage (Figure 8.5).

In aeronautics, the load factor (n) is defined as the ratio of the lift of an aircraft to its weight (Equation (8.1)) and as a ratio of two forces, it is dimensionless. How-
ever, its units are referred to as $g$, because of the relation between load factor and the acceleration of gravity felt on-board the aircraft. A load factor of one, or $1g$, represents conditions in straight and level flight, where the lift is equal to the weight as it can be derived from Equation (8.1).

$$n = \frac{L}{W}$$ (8.1)

Load factors greater or less than one (or even negative) are the result of maneuvers or wind gusts during flight. The lift load force is applied on the aerodynamic center of each rib (1/4 of the chord length). The total thrust that the motor can give is equal to 41.2 N when the total weight of the aircraft is 48 N. That practically means that in a case of a heavy plane such as the U$^2$CAV it is not expected that more than 2 $g$ condition will be experienced during takeoff and climb maneuver. However, in turning flight the load factor is normally greater than 1. For example, in a turn with a 60° angle of bank the load factor is 2. In a balanced turn in which the angle of bank is $\alpha$, the load factor is related to the cosine of $\alpha$ as it is given by Equation (8.2)

$$n = \frac{1}{\cos \alpha}$$ (8.2)

The total design load on the aircraft is given by Equation (8.3) and is the product of the takeoff weight, the safety factor and the load factor that is applied on the aircraft. The safety factor, is a term describing the structural capacity of a system beyond the expected loads or actual loads and for commercial aircraft is set to be equal to 1.5 [77]. Essentially, the factor of safety is how much stronger the system is than it usually needs to be for an intended load. The wing is tested for up to 4 $g$
lift load which is the maximum load that is expected to be applied on this type of aircraft.

\[
Total_{\text{designload}} = 1.5 \cdot W_{\text{T.O.}} \cdot n
\quad (8.3)
\]

Table 8.3 shows the tested configurations and the wing’s weight for each case. The challenging part is to design a wing structure that is light-weight while at the same time is safe. A safe structure is when the stress magnitude which is obtained from the analysis is less than the yield strength of the material. The results of all tested cases (multiple \(g\) conditions) and three configurations are presented in Tables 8.4, 8.5, 8.6. Configuration-II has the support of carbon fiber rods and the ribs that are mounted on the fuselage (fixed points on ANSYS) are made of acrylonitrile-butadiene-styrene (ABS) plastic. That configuration is 31% heavier than the configuration-I. Configuration-III has all ribs made of ABS and the support of 2 carbon fiber rods. This configuration is the heaviest of all and is up to 55% heavier than configuration-I.

<table>
<thead>
<tr>
<th>Yield Strength (MPa)</th>
<th>Configuration I</th>
<th>Configuration II</th>
<th>Configuration III</th>
</tr>
</thead>
<tbody>
<tr>
<td>Balsa Wood Ribs</td>
<td>20</td>
<td>✓</td>
<td>✓</td>
</tr>
<tr>
<td>Pine Spruce Rods</td>
<td>38</td>
<td>✓</td>
<td></td>
</tr>
<tr>
<td>Carbon Fiber Rods</td>
<td>1720</td>
<td>✓</td>
<td>✓</td>
</tr>
<tr>
<td>ABS Plastic Ribs</td>
<td>44</td>
<td></td>
<td>✓</td>
</tr>
<tr>
<td>Weight (gr)</td>
<td>-</td>
<td>652.85</td>
<td>855.61</td>
</tr>
</tbody>
</table>

Table 8.3: Design parameters of the tested configurations.
Figure 8.6: Wing structure design on ANSYS Workbench. Configuration-III when 1g condition is applied (total deformation results).

Figure 8.6 shows the wing structure in ANSYS Workbench. In configuration-I since for the support rods pine spruce wood is used, it is expected that the total deformation will be the maximum compared to the other two configurations (Table 8.4). In addition to the total deformation, the stress on three points is investigated. The strength on the fixed points, the strength on the connections between the rods and the ribs and also the stress on the supporting rods.

In the first configuration, it is observed that at 2g (highlighted in red), the structure is not safe since the maximum yield strength that is applied, is higher than the yield strength of balsa wood. The yield strength of balsa is 20 MPa and all points of interest start the plastic deformation at the 2g condition and for that reason the analysis is terminated.

Table 8.4: Stress values at various g Conditions for wing Configuration-I.

<table>
<thead>
<tr>
<th>Wing configuration - I</th>
<th>g Condition</th>
<th>Max Total Deformation (mm)</th>
<th>Max Equivalent (von-Mises) Stress Applied on the Fixed Points (MPa)</th>
<th>Max Equivalent (von-Mises) Stress Applied on the Ribs (MPa)</th>
<th>Max Equivalent (von-Mises) Stress Applied on the Rods (MPa)</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>1</td>
<td>22.910</td>
<td>20.517</td>
<td>15.389</td>
<td>10.261</td>
</tr>
<tr>
<td></td>
<td>2</td>
<td>45.821</td>
<td>41.035</td>
<td>30.507</td>
<td>20.521</td>
</tr>
</tbody>
</table>
In configuration-II (Table 8.5) the balsa wood middle ribs are replaced with ABS ribs and the material for the supporting rods is carbon fiber. With this modification, the total deformation is reduced by 30% and the plastic deformation on the fixed points occurs at 3g condition. The deformation on the rest of the ribs will start earlier but that failure is not crucial for the design. There will be plastic deformation on the contact of the ribs with the support rods but that will not cause structure failure. This structure is stronger than configuration-I but heavier. The last configuration that is tested (Table 8.6), has the ribs been replaced with ABS plastic and that made the structure the heaviest between the other ones tested. Configuration-III can tolerate bigger loads but still modifications on the contact points of the ribs with the supporting rods need to be considered.

Table 8.5: Stress values at various g Conditions for wing configuration-II.

<table>
<thead>
<tr>
<th>g Condition</th>
<th>Max Total Deformation (mm)</th>
<th>Max Equivalent (von-Mises) Stress Applied on the Fixed Points (MPa)</th>
<th>Max Equivalent (von-Mises) Stress Applied on the Ribs (MPa)</th>
<th>Max Equivalent (von-Mises) Stress Applied on the Rods (MPa)</th>
</tr>
</thead>
<tbody>
<tr>
<td>1</td>
<td>15.994</td>
<td>17.090</td>
<td>17.090</td>
<td>17.090</td>
</tr>
<tr>
<td>2</td>
<td>31.988</td>
<td>34.180</td>
<td>25.635</td>
<td>51.268</td>
</tr>
<tr>
<td>3</td>
<td>47.982</td>
<td>51.270</td>
<td>38.45</td>
<td>76.902</td>
</tr>
</tbody>
</table>

Table 8.6: Stress values at various g Conditions for wing Configuration-III.

<table>
<thead>
<tr>
<th>g Condition</th>
<th>Max Total Deformation (mm)</th>
<th>Max Equivalent (von-Mises) Stress Applied on the Fixed Points (MPa)</th>
<th>Max Equivalent (von-Mises) Stress Applied on the Ribs (MPa)</th>
<th>Max Equivalent (von-Mises) Stress Applied on the Rods (MPa)</th>
</tr>
</thead>
<tbody>
<tr>
<td>2</td>
<td>32.315</td>
<td>38.584</td>
<td>28.940</td>
<td>48.229</td>
</tr>
<tr>
<td>3</td>
<td>48.4472</td>
<td>43.410</td>
<td>43.410</td>
<td>86.809</td>
</tr>
<tr>
<td>4</td>
<td>64.429</td>
<td>57.169</td>
<td>57.169</td>
<td>115.75</td>
</tr>
</tbody>
</table>

Stress analysis of the wing structure is carried out and maximum stress is identified at wing root which is found to be lower than the yield strength of the material for
the first three $g$ conditions. Configuration-III is qualified since it is the safest design for flight testing. Configuration-III is a strong design that adds support to the CC system and can tolerate much more stress than the lift loads that are expected to act during flight. The steps for the implementation of the wing structure are explained next.

### 8.4.2 NACA 0015 Wing Implementation

Before the final CCW is designed and built, it was decided to build a conventional NACA 0015 integrated with the twin-boom inverted V-tail UAV to serve as the baseline wing; relevant flight data using the conventional NACA 0015 will be compared to data collected when using the corresponding CCW. The wing is built using the profile of a symmetric NACA0015, with a span of 2 meters and a chord length of $c = 240$ mm, yielding an aspect ratio of $AR = 8.33$. The geometric parameters of the wing are shown in Table 8.7.

The wing consists of: the ribs (10 each side), which are printed out of ABS plastic (Figure 8.7 (left)) and two carbon fiber rods; the leading- and trailing-edge made out of balsa wood (Figure 8.7 (Right)); and MonoKote tape. MonoKote is applied using heat, which causes the covering to shrink and activates an adhesive backing that is attached securely to the wing model and gives a smooth surface finish (Figure 8.8 (left)). The plastic ribs and the carbon fiber rods add strength to the structure of the wing without adding excessive weight. The process followed to build the NACA 0015 will also be followed to build the CCW. Note that part of the CC system is integrated with the fuselage and other parts (tubing and plenum design) are inside the CCW.
Figure 8.7: Left: The wing structure with the two carbon fiber rods and the ten ribs, equally spaced (100 cm). Right: The balsa leading- and trailing-edge parts and the balsa wood skin.

Figure 8.8: Left: MonoKote tape for smooth surface finish. Right: The UAV with the NACA0015 wings.
Table 8.7: Geometric parameters of the wing.

<table>
<thead>
<tr>
<th>Wing</th>
<th></th>
<th></th>
</tr>
</thead>
<tbody>
<tr>
<td>Airfoil</td>
<td>NACA0015</td>
<td></td>
</tr>
<tr>
<td>Chord</td>
<td>c</td>
<td>0.24  m</td>
</tr>
<tr>
<td>Thickness</td>
<td>t/c</td>
<td>15    %</td>
</tr>
<tr>
<td>Camber</td>
<td>m/c</td>
<td>0     %</td>
</tr>
<tr>
<td>Angle of incidence</td>
<td>ϵ</td>
<td>2 °</td>
</tr>
<tr>
<td>Area</td>
<td>S</td>
<td>0.48  m²</td>
</tr>
<tr>
<td>Wingspan</td>
<td>b</td>
<td>2.06  m</td>
</tr>
<tr>
<td>Half-Span</td>
<td>s</td>
<td>1     m</td>
</tr>
<tr>
<td>Aspect Ratio</td>
<td>AR</td>
<td>8.33</td>
</tr>
<tr>
<td>Dihedral angle</td>
<td>β</td>
<td>0 °</td>
</tr>
<tr>
<td>Height of wing above ground</td>
<td>H</td>
<td>0.3   m</td>
</tr>
<tr>
<td>Sweep angle</td>
<td>Λ</td>
<td>0 °</td>
</tr>
</tbody>
</table>

<table>
<thead>
<tr>
<th>Aileron</th>
<th></th>
<th></th>
</tr>
</thead>
<tbody>
<tr>
<td>Span</td>
<td>b_{ail}</td>
<td>0.50  m</td>
</tr>
<tr>
<td>Chord</td>
<td>c_{ail}</td>
<td>0.03  m</td>
</tr>
<tr>
<td>Maximum Deflection</td>
<td>ξ_{max}</td>
<td>-15&lt;ξ&lt;10</td>
</tr>
</tbody>
</table>

The wing structure consists of skin, ribs and rods sections. The carbon fiber rods carry flight loads and the weight of the wings while on the ground. Other structural and forming members such as ribs are connected to the rods, with balsa wood skin. The wings are the most important lift producing part of the aircraft and the design of wings may vary according to the type of aircraft and its purpose. As it is explained in Section 8.4.3, due to the wing complexity, and the position of the
components (tubing, plenum, etc.) the structure of the wing needs to tolerate the loading applied and to be lightweight without adding extra weight on the plane.

8.4.3 CCW Implementation

The same methodology is followed for the implementation of the CCW. The base case wing (conventional NACA 0015 airfoil shape wing) showed that designing and implementing a wing that weights the same as a foam wing but with a stronger structure, is feasible. Providing lift is the main function of the wings and since CC is used to to enhance lift, a light-weight structure that can tolerate increased wing loadings during flight, is required. The wing consists of two essential parts: the internal wing structure, consisting of rods and ribs; and the external wing, which is the skin and the MonoKote tape. The ribs, besides adding strength to the structure, give the shape to the wing section, support the skin and prevent the structure from buckling or twisting. They also serve as attachment points for the control surfaces, flaps slats and ailerons. In addition, the ribs add support to two plenum designs that are located inside the wing. For that purpose, it was decided to use 20 ribs for the 2 m wing span (Figure 8.9).
The ribs need to be supported and that is done by the carbon fiber rods. The rods are the most heavily loaded parts of the wing. Due to piping and ADS complexity, it was decided that the 2 m wing will be built as a single piece and due to that factor, the rods are reinforced in a way that they carry more force at the roots, than the tips. Since it is expected that the wing will bend upwards, the rods usually carry shear forces and bending moments. The second rod is positioned to prevent the wing from twisting. That way, torsion now includes bending of the two rods, which is termed differential bending (Figure 8.10). It is usually hard to attach the wing to the fuselage. The connection of wings and fuselage are by four strong bolts (stronger than necessary), thereby having sufficient lifetime and keep the wing attached to the fuselage.
Finally, the skin is made from balsa wood and it acts both as a spar-cap, to resist bending and as a torsion-box to resist torsion and to transit aerodynamics forces. MonoKote is applied to give a smooth surface finish (Figure 8.11).

The UC²AV is shown in Figure 8.12. A duct is designed, built and integrated on the belly of the aircraft. The duct is connected with the inlet of the ASU. Aluminum
wire mesh is used on the duct inlet to prevent rocks from entering the ASU’s housing and damage the impeller.

Figure 8.12: The UC²AV.
Chapter 9

Takeoff Performance & Instrumentation

The primary emphasis of this chapter is to discuss the takeoff performance of fixed-wing aircraft. Takeoffs are affected by factors that cannot be accurately measured nor properly compensated for, thus field takeoff tests are important portions of the flight test program of the UC2AV. All takeoffs are recorded for data analysis purposes and all test flights are devoted entirely to takeoff tests in various conditions or configurations including, crosswind operations, wet or icy runway operations and various takeoff weights. The criteria for takeoff performance tests and the required instrumentation are presented here.


development of the UC²AV compared to a conventional UAV.

Figure 9.1: Takeoff performance expected behavior of the UC²AV compared to a conventional UAV.

During takeoff roll, in addition to lift, weight, thrust and drag forces, the aircraft is affected by additional resistance. This force includes wheel bearing friction, tire deformation and the energy that is absorbed by the wheels as the wheel rotation increases. However, this resistance force, which mathematically can be expressed as
\( \mu(W-L) \), becomes smaller as the weight on the wheels is reduced and typical values for dry asphalt runway as the one that is used for all takeoff tests can range between 0.02 and 0.05 [78]. Given the fact that the aerodynamic lift and drag increase during takeoff in direct proportion to the square of the airspeed (Section 3.1.2, Equations (3.2), (3.3)), while the coefficient of resistance \( \mu \) and the aircraft takeoff weight \( W_{T.O.} \) remain constant, the total ground and air distance can be calculated by Equations (9.1) and (9.2) respectively.

\[
\ell_{\text{ground}} = \frac{WV_{T.O.}^2}{2g[F - D - \mu(W - L)]_{avg}} \tag{9.1}
\]

\[
\ell_{\text{air}} = \frac{WV_{G}^2 - 50 + V_{T.O.}^2}{F - D}_{avg} \tag{9.2}
\]

### 9.1.2 Pilot Takeoff Technique & Takeoff Corrections

Gross weight, air density, wind conditions (calm, headwind, tailwind or crosswind), coefficient of friction, etc. are some of the parameters that can significantly effect the takeoff distance and proper consideration must be given to them.

**Pilot Technique**

From the moment an airplane starts its takeoff roll until it reaches a safe maneuvering altitude (above 50 ft), it passes though what can be considered a zone of risk. Individual pilot technique can cause a greater variation in takeoff data than all other parameters combined. Factors that significantly affect takeoff performance among others are: aileron and elevator position during acceleration; pitch rate during rotation and angle of attack at lift off. To eliminate the variation due to pilot’s
individual technique and obtain repeatable data, a specific takeoff technique, where the pilot applies full throttle and the aircraft takes off once the required velocity is achieved, is followed. The pilot before takeoff verifies that the pitch trim is set to takeoff position. During the takeoff maneuver, the pilot’s responsibility is to keep the directional control while the airplane is on the runway. Pitch is not introduced during takeoff and angle of attack at lift-off can be assumed repeatable under the same conditions (maximum takeoff weight, weather conditions, etc.).

Wind Corrections

A day in the field where the wind is higher than 10 knots, not only it is not safe for the aircraft (small-scale UAVs) but also the probability of collecting useful data is low. Since the wind corrections are the first to be applied, to protect the aircraft and collect data that can be easier analyzed, a day of strong winds ($\geq 10$ knots) is a no-fly situation. The wind correction process that is described next is applied in all collected data with winds less than 10 knots. Regardless the wind condition (less or higher than 10 knots), the ground speed and true airspeed are equal in a no-wind situation. The ground speed required with wind is given by Equation (9.3). The sign of $V_W$ is positive for a headwind and negative for a tailwind and it includes always the component of wind velocity parallel to the runway.

$$V_{T.O_N} = V_{T.O} - V_W$$

(9.3)

An empirical relationship that has been developed and works well for steady winds less than 10 knots is presented in [78] and Equation (9.4) gives the ground distance after the wind correction process.
\[ \ell_{\text{ground}} = \ell_{\text{ground}w} (1 + \frac{V_W}{V_{T.O.}})^{1.85} \]  \hspace{1cm} (9.4)

For the air distance (phase II of takeoff) the equation is simpler and an exact determination of wind velocity is difficult to be measured accurately. Equation (9.5) describes the correction that needs to be applied.

\[ \ell_{\text{air}} = \ell_{\text{air}w} + \Delta \ell \]  \hspace{1cm} (9.5)

No corrections are applied to compensate for runway slope, since the runway is inspected and no downhill or uphill points to indicate a sloping runway were found.

### 9.1.3 One-at-a-Time Sensitivity Analysis

Since the objective of the flight tests is the reduction of the takeoff distance that is covered from UC²AV it is important to conduct a sensitivity analysis to identify the factors and variables that affect the most the runway distance. A one-at-a-time type of sensitivity analysis is used to addresses takeoff parameter sensitivity relative to the point estimates chosen for the parameters held constant.

From Anderson’s book [79], an estimate for take-off distance is given by Equation (9.6).

\[ \ell_{T.O.} = \frac{1.44W_{T.O.}^2}{g\rho_\infty S C_{L_{max}} T} \]  \hspace{1cm} (9.6)

Assuming that all variables are constant except the takeoff weight \( W_{T.O.} \), which is increased from 3.5 kg to 4 kg (14.3% increment) Equation (9.6) becomes:
\[ \ell_{T.O.} = \text{Constant} \cdot W_{T.O.}^2 \]  

(9.7)

and with a \( W_{T.O.} \) increase of 14.3\% 

\[ \ell_{T.O.} = \text{Constant} \cdot (1.143)^2 = 1.306 \cdot \text{Constant} \]  

(9.8)

This is translated to a 30.6\% of runway distance increase. Following the same mathematical analysis the predicted takeoff distance is in a close agreement with what is recorded in the field.

Flight data are collected with the Anaconda conventional UAV (two different takeoff weights) and with the Anaconda with the NACA 0015 wing. The takeoff distance that is recorded and corresponds to the average of 5 takeoff flights is presented in Tables 9.1 and 9.2.

Table 9.1: Comparison between the predicted takeoff distance the flight data.

<table>
<thead>
<tr>
<th>Anaconda Conventional UAV</th>
</tr>
</thead>
<tbody>
<tr>
<td>Takeoff Weight</td>
</tr>
<tr>
<td>----------------</td>
</tr>
<tr>
<td>3.5 kg</td>
</tr>
<tr>
<td>4.0 kg</td>
</tr>
</tbody>
</table>
Table 9.2: Comparison between the predicted takeoff distance and the flight data.

<table>
<thead>
<tr>
<th>Takeoff Weight</th>
<th>Takeoff Distance Prediction</th>
<th>Takeoff Distance Covered</th>
</tr>
</thead>
<tbody>
<tr>
<td>3.7 kg</td>
<td>Using Sensitivity</td>
<td>88.2 meters</td>
</tr>
<tr>
<td>4.3 kg</td>
<td>Analysis: 118.8 meters</td>
<td>115.5 meters</td>
</tr>
</tbody>
</table>

Following the one-at-a-time sensitivity analysis method it is also estimated that a headwind which is 10% of the takeoff airspeed it is expected to reduce the takeoff distance approximately by 19%. However, a tailwind which is 10% of the takeoff airspeed is expected to increase the takeoff distance approximately by 21%.

9.2 Instrumentation

Instrumentation plays a critical role in validating the performance of a UAV during all flight envelopes. The key role of the instrumentation system is to track, characterize and validate the performance of the UAV during flight. Since the objective requires focus on the takeoff maneuver, the instrumentation that is described next, focuses on recording the required data to track the takeoff performance of the UC²AV.

To record accurately takeoff distance three cameras are positioned on the runway. Runway markers are placed along the runway, which can be observed from both the cameras and the observers to track the position of the airplane during takeoff. An ultrasonic distance sensor is placed on the belly of the fuselage (close to the front wheel) and tracks the distance from the ground during takeoff. The sensor reads the
distance from the runway and can track the exact point that the front wheel becomes airborne. Weather conditions (runway temperature, humidity, wind condition) are recorded using anemometers and sensors that are located on the runway. The barometric sensor (on-board the UC²AV) records temperature, humidity, pressure and altitude. The takeoff velocity is recorded from a pitot probe, which is located on the nose-tip of the aircraft. All the instrumentation and the position that they are located on the UC²AV are presented in Figure 9.3.

![Figure 9.2: The instrumentation that is on-board the UC²AV.](image)

A typical flight requires the following data channels: one data channel for time stamp (micro-controller); five data channels for the pilot’s inputs (RC receiver); one data channel for pitot sensor (raw data); three data channels (Yaw, Pitch, Roll) for inertial measurement unit (IMU) orientation; one data channel for temperature; one data channel for altitude (high range barometric sensor); one data channel for low range high accuracy altitude (ultrasonic distance sensor). All data are stored in on-board SanDisk memory card (32 GB). The data contains a record of the entire flight; however, maneuver markers (set by the pilot with a switch on a spare RC channel) indicate the areas of interest and allow for a quick review of data in the field.
A block diagram of the instrumentation system is seen in Figure 9.3. The measurement of the attitude angles, $\phi$, $\theta$ and $\psi$ (roll, pitch and yaw respectively), is performed with a VectorNav VN-100 IMU chip mounted on a development board. This sensor incorporates a 3-axis magnetometer, a 3-axis accelerometer and 3-axis gyroscope with extended Kalman filter. The IMU’s outputs include the aircraft attitude expressed as Euler angles or quaternions, linear accelerations, angular rates or magnetic local field. One pitot probe connected to a differential pressure sensor, is located at the front of the fuselage to measure true airspeed. To record the altitude and temperature a barometric pressure sensor (Bosch BMP085 transducer mounted on a Sparkfun breakout board) is used. An ultrasonic distance sensor located on the lower front part of the fuselage is used to locate the exact moment that the front wheel becomes airborne. Table 9.3 lists the sensors used on-board the $UC^2AV$ and their specifications.
Table 9.3: Instrumentation/Sensor Specifications for UC²AV.

<table>
<thead>
<tr>
<th>Component</th>
<th>Manufacturer</th>
<th>Part Number</th>
<th>Specifications</th>
</tr>
</thead>
<tbody>
<tr>
<td>Microprocessor</td>
<td>Arduino</td>
<td>Arduino</td>
<td>Microcontroller : Atmega2560</td>
</tr>
<tr>
<td></td>
<td></td>
<td>Mega 2560</td>
<td>Operating Voltage : 5V</td>
</tr>
<tr>
<td></td>
<td></td>
<td></td>
<td>Digital I/O Pins : 54 with 15 PWM pins</td>
</tr>
<tr>
<td></td>
<td></td>
<td></td>
<td>Analog Inputs : 16</td>
</tr>
<tr>
<td></td>
<td></td>
<td></td>
<td>Weight : 37 g</td>
</tr>
<tr>
<td>RC Transmitter/Receiver</td>
<td>Flysky</td>
<td>FS-i6</td>
<td>Frequency range : 2.405 to 2.475 GHz</td>
</tr>
<tr>
<td></td>
<td></td>
<td></td>
<td>Number of Channels : 6</td>
</tr>
<tr>
<td>Inertial Measurement Unit</td>
<td>Vectornav</td>
<td>VN-100</td>
<td>3-axis accel/gyro/mags. with on-board extended Kalman Filter</td>
</tr>
<tr>
<td></td>
<td></td>
<td></td>
<td>Gyro range : ± 2000 °/s, linearity &lt;0.1% FS</td>
</tr>
<tr>
<td></td>
<td></td>
<td></td>
<td>Accelerometer range : ± 16 g, linearity &lt;0.5 % FS</td>
</tr>
<tr>
<td></td>
<td></td>
<td></td>
<td>Overall weight : 37 g</td>
</tr>
<tr>
<td>Pitot Sensors</td>
<td>Freescale Semiconductor</td>
<td>MPXV7002DP</td>
<td>Pressure range : ± 2 kPa</td>
</tr>
<tr>
<td></td>
<td></td>
<td></td>
<td>Accuracy : 2% FS</td>
</tr>
<tr>
<td></td>
<td></td>
<td></td>
<td>Sensitivity : 1 V/kPa</td>
</tr>
<tr>
<td></td>
<td></td>
<td></td>
<td>Response time : 1ms</td>
</tr>
<tr>
<td>Barometric Sensor</td>
<td>Bosch Sensortec</td>
<td>BMP085</td>
<td>Pressure range : 30 to 110 kPa</td>
</tr>
<tr>
<td></td>
<td></td>
<td></td>
<td>RMS noise : 0.1 m</td>
</tr>
<tr>
<td></td>
<td></td>
<td></td>
<td>Weight : 10 g</td>
</tr>
<tr>
<td>Ultrasonic Distance Sensor</td>
<td>HC-SR04</td>
<td></td>
<td>Ultrasonic Frequency : 40 kHz</td>
</tr>
<tr>
<td></td>
<td></td>
<td></td>
<td>Resolution : 1 cm</td>
</tr>
<tr>
<td>Memory Module</td>
<td>Sparkfun OpenLog</td>
<td>Dev-09530</td>
<td>Baud rates : 2400 to 115200</td>
</tr>
</tbody>
</table>
Chapter 10

Flight Testing

The previous chapter illustrated the theoretical aspects of the takeoff performance and the instrumentation that is needed on-board to record the required data. This chapter presents in detail steps that are followed to prepare the unmanned aircraft for flight tests. The process of the data collection is presented followed by mathematical analysis to derive the aerodynamic parameters that are required for accurate takeoff distance calculations. At first, a data analysis for the first configuration (Anaconda platform with NACA 0015 wing) is conducted, to evaluate the on-board sensors. Then an analysis from collected data of the UC²AV flights is conducted and the effectiveness of CC during takeoff (runway distance reduction) is evaluated.

Before moving on the aircraft configuration, aircraft flight controls and the analysis of the results, it is important to state the limitations and the assumptions that are taken in this analysis. To calculate the analytical solution for the ground effect, it is assumed that since the weather in Denver, where the flight tests take place is dry, the value of the coefficient of resistance ($\mu$) is expected to be at the lower side of the range that is given from theory. Also the Oswald efficiency number, which is a
correction factor that represents the change in drag with lift of a three-dimensional wing as compared with an ideal wing having the same AR and an elliptical lift distribution, is taken to be equal to 0.95 since the wings that are tested are rectangular wings with zero leading and trailing edge sweep and no winglets. Also, calculating the average lift off velocity and the average density under the same weather conditions and since the standard deviation is low, it is another assumption that is made. The last assumption that is considered during flight testing is that the takeoff angle of attack is considered the same in all flight tests that are conducted under similar weather conditions.

10.1 Aircraft Controls & Data Collection

10.1.1 V-Tail Controls

A V-tail (Vee-tail) aircraft configuration, also known as butterfly tail is an unconventional arrangement of the tail control surfaces, which replaces the traditional fin and horizontal surfaces with two surfaces set in a V-shaped configuration. The chosen aircraft (Anaconda RMRC), has an inverted V-tail configuration, which combines the function of the elevators and rudder. The rear of each of the two surfaces are hinged and called ruddervators. The ruddervators provide the same control effect as conventional control surfaces, but through a complex control system that actuates simultaneously the control surfaces. A predefined setting for controlling V-Tail aircraft is set on the transmitter that mixes the rudder and elevator controls to achieve the coordinated movements of the two servos. However this setup influences the pitch control when the pilot corrects the yaw while the aircraft is still on the runway during the takeoff maneuver.
As mentioned in Section 9.1.2, the pilot’s technique can cause a greater variation in takeoff data than all other parameters combined and for that purpose, the Pulsed Width Modulation (PWM) pilot’s inputs for roll, pitch and yaw are recorded. PWM is the representation of the signals used to control servos and motors, which is a technique to control analog components using a digital microcontroller. A square wave signal is generated. A fast alternation of an up-down pattern simulates analog signals ranging from 0 V (off) to maximum voltage (on). The servos nominal voltage is 5 V and their update rate is 50 Hz, the basic frame period is therefore 20 ms. Tracking pilot’s inputs, leads to a repeatable takeoff maneuver process, where the collected data are checked to make sure they are not biased from the pilot’s takeoff technique. The data are stored on an SD-card and a MATLAB® code is used to plot those data. An example of the plotted data is shown in Figure 10.1. That procedure, even if it is conducted off-line helps to identify the influence of the pilot on pitch control and if it is found to be crucial, the flight test is repeated. Figure 10.2 shows the pitch angle that is recorded from the IMU and the the recorded data from the ultrasonic sensor. Both sensors are synchronized in time, which makes it possible to identify the exact point (time) that the front wheel becomes airborne.

Registered TM for The Mathworks, Inc
Figure 10.1: Bottom: Pilot input (roll, pitch, yaw and throttle). Top: The pitch and roll response and the readings of the three pitot probe sensors.

Figure 10.2: Top: The pitch angle recorded from the IMU. Bottom: The recorded data from the ultrasonic distance sensor during takeoff.
10.1.2 Aircraft Flight Controls

The flight controls configuration and the relationship of the microcontroller board (Arduino®) with the chosen I/O devices is illustrated in Figure 10.3. The radio transmitter (TX) is the SFlysky model, while the on-board receiver (RX) is the 6-channel R610 Spektrum DSM2, the frequency is 2.4 GHz. The sensors (pitot (V), barometric (z) and ultrasonic (ζ)) that are already illustrated in Section 9.2 transmit data to the Arduino® board. The outputs of the controller board goes to the aircraft motor and to the six servos controlling the actuators. Power is supplied by two 4-cell LiPo batteries with 5100 mAh, the motor regulator is a 80 Amp ESC (electric speed control) with switch mode BEC (battery eliminator circuit) by Tiger Motor and the propeller is a 15x4E. The servos and the propulsion system are the recommended parts from the Anaconda RMRC manufacturer.

Figure 10.3: Anaconda aircraft integrated with NACA0015 conventional wing flights controls data flowchart.

\(^2\)Registered TM for Creative Commons
10.1.3 The Airfield

The flight tests are performed at Miniature Aero Sportsters RC field. The facility includes a weather station and one paved runway of 120 meters, paved taxiways and pit area (Figure 10.4). The airfield is located at an altitude of 1587.760 m / 5209.185 feet above sea level. Flight tests are conducted below 400 feet above ground level because recreational use of airspace by model aircraft is covered by FAA, which generally limits operations to below 400 feet above ground level and away from airports and air traffic.

![Figure 10.4: Miniature Aero Sportsters: Remote Control (RC) field.](image)

10.2 Ground Effect

Before analyzing and evaluating the flight test data, it is important to calculate the ground effect on the aircraft. An aircraft acting in the presence of the ground (runway) has different aerodynamic properties than in cruise flight. The ground improves the efficiency of the aircraft by reducing the downwash at the wing and hence reducing the induced drag parameter. First, the analytical solution is calculated for the conventional UAV taking into consideration the wing specifications and from the collected data, the experimental ground lift coefficient ($C_{Lg}$) is calculated. The away-from-ground induced drag parameter (K) is given by Equation (10.1) and the
in-ground-effect induced drag parameter is affected by a correction factor $\phi$, which is given by Equation (10.2).

$$K = \frac{1}{\pi ARe}$$  \hspace{1cm} (10.1)

$$\phi = \frac{\left(\frac{16H_b}{b}\right)^2}{1 + \left(\frac{16H_b}{b}\right)^2}$$  \hspace{1cm} (10.2)

Using the wing specifications from Table 8.7 in Section 8.4.2 and using the Oswald factor ($e$) value of 0.95 for a rectangular wing configuration [80], it is derived induced drag parameter ($K$) is equal to 0.040 and the correcting factor ($\phi$) for that aircraft is equal to 0.844. Using Equation (10.3) it is calculated that the in-ground-effect induced drag parameter ($K_g$) is equal to 0.0338.

$$K_g = \phi K$$  \hspace{1cm} (10.3)

To calculate the ground lift coefficient for takeoff distance ($C_{Lg}$), Equation (10.4) is used. For the first configuration where the Anaconda platform is used with the conventional NACA 0015 wing (Figure 10.5) the ground lift coefficient is calculated to be 0.369. The coefficient of resistance ($\mu$) is taken to be equal to 0.025 since a dry asphalt runway is used (Section 9.1) [78]. The range that is given on the resistance coefficient is very wide ($0.02 < \mu < 0.05$) and given the fact that 0.05 corresponds to a wet runway, it is taken a value close to 0.02 since the flight tests are conducted in Denver, where usually the weather is dry during warm days and humidity is less
than 5 % with high temperatures. The testing days the weather was warm with
temperature higher than 75 F and humidity less than 10%.

\[ C_{L_g} = \frac{\mu}{2K_g} \]  

(10.4)

Figure 10.5: Anaconda with the conventional NACA 0015 wing integrated.

10.3 Flight Data Analysis

Flights with two different configurations are conducted. The first configuration
is the Anaconda fuselage with the conventional NACA 0015 wing which is a flapless
configuration. The second platform consists of the Anaconda fuselage but modified
with the ASU on-board and the duct at the belly of the aircraft. Integrated on that
platform is the CCW with the CC-system and the dual radius flaps with a maximum
deflection of 45 degrees. The first configuration is used to collect data and evaluate
the sensors. The data are collected and the aerodynamic parameters are calculated
and the results are compared with the analytical solutions.
10.3.1 Anaconda with NACA 0015 Wing Flight Data

This aircraft configuration weighs 3.7 kg (36.28 N) and the wing specifications can be seen in Table 8.7 in Section 8.4.2. The data (twenty takeoff maneuvers) that are summarized on Table 10.1 are taken on three different days. The weather conditions were almost the same and the data are categorized according to the wind conditions. In some cases the wind direction was not predictable and it was changing during the takeoff maneuver in a way that it was not possible to track. In reality, a crosswind will not influence the takeoff distance, but in some cases a crosswind was interchangeable with a headwind or tailwind and that is depicted on the data. As it is explained in Section 8.7, the conventional NACA 0015 wing is initially designed and built to test the wing structure in real conditions and flight test. Also, this configuration (Anaconda integrated with the NACA 0015) is used to evaluate the data that are recorded. To evaluate the data, aerodynamic coefficients are first calculated analytically using the corresponding equations and then are compared with the recorded data.

The definition of calm winds is winds that are moving with speeds less than 1 knot (0.5 m/s). Focusing on the twelve takeoff flights that are conducted under calm winds, the average liftoff velocity ($V_{L.O.\text{Avg}}$) is equal to $19.575 \pm 0.95$ m/s. The average velocity is calculated since the flight tests are conducted under the same weather conditions and the variation of density is small ($\rho_{\text{Avg}} = 1.002 \pm 0.0097$ kg/m$^3$).

During the takeoff maneuver and until the aircraft becomes airborne, the lift force is increasing as the aircraft speed is increasing. At the point that the aircraft becomes airborne the lift force becomes equal and higher than the aircraft’s takeoff weight. Using Equation (10.5) and solving for $C_{Lg}$ it is derived that the ground lift
coefficient is found to be $\geq 0.3823$. Taking into consideration the analytical value of the ground lift coefficient (0.369) the percentage error is equal to 3.5%. The error depends on many parameters but mainly assumptions that are considered to derive both the analytical and experimental ground lift coefficient values.

$$L \geq W \implies \frac{1}{2} \rho V_{L.O., Avg}^2 C_{Lg} S \geq W \quad (10.5)$$

<table>
<thead>
<tr>
<th>Flight Number</th>
<th>Air Density ($\text{kg/m}^3$)</th>
<th>Wind Condition</th>
<th>Wind Speed (m/s)</th>
<th>Liftoff Time (s)</th>
<th>Takeoff Time (s)</th>
<th>Ground Distance (f, m)</th>
<th>Air Distance (f, m)</th>
<th>Takeoff Distance (f, m)</th>
<th>Liftoff Velocity (m/s)</th>
<th>$C_{Lg}$</th>
</tr>
</thead>
<tbody>
<tr>
<td>1</td>
<td>1.050</td>
<td>Crosswind</td>
<td>1.0</td>
<td>8.4</td>
<td>10.6</td>
<td>90</td>
<td>44.26</td>
<td>134.26</td>
<td>20.2</td>
<td>0.356</td>
</tr>
<tr>
<td>2</td>
<td>1.010</td>
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<td>90</td>
<td>54.94</td>
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<td>-</td>
<td>7.75</td>
<td>10.73</td>
<td>84</td>
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<td>146.91</td>
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<td>-</td>
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<td>11.5</td>
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<td>11.25</td>
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<td>-</td>
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<td>11.3</td>
<td>84</td>
<td>51.44</td>
<td>135.14</td>
<td>18.7</td>
<td>0.371</td>
</tr>
<tr>
<td>19</td>
<td>0.997</td>
<td>Calm</td>
<td>-</td>
<td>8.9</td>
<td>11.4</td>
<td>96</td>
<td>52.82</td>
<td>148.82</td>
<td>17.9</td>
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<tr>
<td>20</td>
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<td>96</td>
<td>49.47</td>
<td>145.47</td>
<td>19.9</td>
<td>0.372</td>
</tr>
</tbody>
</table>

In general, aircraft use the flow of wind over the wings to generate lift to be able to fly. However, by taking off into the wind (headwind) the aircraft lifts off sooner and this will result in a lower ground speed and therefore a shorter takeoff run for the aircraft to become airborne. As it is mentioned in Section 9.1.3, a headwind which is 10% of the takeoff airspeed is expected to reduce the takeoff distance approximately by 19% in theory and that can be also seen from the flight data that are collected under a headwind condition. In fact, the average liftoff velocity is given from twelve
runs, which are conducted under calm wind conditions \( \left( V_{L.O., Avg} = 19.575 \text{ m/s} \right) \). The calculated velocity is then compared to the velocity of the three headwind cases and the results are presented in Table 10.2. As ground distance during takeoff the average of the runs under calm wind conditions is taken and that is equal to \( \ell_{L.O., Avg} = 88.75 \pm 7.2 \) m. It is shown that the runway distance the UAV covers before it becomes airborne is affected by a headwind and is reduced by almost the expected percentage. The flight test data are in agreement with the analytical solutions.

Table 10.2: Theoretical and flight test data comparison.

<table>
<thead>
<tr>
<th>Flight Number</th>
<th>Wind Condition</th>
<th>Wind Speed (m/s)</th>
<th>Ground Speed (m/s)</th>
<th>( \lambda ) (%)</th>
<th>( \ell_{L.O., Avg} ) (m)</th>
<th>( \ell_{L.O} ) (m)</th>
<th>Runway Reduction (%)</th>
</tr>
</thead>
<tbody>
<tr>
<td>6</td>
<td>Headwind</td>
<td>1</td>
<td>19.575</td>
<td>5.1</td>
<td>88.75</td>
<td>75</td>
<td>15.5</td>
</tr>
<tr>
<td>7</td>
<td>Headwind</td>
<td>1.5</td>
<td>19.575</td>
<td>7.7</td>
<td>88.75</td>
<td>72</td>
<td>18.9</td>
</tr>
<tr>
<td>8</td>
<td>Headwind</td>
<td>2.2</td>
<td>19.575</td>
<td>11.2</td>
<td>88.75</td>
<td>66</td>
<td>25.6</td>
</tr>
</tbody>
</table>

Since the main objective of the flight testing is the reduction of the runway distance, focus is given on accurate measurements on the runway distance that is covered during the takeoff maneuver under different wind conditions. Aside from the observers, video cameras and the dashed line indicators on the runway, the ultrasonic sensor indicated the exact time that the front wheel becomes airborne. The runway distance during the ground phase is recorded and presented in Table 10.1. The distance that is covered by the aircraft since it became airborne until the point that reached a 50 ft (15 meters) distance from the ground it is calculated based on the data that are recorded from the ultrasonic sensor, the pitot probe and the barometric sensor. The sampling rate of the data is 6.6 Hz (sampled every 0.15 seconds). To calculate the distance that corresponds to the the takeoff climb, Equations (10.6) and are (10.7) used.

134
\[ \ell_{\text{air}} = \sum V_{\infty,x} \cdot dt \]  \hspace{1cm} (10.6)

where \( V_{\infty,x} \) is given by:

\[ V_{\infty,x} = V_{\infty} \cdot \cos(\alpha) \]  \hspace{1cm} (10.7)

The total takeoff distance (\( \ell_{\text{T.O.}} \)) is calculated by Equation (10.8).

\[ \ell_{\text{T.O.}} = \ell_{\text{L.O.}} + \ell_{\text{air}} \]  \hspace{1cm} (10.8)

Under calm weather conditions the average takeoff distance \( \ell_{\text{T.O.\ Avg}} \) is equal to 145.6 \( \pm \) 6.5 m. From the data on Table 10.1 it is derived that the total takeoff distance (\( \ell_{\text{T.O.}} \)) under calm weather conditions, using this configuration, cannot exceed 152.08 m and cannot be less than 135.14 m. This analysis indicates that the experimental data are in agreement with the analytical solutions and the sensors and the methods that are used to calculate the aerodynamic forces and aircraft performance during takeoff are reliable. The next section presents flight data from the \( \text{UC}^2\text{AV} \) configuration.

10.3.2 \( \text{UC}^2\text{AV} \) Flight Data

10.3.3 Flight Controls

The flight controls configuration and the relationship of the microcontroller board (Arduino\textsuperscript{\textregistered}) with the chosen I/O devices stays the same as in Section 10.1.2 but in
addition a second radio transmitter (Spektrum® DX8³) with an on-board receiver is used to control the ASU and the dual radius flaps as Figure 10.6 illustrates.

Figure 10.6: The UC²AV flight controls data flowchart.

10.3.4 Preflight Data Analysis

Before the flight tests are conducted, the flow uniformity needs to be tested and evaluated in order to ensure that the activation of flow in the CCW will not destabilize the aircraft during flight with uneven effects across the wing span. Also, the momentum coefficient values for half and full RPM speeds of the ASU need to be measured and recorded. As it is previously mentioned (Section 5) two plenum designs are placed inside the CCW at a distance of 500 mm away (Figure 10.7). The Anaconda has a pusher configuration (as opposed to a tractor configuration), where the thrust has a pusher configuration. A propeller on the aircraft operates in a non-uniform flow field produced by the aircraft components and axial, vertical

³Registered TM of Horizon Hobby, Inc
and horizontal velocity increments are produced upstream and downstream of the wing. In general, the vortex wakes that are created from the propeller, tend to deform and roll up which produces a so-called slipstream tube with strong gradients in various flow quantities both in streamwise and radial direction [81]. To avoid affecting the CC performance and stability and control issues that may affect the overall performance of the aircraft, both plenum designs are placed at a distance of 250 mm from the wing root.

![Figure 10.7: The UC²AV’s CCW Solidworks design.](image)

### 10.3.5 Flow uniformity

CC control is applied on 40% of the span and two plena are positioned symmetrically on the left and right semi span. The flow uniformity across the span is evaluated and the plot in Figure 10.8 presents the performance of each plenum and the deviation of the velocity from the average line (shown in black) at two cases (half RPM and full RPM applied) is shown. The pitot probe (Section 6.3) is used to collect data at 5 points along the slot exit of both plenum designs. The experimental
Figure 10.8: Experimental results representing the performance of the $V_{jet}$ at the slot across the span on each plenum for different RPM values. The deviation of the $V_{jet}$ from the average line (shown in black) at both RPM value is shown.

The procedure is repeated three times each for two different RPM values since high flow uniformity performance and repeatability before the system is ready to be integrated on-board the UC²AV is required. As Figure 10.8 demonstrates, only in one point but in both cases (both RPM cases) on the plenum located at the left side of the wing the required performance is not achieved. No significant flow abnormalities causing non-uniformity at the slot are detected and it is concluded that the non-uniformity of the slot height (average slot height ($h=0.4\pm0.05$ mm) may have caused during the fabrication introducing an error to the CC system. Further investigation is being carried out to increase precision in the fabrication process.

10.3.6 Momentum Coefficient of Blowing

The momentum coefficient $C_\mu$ is a critical parameter in understanding the efficiency of blowing using CC and the actual momentum coefficient at the jet is already defined in Section 3.2.2, Equation (3.7). For flow uniformity testing (preflight tests) two momentum coefficient values are tested, which correspond to half power of the motor that runs the ASU ($C_\mu$ equals to 0.13) and full power ($C_\mu$ equals to 0.165).
For flight testing only the full power is used. For momentum coefficient of blowing ($C_\mu$) calculations, a free stream velocity of 19.575 m/s is used, which corresponds to the average liftoff velocity during take off ($V_{L.O.,Avg}$). However, the ASU is turned on before the takeoff maneuver starts, therefore it is expected that at the time of takeoff the actual $C_\mu$ is bigger than the estimated value.

10.3.7 Flight Data Analysis

Seven experimental flight tests using the CC system integrated on the UC²AV are conducted. All flights are conducted the same day thus the weather conditions are the same for all seven flights. After the first three flights, a malfunction on the ultrasonic sensor and the pitot probe did not allow to gather all the required information for calculating the lift coefficient on the ground. The data collected after that malfunction are analyzed off-line with the use of videos, the observers and the line indicators. Figure 10.9 shows the pilot input at flight 2 of Table 10.3. The pilot corrects the pitch angle after the aircraft is up and away from the runway. Throttle is at the maximum since the beginning of the maneuver.
Figure 10.9: Top: The pitch and roll response and the readings of the three pitot probe sensors. Bottom: Pilot input (Roll, Pitch, Yaw and Throttle).

The takeoff weight of the UC²AV is 4.7 kg (46.1 N). The dual radius flap deflection is in all cases 30°. In all flight tests, the ASU is operated at maximum RPMs ($C_\mu = 0.165$) and it used only during the takeoff maneuver. From the data that are presented in Table 10.3 it is observed that CC is effective and can reduce the ground distance up to 50%. A comparison between the first and second flight shows that the lift coefficient on the ground, using the ASU, it is enhanced by 93%.

Table 10.3: CC Flight test data.

<table>
<thead>
<tr>
<th>Flight Number</th>
<th>ASU</th>
<th>Dual Radius Flap Deflection (°)</th>
<th>Air Density ($kg/m^3$)</th>
<th>Wind Condition</th>
<th>Liftoff Time (s)</th>
<th>Takeoff Time (s)</th>
<th>Ground Distance (m)</th>
<th>Air Distance (m)</th>
<th>Takeoff Distance (m)</th>
<th>Liftoff Velocity (m/s)</th>
<th>$C_{l_{off}}$</th>
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</thead>
<tbody>
<tr>
<td>1</td>
<td>Off</td>
<td>30</td>
<td>0.992</td>
<td>Calm</td>
<td>6.94</td>
<td>15.31</td>
<td>102</td>
<td>78</td>
<td>160</td>
<td>18.2</td>
<td>0.57</td>
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<td>2</td>
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<td>Calm</td>
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<td>49</td>
<td>115</td>
<td>103</td>
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<td>1.10</td>
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<td>15.67</td>
<td>54</td>
<td>55</td>
<td>109</td>
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<td>80</td>
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<td>7</td>
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<td>42</td>
<td>-</td>
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A closer look at the ground distance (Table 10.3) shows that the lift enhancement achieved with CC is translated to a runway distance reduction of 53%. The takeoff air distance is not reduced and that mainly has to do with the lower speed that the aircraft is flying up and away the runway. The aircraft leaves the runway with a smaller pitch angle and the climb is smooth mainly due to the fact that pitch moment is introduced as an effect of CC. The UAV is more stable and the angle of attack is increased smoothly during takeoff and as a result, it reaches gradually the 50 ft ceiling for takeoff while it is flying at a lower speed. Data for liftoff velocity are not collected for all seven flights, however, it is observed that in all cases CC is effective and when it is used, the ground distance is reduced significantly.

The importance of the results needs to be highlighted because based on the literature review, published flight data confirming that CC can be applied on unmanned aircraft and achieve short takeoff envelopes do not exist. Additional tests need to be carried out in order to confirm repeatability at different days and weather conditions. The collected data are sufficient to confirm that CC can be applied on UAVs. Significant runway reduction can be achieved with minimum power penalties and low blowing rates.
Chapter 11

Conclusions & Future Work

11.1 Conclusions

In this research work, the efficiency of CC on lift enhancement during takeoff maneuvers on small-scale unmanned aircraft is evaluated. Tests in a controlled laboratory environment (wind tunnel testing) and in real flight are conducted. The present dissertation establishes the foundations and technical details of a comprehensive and verifiable theoretical, computational and experimental methodology for design and development of UAV with enhanced operability, functionality, and ability to execute complex missions.

2-D and 3-D low Reynolds number wind tunnel tests on different airfoil shapes are conducted. Two-dimensional lift and drag data are collected and incremental lift coefficient and lift-to-drag ratio results are presented to support CC blowing effects on the qualified CCW configurations. The results indicate that upper slot blowing can be effective for lift enhancement even at low blowing rates. The parameters varied in this study showed that the largest lift augmentation is achieved with the
shorter dual radius flap (DRF10). On the other hand, the longer flap experienced lower lift enhancement that resulted at lower lift-to-drag ratios especially at $\delta_f = 0^\circ$. The CCW is configured to test CC strategies at realistic flight Reynolds numbers. CFD and experimental tests of a plenum for flow uniformity show that the plenum is capable of distributing air evenly across the span. A detailed study on an air supply unit capable of supplying the required mass flow for CC is conducted. CFD and experimental results of a complete CC system that distributes the air with reduced losses evenly on the plenum designs is presented.

A CC system that consists of the ADS, the ASU and two plena is designed, built, integrated on the UC2AV and experimentally tested. The CC system, weighs 650 g (1.433 lbs) and it provides sufficient mass flow for upper slot TE blowing on the 1/3 of the span of the aircraft. Using this system on-board the UC2AV it is demonstrated that sufficient lift increment can be achieved capable to reduce the runway distance up to 50% during takeoff. The use of a more powerful motor that provides higher thrust to achieve a shorter takeoff distance may be considered. However, the propeller plays an important role since its purpose is to convert power into thrust. The relationship between power, thrust and speed is not linear and if power and propeller efficiency are kept constant, then propeller thrust decreases as true airspeed increases. Generally, a bigger propeller has a better efficiency than a small one but a small propeller can keep the efficiency in a wider speed range. The aerodynamic drag increases with the square of speed, thus, it takes eight times the power to double the airspeed of a given configuration [82]. Considering a big propeller on an aircraft, the acceleration will be better but the aircraft will not be able to fly faster and the power consumption will be higher. The motor and the propeller that are used, are recommended by the manufacture of the Anaconda RMRC. The use of a more powerful motor with the
equivalent propeller on-board may reduce the runway distance during takeoff, but
the power consumption will be increased not only during takeoff but also during the
rest of the flight.

The obtained results demonstrate that the use of Coandă-based CC can be ap-
plied on small-scale unmanned aircraft and regardless of the space and power lim-
itations, significant lift enhancement can be achieved. A series of flight tests on a
platform with a conventional configuration proved that the sensors that are used to
track the aerodynamic parameters are reliable. The data that are collected from this
configuration are compared with theoretical results (using certain assumptions) are
found to be in a close agreement.

In conclusion, the UC²AV has potential applications in commercial airliners, busi-
ness jets, cargo planes, PAVs (personal aerial vehicles) and regional transports be-
cause equipping these aircraft with cruise-efficient high-lift devices can enhance air-
port options, give the user more valid runway choices at existing airports, and help
alleviate environmental noise problems near airports by allowing steeper climb-outs.
The technology developed in this project directly supports all the above goals.

11.2 Future Work

The research performed up to this point can be advanced along three directions:
hardware modifications on the UC²AV, CCW modifications and wind tunnel tests
and morphing flap integration on the CCW. The aircraft developed throughout this
research is suitable for demonstrating CC concepts. However, since it is the first
prototype during the implementation process, certain hardware issues that have to
do with the ASU integration can be avoided. The ASU needs to be modular to
avoid malfunctions and each part must be easier to replace. Numerous wind tunnel tests are conducted to identify the wing configuration that gives the maximum lift enhancement. However, future developments to the present work should target more wind tunnel 2-D and 3-D tests. This involves the design and development more CCW wind tunnel models along with more designs of dual radius flaps with different radius ratios \((r_2/r_1)\).

A flap system employs a sharp trailing edge which increases the jet thrust recovery during deployment and operation and greatly reduces the pressure drag in cruise configuration. Of the existing CC flap configurations that have been developed, the dual radius is the configuration further investigated. The dual radius flap first acquires the benefit of the smaller radius by turning the slot flow over a larger angle in a smaller chord-wise distance, which occurs because of the high momentum the flow still carries from being ejected from the slot. Then the larger radius keeps the flow attached as it travels along the flap and its high momentum energy is reduced [83]. The flap geometry can be optimized and keep a high separation angle. However, it should be noted that although flap geometry may be optimized to maintain a high separation angle and to improve performance during a specific flight envelope, however, flap geometry, in general, will not perform at its maximum efficiency at all flight envelopes. To overcome this limitation, wing structures equipped with morphing high lift devices will allow for the wing to adapt its shape smoothly under different loading conditions and to achieve near-optimal lift and drag profiles in different phases of the flight, leading to enhanced aerodynamic performance and, consequently, to considerable fuel savings. Under this concept, the flap is shaped, deflected, or deformed to respond to sudden changing conditions within the (unmanned) aircraft’s flight envelope. Shaping produces much more complex modes
of actuation than those achieved using conventional control surfaces. Combining a morphing flap with the dual radius flap geometry will allow for the ejected from the slot air to remain attached at all times, resulting in optimum lift enhancement during flight.
Bibliography


151


155