Development of Flexible-Rib Morphing Wing System

by

Yue Wang

A thesis submitted in conformity with the requirements for the Degree of Master of Applied Science, Department of Mechanical and Industrial Engineering University of Toronto

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University of Toronto

2015

Abstract

This study is concerned with the development of a novel morphing wing system for use in relatively small (10 kg) Unmanned Aerial Vehicles (UAVs). To achieve improved flight performance with limited weight penalty, camber-adjustable morphing wing strategy was adopted and realized via the use of Flexible Rib Morphing Wing System (FRMWS). Detailed morphing wing structure was designed using a novel flexible-rib mechanism which adheres to the safety requirements specified by FAR23 standards. Finite Element structural and aerodynamic analyses were conducted to validate the advantages and suitability of the design. Furthermore, wind tunnel and actual load tests were conducted in order to validate the new design. In addition, a test plane was refitted and flight test were conducted. The result of the flight test demonstrated the improved performance and the reliability of the new morphing wing system.
Acknowledgements

I offer my sincere appreciation and gratitude to Professor S.A. Meguid for his continued technical guidance and ideas that helped to elevate the thesis to its current level and for proof reading of the thesis. I also wish to thank him for providing financial support during my Master’s program of studies.

I would also like to thank Professor Yu Su and my colleague Pieter Verberne for his input to this thesis and the sponsors of the current research who wish to remain anonymous for their input and financial support.

It has also been a great pleasure to work with all of my fellow researchers in the Mechanics and Aerospace Design Laboratory (MADL). I would like to thank all the colleagues in the lab for their kind help. Also I would like to specially thank Yasir Malang for his help during the flight test with his professional flying skill.

Finally, I would like to offer my gratitude to my parents and my friends. It is difficult to express how much I appreciate their understanding and unlimited support during the different stages of my studies. They are always my powerful backing for my final success.
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### Notations

<table>
<thead>
<tr>
<th>Symbol</th>
<th>Description</th>
</tr>
</thead>
<tbody>
<tr>
<td>$C_l$</td>
<td>Coefficient of lift (airfoil)</td>
</tr>
<tr>
<td>$C_d$</td>
<td>Coefficient of drag (airfoil)</td>
</tr>
<tr>
<td>$Re$</td>
<td>Reynolds Number</td>
</tr>
<tr>
<td>$V$</td>
<td>Average flow speed</td>
</tr>
<tr>
<td>$l$</td>
<td>Characteristic length</td>
</tr>
<tr>
<td>$ν$</td>
<td>Kinematic viscosity</td>
</tr>
<tr>
<td>$y^+$</td>
<td>Non-dimensional wall distance</td>
</tr>
<tr>
<td>$y$</td>
<td>Distance from the wall to the cell center</td>
</tr>
<tr>
<td>$μ$</td>
<td>Molecular viscosity</td>
</tr>
<tr>
<td>$ρ$</td>
<td>Air density</td>
</tr>
<tr>
<td>$τ_w$</td>
<td>Wall shear stress</td>
</tr>
<tr>
<td>$δ$</td>
<td>Total thickness of boundary layer</td>
</tr>
<tr>
<td>$x$</td>
<td>Coordinate of the position on the board</td>
</tr>
<tr>
<td>$E_{1,2,3}$</td>
<td>Young’s modulus of the plywood</td>
</tr>
<tr>
<td>$ν_{12,23,13}$</td>
<td>Poisson’s ratio of the plywood</td>
</tr>
<tr>
<td>$G_{12,13,23}$</td>
<td>Shear modulus of the plywood</td>
</tr>
<tr>
<td>$F_d$</td>
<td>Load undertaken by each rocker</td>
</tr>
<tr>
<td>$F_{s1}$</td>
<td>Driving force on the rocker</td>
</tr>
<tr>
<td>$T$</td>
<td>Torsion needed on the rocker</td>
</tr>
<tr>
<td>$C_{ln}$</td>
<td>Slope of coefficient of lift with angle of attack (airfoil)</td>
</tr>
<tr>
<td>$A$</td>
<td>Aspect ratio</td>
</tr>
<tr>
<td>$M$</td>
<td>Mach number</td>
</tr>
<tr>
<td>$S_{\text{exposed}}$</td>
<td>Area of exposed wing planform</td>
</tr>
<tr>
<td>$S_{\text{ref}}$</td>
<td>Area of reference wing planform</td>
</tr>
<tr>
<td>$F$</td>
<td>Fuselage lift factor</td>
</tr>
<tr>
<td>$Λ_{\text{maxt}}$</td>
<td>Sweep angle of the wing at the thickest airfoil position</td>
</tr>
<tr>
<td>$C_L$</td>
<td>Coefficient of lift (wing)</td>
</tr>
<tr>
<td>$C_D$</td>
<td>Coefficient of drag (wing)</td>
</tr>
<tr>
<td>$C_{d0}$</td>
<td>Coefficient of parasite drag (airfoil)</td>
</tr>
<tr>
<td>Symbol</td>
<td>Description</td>
</tr>
<tr>
<td>--------</td>
<td>-----------------------------------------------</td>
</tr>
<tr>
<td>$C_{D_0}$</td>
<td>Coefficient of parasite drag (wing)</td>
</tr>
<tr>
<td>$C_{D_i}$</td>
<td>Coefficient of induced drag (wing)</td>
</tr>
<tr>
<td>$e$</td>
<td>Span efficiency factor</td>
</tr>
<tr>
<td>$Q$</td>
<td>Inviscid index of span efficiency factor</td>
</tr>
<tr>
<td>$u$</td>
<td>Calibration factor of $Q$</td>
</tr>
<tr>
<td>$s$</td>
<td>Index of extra lift-dependent drag</td>
</tr>
<tr>
<td>$d_r$</td>
<td>Fuselage diameter</td>
</tr>
<tr>
<td>$b$</td>
<td>Wing span</td>
</tr>
<tr>
<td>$P$</td>
<td>Viscous index of span efficiency factor</td>
</tr>
<tr>
<td>$K$</td>
<td>Test-related index of $P$</td>
</tr>
</tbody>
</table>
## Abbreviations

<table>
<thead>
<tr>
<th>Abbreviation</th>
<th>Description</th>
</tr>
</thead>
<tbody>
<tr>
<td>FRMWS</td>
<td>Flexible-rib morphing wing system</td>
</tr>
<tr>
<td>UAV</td>
<td>Unmanned aerial vehicle</td>
</tr>
<tr>
<td>SMA</td>
<td>Shape memory alloy</td>
</tr>
<tr>
<td>FAR23</td>
<td>Federal aviation regulation Part 23</td>
</tr>
<tr>
<td>CFD</td>
<td>Computational fluid dynamics</td>
</tr>
<tr>
<td>FEM</td>
<td>Finite element method</td>
</tr>
<tr>
<td>SMP</td>
<td>Shape memory polymer</td>
</tr>
<tr>
<td>SMC</td>
<td>Shape memory composite</td>
</tr>
<tr>
<td>PBP</td>
<td>Post-buckled Pre-compressed</td>
</tr>
<tr>
<td>MFC</td>
<td>Macro fiber composite</td>
</tr>
<tr>
<td>RSTA</td>
<td>Reconnaissance, Surveillance and Target Acquisition</td>
</tr>
<tr>
<td>CAD</td>
<td>Computer-aid design</td>
</tr>
<tr>
<td>AOA</td>
<td>Angle of attack</td>
</tr>
</tbody>
</table>
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Chapter 1
Introduction

1.1 Morphing Wing Concept and Problem Statement

Research in morphing wing has been initiated as a result of observing flying birds. The high flexibility of the bird’s wing allows its configuration to be widely adjusted based on its flying mission requirements. Taking the bald eagle as an example, when it glides in non-turbulent its wings stretch widely in order to attain higher lift-drag ratio. In contrast, the wings fold substantially during diving in order to reduce drag. The comparison of the flying patterns of the bald eagle is shown in Figure 1.1 [1][2].

![Figure 1.1 Different flight configuration of bald eagle](image)

Both the demands of multi-tasking Unmanned aerial vehicle (UAV) system [3] and the flying pattern of birds led to the exploitation of the morphing wing concept. Accordingly, morphing wing can be defined as the design and development of a flexible wing system that can adjust its configuration during varied flight missions in order to attain the best flying performance for their varied missions.

In comparison with fixed wing designs, the advantages of morphing are numerous. It can increase the lift-to-drag ratio of the wing during both high-speed and low-speed cruise, increase the maneuverability of the plane, even expand the plane envelop [4]. Specifically, morphing wing can offer the following[5]: improved aircraft performance that leads to expansion the flight envelope, replacing conventional control surfaces for flight control thus reducing weight and design complexities and provide stealth capability [6], reducing parasite drag and vibration and
flutter [7][8]. In addition, the control of the local flow by making small morphing adjustments to the wing surface can also help improve the aerodynamic force distribution [9].

Due to the strict rules off flight vehicle design (such as FAR23 standard) and the restrictions of flexible actuating material, very few realistic solutions have been developed for morphing wing application that have been devised for UAVs. It is meaningful to develop a solution of morphing wing application for small UAVs.

1.2 Research Objectives

The objective of this study is to develop a novel morphing wing solution to be adopted in wing system. Specifically, the aims of the study are to:

1. develop novel conceptual design of morphing wing systems in order to improve the flight performance of UAV without incurring large weight penalty,
2. carry out structural and aerodynamic analyse to study the effect of the new morphing wing design upon the integrity and performance of the morphed wing and the adherence to aircraft design standards such as FAR23,
3. design, build and test a functional prototype with the new morphing wing system to examine its function, reliability, and weight penalty, and
4. install the designed morphing wing to a small UAV and conduct the necessary flight test to test the reliability and the performance of the new design.

1.3 Method of Approach

The developed morphing wing concept should meet several requirements: flight performance, structural integrity, response time, weight penalty and cost effectiveness.

Aerodynamic and structural analysis will be conducted during the design process. Aerodynamic analysis will be performed using the commercial code ANSYS (Fluent). The result of the analysis is used to demonstrate the improvement brought about by the newly devised morphing wing system and to provide the design load for the structural analysis. Structural analysis will also be conducted using ANSYS with adopted loading factors specified by FAR23 standards for
aircraft design. In this work, the loading factor will be determined based on the characteristics of the refitted model plane.

An actual testing wing will be designed and manufactured. It will be tested under static condition and during the flight test. A test UAV will be refitted and equipped with the morphing wing for all the tests in order to demonstrate the applicability of the new morphing wing system.

1.4 Layout of Thesis

The thesis is divided into six chapters. Chapter One justifies the undertaking of the study and outlines the adopted methodology. Chapter Two presents a critical literature review of the existing projects of morphing approach prototypes. Chapter Three relates to the concept development including basic information of the UAV and decision matrix of morphing method. Chapter Four presents the development and analysis process of the Flexible-Rib-Morphing-Wing-System (FRMWS), including the morphing approach, the aerodynamic CFD analysis, the mechanism performance, the wing structure design and the structural analysis. Chapter Five describes the functional prototype manufacturing and testing, including the manufacturing, the wind tunnel test, the static load ground test and the flight test. Chapter Six concludes the work and presents areas for possible future exploration.
Chapter 2
Literature Review

2.1 History of Morphing Wing

The very first idea of morphing in a real plane can be retrospect to Wright Brother’s plane in 1903. The Flyer, which is the first powered flight, has been equipped with a twisting wing for rolling control [10]. Similar sweeping-wing system is still equipped to some modern flights and bomber aircrafts, such as Northrop Grumman F-14, Rockwell B-1, Mikoyan Mig-23 and Tupolev Tu-22.

However, the disadvantage of the existing designs is also obvious. These designs suffer from high weight penalty or incapable of meeting the requirements of the entire flying envelope. Hence, the sweeping-wing design has been seldom seen in modern aircrafts. Nowadays, many planes with sweeping wing have retired and most high-lift devices are only allowed to be used during takeoff and landing process.

2.2 Classifications of the Morphing Wing

Research has been carried out to study different aspects of morphing wing systems. To keep it simplified, the studied topic of morphing wings can be divided into the shape parameters (what to morph), the benefits (why to morph), and the realization of morphing wings (how to morph) [11].

The wing shape parameters can be classified as in-plane parameters or out-of-plane parameters, following a similar classification scheme summarized in Sofla and Meguid’s recent review [12]. The in-plane parameters refer to geometries pertaining to the X-Y plane and the out-of-plane parameters are geometries that involve Z-direction changes. This classification for shape morphing is outlined in Figure 2.1.
2.3 Current Research

Current research in wing morphing has focused on the requirement of high-power density actuators, the structural flexibility, flexible skins and control development [13].

The planform adjustments of the wing include the span change, chord length change and the sweeping angle change. Most of the adjustments can directly lead to the change of the wing configuration and wing area, resulting in a change in lift force, drag force, span efficiency factor and the maneuverability of the aircraft.

A variety of concept related to the advantages and optimization of the wing-planform change morphing wing have been discussed in [14-19]. In addition, several prototypes have been developed to investigate the performance of the adaptive structure, as well as the optimization of the mechanism. Reed et al. [20][21] developed a morphing wing structure with adjustable chord by using shape memory polymer (SMP), or shape memory composite (SMC). In the work of Smita Bharti et al. [22] a design of the internal wing structure using tendon actuated cellular
mechanism was presented. Filippo Mattioni et al. [23] proposed a similar approach similar to Bharti’s work by using bistable plate structures to achieve the beam bending during the sweeping. In addition, some researches of wing-planform change have been moved to actual applications on UAVs [24-27].

Wing area and aspect ratio are not changed during the out-of-plane transformation. The aerodynamic properties are changed through the change in the cross-section area of the wing such as camber and angle of attack. Compared with the wing planform alternation, the deformation in this category is much smaller. It leads to a reduction of actuator usage and the structural weight, leading to a decrease in structural complexity.

A variety of theoretical studies concerned with out-of-plane deformation morphing have also been conducted [28-30]. A large number of prototypes have been proposed with the use of smart materials such as shape memory alloy (SMA) and piezo-electric material [31-41]. Specifically, some projects have progressed to the test plane stage. Roelof Vos et al. [42][43] used post-buckled pre-compressed (PBP) piezoelectric bender actuators in a deformable wing structure to operate the camber and thereby conduct the roll control on a subscale UAV. Onur Blight et al. [44] made a solid-state control surface for a small novel unmanned aircraft. They use Macro-Fiber Composite (MFC) for the actuator. His research demonstrated the potential of morphing wing flight and provided insight into the major challenges associated with this technology.

Inflatable wing is a special wing structure of morphing and it can also be treated as a special case of out-of-plane transformation. It offers a new way of approaching morphing with soft wing structure. Some projects have concentrated on type of design as demonstrated in Refs. [45-47].
Chapter 3
Morphing Wing Conceptual Design

3.1 Specification of Selected UAV

According to the reviewed work in Chapter 2, it is noticed that different morphing approaches may suite the design of UAVs in their own ways with different dimensions and configurations. Before setting the approach for the morphing strategy in this work, specification of UAVs should be determined in advance.

Considering the application range of the morphing wing, also with the difficulties of design and manufacture in this project, the focus of the project is on the morphing of UAV for Reconnaissance, Surveillance and Target Acquisition (RSTA) [48].

The specification of the test plane can be seen as the baseline for the design of the morphing scheme. The primary specifications of the test plane and morphing wing are listed in Table 3.1.

<table>
<thead>
<tr>
<th>Length (m)</th>
<th>Wingspan (m)</th>
<th>Max Weight (kg)</th>
<th>Engine type</th>
<th>Airspeed (km/h)</th>
<th>Cruise altitude (m)</th>
</tr>
</thead>
<tbody>
<tr>
<td>1.5-2</td>
<td>1.5-3</td>
<td>Less than 10</td>
<td>Electric or piston</td>
<td>Less than 200</td>
<td>Less than 400</td>
</tr>
</tbody>
</table>

Table 3.1 Primary specifications for morphing wing test UAV

The specification listed above will be the initial approach of morphing wing design. However, minor changes may apply during the detailed design process.

3.2 Decision Matrix for Morphing Wing Concept

The design matrix and morphing wing performance evaluation will be judged by referring to the following items:

(1) **Increase of UAV performance:** It includes the consideration of several elements such as the lift force, the drag force, the lift-to-drag ratio, the maneuverability, the range and endurance.

(2) **Integrity of the structure:** Different deforming pattern will be considered based on different morphing approaches and noting that a specific deforming pattern may affect the
structure reliability. The number of the load transmitting paths, strength and stiffness of the structure, dangerous point and the maximum stress and moment will be evaluated for this item.

(3) **Weight penalty:** Small deformation in morphing structures can be directly actuated by smart materials. However, large displacements that can accommodate the associated applied loads in morphing structures can only be attained by using servo motors or hydraulic pumps. Therefore, the weight penalties will depend on the actuating mechanisms that are being used/considered.

(4) **Cost and risk:** Different morphing methods require different control parts, leading to the difference in the cost. In addition, the current condition such as the ability to access advanced materials and the manufacturing facilities will influence the level of risk in this project. For this reason, there could be several approaches that cannot be used in this work due to the level of complexity and the simplest approach will get the highest rank.

### 3.3 Selection Based on Decision Matrix

The decision and marking approach for the six morphing methods, which are mentioned in the former chapter, are mainly based on the literature review, as well as the general rules for plane design. The decision matrix including the score for each method is listed in Table 3.2 and Figure 3.1.

<table>
<thead>
<tr>
<th></th>
<th>Span change</th>
<th>Chord length change</th>
<th>Sweep angle change</th>
<th>Chord-wise bending &amp; Airfoil adjustment</th>
<th>Span-wise bending</th>
<th>Wing twisting</th>
</tr>
</thead>
<tbody>
<tr>
<td>Increase of flying performance</td>
<td>6</td>
<td>4</td>
<td>3</td>
<td>5</td>
<td>1</td>
<td>2</td>
</tr>
<tr>
<td>Reliability of structure</td>
<td>3</td>
<td>4</td>
<td>2</td>
<td>6</td>
<td>1</td>
<td>5</td>
</tr>
<tr>
<td>Weight penalty</td>
<td>3</td>
<td>1</td>
<td>5</td>
<td>6</td>
<td>2</td>
<td>4</td>
</tr>
<tr>
<td>Cost and risk</td>
<td>4</td>
<td>2</td>
<td>5</td>
<td>6</td>
<td>1</td>
<td>3</td>
</tr>
<tr>
<td><strong>Total score</strong></td>
<td><strong>16</strong></td>
<td><strong>11</strong></td>
<td><strong>15</strong></td>
<td><strong>23</strong></td>
<td><strong>5</strong></td>
<td><strong>14</strong></td>
</tr>
</tbody>
</table>

*Table 3.2 Table summarizing decision matrix*
Figure 3.1 Graphical representation of decision matrix

It is noticed from the results of the decision matrix that the chord-wise bending and airfoil adjustment has the highest potential. Therefore, the chord-wise bending and the airfoil adjustment were selected as the design strategy for morphing wing design in this project.
Chapter 4
Development of Flexible-Rib Morphing Wing System

In this chapter, the detailed design processes is presented for the Flexible-Rib Morphing Wing System (FRMWS). The design processes include the morphing mechanism, aerodynamic analysis, structural integrity, manufacturing, ground testing and actual flight test of UAV.

4.1 Morphing Approach

Airfoils determine the aerodynamic behavior of UAV and can be used to effectively morph the wing of UAV. Specifically, camber rate of the airfoils can be adjusted to meet the demands of low-speed and high-speed flying situations. Take NACA 4-digit airfoil as the example[49]. We choose NACA0012 as the base airfoil. This is a symmetric airfoil with the thickness being 12% and no camber. Let us also consider three airfoils: NACA2412, NACA4412 and NACA6412 for comparison. The only difference between these four airfoils is the camber.

The aerodynamic behavior of these four airfoils has been tested through wind-tunnel test result [50], as depicted in Figure 4.1.

![Figure 4.1 C_l-C_d curve of different airfoils and morphing wing (data based on [50])](image-url)

The C_l-C_d variation is presented in Figure 4.1, where C_l represents the coefficient of the lift of the airfoil and C_d represents the coefficient of drag of the airfoil. As depicted in Figure 4.1 there
exits differences in the aerodynamic behaviors among the selected airfoils. It is noticed that the line of the airfoil which has lower camber rate is located in the left side of the chart, which means they have lower coefficient of drag ($C_d$) with low coefficient of lift ($C_l$). On the contrary airfoils with higher camber have lower $C_d$ with high $C_l$. Since low $C_l$ is needed during high airspeed cruise and high $C_l$ is needed during low airspeed cruise, NACA0012 and NACA2412 are considered to be the most suitable for high-speed flying, while NACA412 and NACA6412 are considered to be the most suitable for low-speed flying.

For instance, the morphing wing is able to maintain the highest lift-to-drag ratio during configuration adjustment from NACA6412 (for low-speed gliding) to NACA0012 (for high speed cruise). It is also shown clearly in Figure 4.2 (red line) that morphing wing can always lie at the bottom of the chart, which means the highest lift-to-drag ratio can be kept. Thus it is evident that the flexible-airfoil morphing wing possesses better performance than traditional fixed-airfoil wing.

In summary, the chord-wise bending and airfoil adjustment method is an appropriate approach to improve the aerodynamic performance of the considered UAV. A novel morphing wing will be designed and developed in order to achieve the set objective of this project.

### 4.2 Morphing Mechanism

In order to achieve the camber adjustment, the author has introduced a novel mechanism. The mechanism is the combination of a series of units [51]. Figure 4.2 shows schematic of the mechanism model of a typical unit. It contains two sliding joints that are located at the ends of it and two revolute joints that are located in the middle span.

![Mechanism model of a typical unit](image)

**Figure 4.2** Mechanism model of a typical unit

The adjacent units’ connection is shown in Figure 4.3. It can be seen that in one combination, unit 1 and unit 3 are connected with the sliding pair. Unit 1 and unit 2, as well as unit 3 and unit 2, are connected by the revolute pairs. When unit 1 rotates relative to unit 2 around their
revolute pair, unit 3 can also be driven by the sliding pair and unit 3 will also rotate relative to unit 2. This is the basic movement of the mechanism. The angle between unit 1 and unit 3, as well as unit 2 and unit 3, can be changed from 0 degree (at this time three units lie on one line) to the maximum angle designed.

![Figure 4.3 Combination joint of adjacent units](image)

The flexible camber consists of the combination mentioned above, as shown in Figure 4.4. According to the description of the movement of one combination, the whole flexible camber can change its configuration from a straight line to a curve (configuration in Figure 4.4).

![Figure 4.4 A schematic of flexible camber](image)

In addition, one can find that the mechanism stays with only one degree of freedom in regardless of the number of the units (the degree of freedom of a mechanism is defined as the number of independent parameters that is required to describe the motion of a system). It offers the aircraft designer with more flexibility that the number of units can be arbitrarily chosen based on the requirement of the airfoil configuration. In addition, only one driving force is needed.
The basic unit of the flexible rib design should be based on the actual airfoil shape in order to achieve the desired aerodynamic performance for the specific mission. In order to achieve this goal, 3-slice sandwich is designed as the solution for the flexible rib. The basic units 1, 2, 3 and 4 are made up of two outer slices and one inner slice. Basic unit 5 only contains one slice, as depicted in Figure 4.5.

![Diagram of flexible rib units](image)

**Figure 4.5 The assemblage of the flexible camber (5 units)**

The flexible rib can be assembled by placing the inner piece of the unit into the chevron between the two outer slices in the adjacent unit. Pins are put into the holes on units and the flexible camber is assembled.

In the following sections, we describe the approach adopted in the aerodynamic modeling and simulations as well as the structural integrity analysis of the newly developed morphing wing. For simplicity, the aerodynamic analysis is limited to two-dimensional.
4.3 Aerodynamic Analysis of Morphing Wing

4.3.1 Introduction

The objective of the aerodynamic analysis is to provide a reference for selecting the number of units for the flexible rib design. We wish to consider both the aerodynamic behavior and the complexity of the structure. As a result, the model with a relatively good aerodynamic characteristics and the least number of units will be chosen. Furthermore, the result of the aerodynamic analysis will offer the direct aerodynamic load for structural analysis.

Considering the difficulties of manufacture, the shape of the flexible ribs are analyzed with 3, 4, 5, 6 basic units as well as NACA6412 (the ideal objective). In order to simplify the problem and get more precise result, 2-D airfoil models are used and the result is calibrated to that of 3-D wings. The fluid mesh is established by ANSYS Gambit 2.4.6 and the analysis is completed by Fluid Flow (FLUENT) toolbox in ANSYS 14.5 Workbench.

4.3.2 Selection of Transport Equation

Turbulence model selection is one of the most important points before the calculation. The Spalart-Allmaras model is a one-equation model that solves a modeled transport equation for the kinematic eddy viscosity. Detailed description of the Spalart-Allmaras model can be found in Reference [52]. This model fits aerospace applications such as airfoil and wing fluid analysis and has already been proven to provide good results for boundary layers subjected to adverse pressure gradients. Furthermore, proper lift-to-drag ratios can be given through this model. Comparing with two other turbulence equations such as k-ε and k-Ω models, Spalart-Allmaras model offers a better approach to the current problem. [53-55].

In order to simplify the study, 2-D airfoil model is established. However, the aerodynamic performance of 2-D airfoil will only be valid if 3-D wing has an infinite aspect ratio. When the wing has limited aspect ratio, 3-D aerodynamic indices will be somewhat different from those of the 2-D airfoil. This situation is reflected mainly by the change of lift coefficient (C_l) and drag coefficient (C_d). In order to validate the 2-D analysis of a wing with limited aspect ratio,
calibrations are conducted based on 2-D airfoil aerodynamic data. The detailed calibration process is provided in Appendix A.

4.3.3 CFD model

In this project, five models are conducted: they are morphing wing airfoil with 3, 4, 5, 6 basic units and NACA6412 airfoil. The result of NACA6412 airfoil is used for validating our selection of the parameters, as well as to enable the comparison with other four airfoils. The sketches of the five airfoils are shown in Figure 4.6.

![Figure 4.6 Sketches of candidate airfoils for morphing wing and NACA6412 airfoil](image)

In the following, the numerical results based on the configuration of NACA6412 will be validated by comparisons with experimental test data.

In order to conduct aerodynamic analysis, the airfoil should be placed into a large control volume to avoid/limit boundary effects on the resulting pressure distribution. In this case, the minimum control dimension of the control volume was selected to be more than 7.5 times of the chord length. The aerodynamic analysis is based on FLUENT package in ANSYS workbench 14.5. Fluent is widely used for fluid dynamic numerical calculation, especially for the application in aerodynamic field.
Since the viscous Spalart-Allmaras model is used for the calculation, it is important to establish the boundary layer near the airfoil surface during the discretization when drawing the mesh. The Reynolds Number (Re) of the flow is defined as the ratio of inertial force to viscous force and it plays an important role in determining the boundary layer. Re can be determined by Equation 4.1 [56], as:

\[
Re = \frac{Vl}{\nu}
\]  

(4.1)

In Equation 4.1, \(V\) is the average flow speed, which is set to be 20m/s in the analysis. \(l\) is the characteristic length, which is equal to the chord length of the airfoil (360mm), and \(\nu\) is the kinematic viscosity, which is \(1.48 \times 10^{-5}\)m²/s (at the temperature of 15°C and under the atmospheric pressure of 101325Pa). As the result, the Reynolds number in this study is calculated to be around 485000.

The resolution of the mesh at the boundary layer has a great influence to the precision of the numerical solution. Based on Reference [57], the resolution of the mesh near the boundary can be determined by a non-dimensional “wall distance”, \(y^+\), which is numerically defined by

\[
y^+ = \frac{y}{\mu \sqrt{\rho \tau_w}}
\]  

(4.2)

where \(y\) is “the distance from the wall to the cell center”, \(\mu\) is the molecular viscosity, \(\rho\) is the air density and \(\tau_w\) is the “wall shear stress”. According to Reference [57], the result of Spalart-Allmaras model is reliable only when \(y^+ \geq 30\) or \(y^+\) is very small (smaller than or on the order of 1).

Theoretically, \(y^+\) can only be accurately determined based on the results of the aerodynamic simulations. Hence, an initial approximation needs to be made for mesh resolution of the boundary layer. Reference from NASA [58] offers an online solver to calculate the viscous grid spacing. With the Reynolds number of 485000, the reference length of (length of the chord) 360mm and the objective \(y^+\) value of 0.8, the first layer thickness is estimated to be 0.013mm. After completing the analysis, \(y^+\) should be checked in order to validate the result based on the selected turbulence model and the setting of the boundary layer.
For the total thickness of the boundary layer, we assume incompressible laminar boundary layer of the flat board type, the situation of which is similar to this problem, as the reference. The numerical solution for the Falkner-Skan Problem [55] shows that the maximum thickness $\delta$ of the laminar boundary layer is

$$\delta = \frac{5.0x}{\sqrt{\text{Re}}}$$

where $x$ is the coordinate of the position on the board and Re is the Reynolds number of the free stream. The maximum $x$ is chosen to be 0.36m and Re is calculated as 485000. So in this way we can get the maximum thickness of the boundary layer as around 2.60mm.

Based on the above stated information we selected the first row thickness=0.013mm, growth factor=1.2 and 20 rows for establishing the boundary layer. The total thickness of the boundary layer is 2.43mm.

It was noticed that the boundary layer can be better approximated by quadrilateral elements than by triangle elements in 2-D analysis [3]. Therefore, quad elements were applied to the boundary layer. The established boundary layer is shown in Figure 4.7.

![Figure 4.7 Boundary layer elements near airfoil](image)

The grid generation was performed in GAMBIT. There is a choice of using triangular or quadrilateral elements to mesh the model. In fact, for simple geometries where quadrilateral
elements can sufficiently approximate the geometry, quadrilateral elements can offer a more stable solution compared with that offered by triangular elements. This is because triangular elements are sensitive to skew angles caused by high aspect ratios and quadrilateral elements are capable of being stable even at high aspect ratios. For simple-geometry problems quadrilateral elements are desirable because a few high aspect ratio quadrilateral elements are capable of replacing a much higher number of triangular elements. Hence, quadrilateral elements were selected in the meshing [3].

The model geometry of the 2-D airfoil is relatively simple. Thus the whole analysis area can be divided into six sections and each section is mapped with complete quadrilateral elements. Taking the 4-basic-unit model as an example, it contains 181950 elements. The meshed geometry and control volume is shown in Figure 4.8.

![Airfoil Mesh Boundary of control volume](image)

**Figure 4.8 Meshed geometry and control volume**

For the boundary conditions of the CFD analysis (Fluent), the airfoil boundary is defined as “wall”. All the other boundary lines including the curve and the top and bottom lines connected to it are defined as “velocity inlet”. The line that is located at the outflow side is defined as “outflow”.
Solution is conducted with FLUENT 14.5 with pressure-based, absolute velocity formulation, steady time setting and double precision environment. Spalart-Allmaras (Eqn. 1) model is chosen for the analysis and all the constants are set with the existing default value based on Reference [57]. Since the flow is treated as incompressible flow and heat transfer is ignored, energy equation is set to off and the air has the constant density 1.225kg/m³ [59].

Second order upwind momentum and modified turbulent viscosity are set for spatial discretization. For convergence monitor, continuity residual is set to $1 \times 10^{-5}$. All the other constants were remained as default setting.

In order to validate the developed CFD model, the configuration based on NACA6412 airfoil was selected. The computational results are compared with the wind-tunnel test reported in Reference [50]. Furthermore, $y^+$ chart of the 4-unit model with AOA=0 degrees is set as a reference in order to validate the setting of the boundary layer setting. These are summarized below.

Reference [50] provides the wind tunnel test data of NACA6412 airfoil under different Reynolds numbers and is used for drawing the curves shown in Figure 4.9. The figure also contains Fluent results.

It can be seen that with low angle of attack, FLUENT result matches well with the wind tunnel test result. From -4 degrees to 8 degrees, both results of the coefficient of lift are almost the
same. The discrepancy between the predictions and the test findings of drag coefficient compared up to about 8 degrees is also less than 15%.

In the case where AOA is higher than 8 degrees, greater discrepancy is observed. This is due to the approximations made in model development and analysis. This angle is also close to the stall angle, as indicated in Ref. [54]. Thus, only the results with AOA less than 8 degrees will be used in our analysis.

In addition, the value of the non-dimensional wall distance $y^+$ should be investigated in order to validate the settings of the boundary layer. Here we use the 4-unit model with AOA=0 degree as an example to show the distribution of $y^+$. The distribution of the value of $y^+$ on the airfoil surface is shown in Figure 4.10.

![Figure 4.10 $y^+$ distribution of the 4-basic-unit model with AOA=0 degrees](image)

It is noticed in Figure 4.10 that all the value of $y^+$ is smaller than or close to unity. It fits the request of using Spalart-Allmaras model for turbulence flow calculation, implying the reliability of the selected model parameters.

The aerodynamic performances of four models corresponding to four different geometries are investigated.
4.3.4 Results and Discussions

Let us now examine the effect of the number of basic units used upon the corresponding $C_L$ and $C_D$ data with angle of attack varying from -4 degrees to 8 degrees. The relationship between $C_L$ and $C_D$ of the morph wings, as well as NACA6412 wing data, is drawn in Figure 4.11.

![Figure 4.11 C\_L-C\_D curve for four morphing wings and NACA6412 wing](image)

The figure shows that with the increase of the number of units, the lift and drag ratio increases. At the same time, due to the shape difference between the morphing airfoil and ideal goal represented by airfoil NACA6412, all the morphing wings have slightly lower lift and drag ratios compared with NACA6412 wing.

In order to choose how many units to be used for morphing wing design, we should take two aspects into consideration. One is the aerodynamic performance and the other is the complexity of the mechanism. Generally speaking, with the decrease of the number of units, the complexity of the structure will decrease. However, that leads to a reduction in the aerodynamic performance of the wing. According to all the above stated reasons, it is felt that the selection of 5-unit morphing airfoil is appropriate.

In the following analysis, the numerical aerodynamic CFD result of 5-basic-unit morphing airfoil is compared to the date from a wind-tunnel test of the NACA0012 airfoil [50] in Figure 4.12. Recall that NACA0012 is the initial airfoil configuration.
Figure 4.12 $C_L$-$C_D$ curve of NACA0012 wing with lowest camber airfoil and highest-camber of 5-basic-unit model

It is observed in Figure 4.12 that the two curves intersect at the position of $C_L=0.55$. On the left side of the intersection, the wing with lower airfoil camber (NACA0012) possess higher lift-to-drag ratio. On the right side of the intersection the wing with higher airfoil camber (5 units) possess higher lift-to-drag ratio.

Specifically, take two situations as an example. According to Figure 4.12, when the plane is cruising at high-speed and assuming that the value for the lift $C_L$ is about 0.2, the corresponding $C_D$ of the wing with a smaller airfoil camber (NACA0012) is 0.0125. Meanwhile, the corresponding $C_D$ of the wing with larger airfoil camber (5 units) is 0.016. The drag force of the wing with smaller airfoil camber is 27% less than the larger airfoil camber. On the contrary, when the plane is gliding at low-speed, and assuming that $C_L$ is 0.7, the corresponding $C_D$ of the wing with larger airfoil camber is 0.044, while $C_D$ of the wing with smaller airfoil camber is 0.058. The drag force of the wing with larger airfoil camber is 25% less than that of the smaller airfoil camber.

Morphing wing can adjust its airfoil between the larger camber and the smaller airfoil camber. If the fixed wing has smaller airfoil camber, morphing wing will have the same $C_D$ at high-speed cruise. And, in the case of low-speed gliding, morphing wing can change the airfoil to the larger camber, while the fixed wing will still have lower airfoil camber. Thus, morphing wing will experience 25% less drag force than the fixed wing. Similarly, we will observe the same if the fixed wing has larger airfoil camber.
In conclusion, the plane with 5-basic-unit morphing wing can lead to a reduction in the drag force by as much as 25% through changing the airfoil on varied flight mission, compared with the plane fixed wing designs.

4.4 Geometry of the Test Morphing Wing

4.4.1 Design of Fixed Wing Section

The basic structure of the morphing wing is single-spar-single-wall configuration. With the consideration of the morphing wing area, the main spar will be moved forward slightly when compared with the original design. The distance between the central point of the main spar and the leading edge of the wing is 95mm, around 1/4 of the chord length.

To allow for maximum deformation of the flexible rib, the location of the rear spar had to be moved closer to the leading edge, as compared with the original design of the wing. The distance between the spar and the leading edge is now 140 mm. Thereby, the width of the central box will be only around half of the original fixed-wing design and this diminishes the ability of the wing to undertake twist. To compensate for this weakness, the main spar was modified into the box spar in order to increase the load-bearing ability. In addition, the whole leading edge of the wing, before the main spar, will be covered by a thin wooden skin in order to form a new wing box. The combination of these methods will significantly increase the ability of the wing to undertake the aerodynamic loads.

According to the original design, two parts of the wing are assembled with a central beam. The wing is connected on the fuselage by 4 screws and 2 wing struts.

4.4.2 Design of Morphing Flexible Wing Section

The shape of the flexible rib is shown in Figure 4.13.
As mentioned earlier, only one driving force is needed to move the flexible rib between the different configurations. In order to drive this mechanism, a servo motor driven rocker-slider mechanism is adopted. Figure 4.14 demonstrates the servo-motor-rocker driving system. The driving point of the rib is chosen at the tail tip area, as shown in the figure. A carbon-fiber rod connects all of the ribs allowing all of the ribs to bend at the same rate and to the same configuration.

The rocker is driven by the servo motor. Considering the moment output of the servo motor as well as the distance of travel of the control point, the length ratio of the arm between the driven side and slave side of the rocker is 0.59. The servo-motor-driving joint on the rocker is also a sliding pair. The transmission angle on this pair is designed between 45° and 90°.

To reduce the loads and forces acting on the ribs, rocker assembly and servo motor; three servo-motor-rocker assemblies are distributed along the flexible rib portion of the wing and will drive the carbon fiber rod simultaneously. The joint between the rocker and rod is selected to be a sliding pair instead of a revolute pair. However, this character provides more flexible at the pivot point.
In order to keep the surface smooth, fixed part of the rib will be covered by board and width-change gap of the surface will be covered by flexible material such as rubber, as shown in Figure 4.15. Since the area of the width-change gap on the surface is very small and the variety is also very small, the rubber skin will not be widely used and its effect on wing aerodynamic will be insignificant.

![Figure 4.15 Rubber covered on the surface of the wing](image)

The final wing structure is shown in Figure 4.16. It shows two different configurations of the morphing wing: smallest camber configuration and biggest camber configuration.

![Figure 4.16 Final morphing wing structure](image)

4.4.3 Material Selection and Weight Estimation of Test Wing

The predominate material used for the constructing of the wing is wood. Birch wood laminate, balsa wood and paulownia wood are all utilized in the developed wing based on their advantages. The flexible rib is the weakest part of the design; therefore, the stronger birch wood laminate was used for the construction.
The weight of the original wing is 850g for each side. Considering 25% performance increase from the morphing wing, as mentioned in Chapter 4.4.3, and the enlarged wing area, the designed weight of the morphing wing should be controlled at less than 1110g for each side in order to govern the weight penalty. The weight estimation of the wing structure for one side is 760g. Considering the additional weight from the adhesive and accessories estimated to add another 15%, the final weight of the morphing wing structure will be 875g.

4.5 Structural Analysis of FRMWS

4.5.1 Introduction

Static structural analysis was conducted on the flexible rib. There are two purposes for conducting the structural analysis: one is to examine the safety and reliability of the flexible rib under the applied aerodynamic load, the other is to evaluate the required driving force, which is required for the selection of the servo motor in the system.

Finite element analysis (FEA) will be employed to conduct the structural analysis. Specifically, the commercial code ANSYS 14.5 Workbench and APDL 14.5 are used in the finite element analysis.

4.5.2 Finite Element Model

The geometry of the flexible rib was imported into the finite element model for structural analysis. Six models of the flexible rib with different element density were established for convergence analysis. The number of the element varies from 16441 to 73302. A representative meshed model is shown in Figure 4.17.
Three dimensional hexahedron and prism solid elements are used in the analysis. The hexahedron element type, SOLID186 [60] in ANSYS is utilized as it is a higher order element containing midside nodes. It can be degenerated into the tetrahedral SOLID187 element. Comparing with other beam or shell elements, SOLID 186 has the least assumption and the highest precision. To conduct the analysis, reduced integration was utilized to reduce the analysis time and to avoid the effect of shear locking. To account for contact in the model, CONTA174 and TARGE170 elements are applied on the surface of SOLID186 mesh elements in contact area.

The material of the flexible rib is birch plywood; therefore the analysis utilizes the orthotropic constitutive law since the material properties vary with orientation. The properties of the birch plywood can be found in Ref. [61][62]. The strength of the birch plywood is 11MPa. Elastic properties of plywood are listed in Table 4.1 and the coordinate system is shown in Figure 4.18.

<table>
<thead>
<tr>
<th>E₁</th>
<th>E₂</th>
<th>E₃</th>
<th>v₁₂</th>
<th>v₁₃</th>
<th>v₂₃</th>
<th>G₁₂</th>
<th>G₁₃</th>
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<td>0.547</td>
<td>0.466</td>
<td>910</td>
<td>431</td>
<td>264</td>
</tr>
</tbody>
</table>

**Table 4.1 Elastic properties of 3-layer birch plywood (E and G in MPa) [61]**

![Figure 4.18 Coordinate system of birch plywood as per Ref. [61]](image)

The root of the flexible rib was set to fixed support, to replicate the connection of the rib to the spar. The carbon-fiber connecting rod in the FRMWS will restrict the movement of the driving point; therefore this point was set to a fixed support in the direction of the driving force. The direction was determined by the position of the sliding pair between the rocker and carbon-fiber
rod (mentioned in chapter 4.4.2) for the configuration being analyzed. The support is shown in Figure 4.19.

![Supports on FE model](image)

(a) Fixed support at root  (b) Directional fixed support at driving point

Figure 4.19 Supports on FE model

A variety of contacts was employed in the model to be as realistic as possible. According to Ref. [63], bonded contact is sufficient to account to the adhesion of the wood pieces for the different layers in the unit. Frictionless contact was applied on the contact surfaces between basic units and between the pin and the basic unit. Examples are shown in Figure 4.20.

![Contacts](image)

(a) Bonded in one unit  (b) Frictionless between units  (c) Frictionless between pin and unit

Figure 4.20 Contacts

The analysis was conducted under the most severe aerodynamic load. According to Ref. [64], the pressure distribution with fixed camber and varied AOA acquired from aerodynamic analysis, load with the biggest camber rib configuration and AOA=0 degree was determined mostly critical. Following FAR23 standard, 3.8G overload was applied and the final aerodynamic pressure distribution on each unit is shown in Figure 4.21 and the application of the pressure on the model is shown in Figure 4.22.
Figure 4.21 Pressure distribution on each unit
(Absolute lower value for upper surface, absolute higher value for under surface)

Supporting force on the driving point is taken as the convergence index. Figure 4.23 displays the relationship between the magnitude of the force and the number of the meshes. It can be seen that the force is stable near 1.72N when the number of the elements has risen to over 40000. Therefore, the result of FEM is reliable when the number of elements is over 40000. Since, the results from 73302-element model are available this model was used for discussion purposes.
4.5.3 Results and Discussion

The maximum principal stress of the flexible rib is shown in Figure 4.24 and 4.25. For most area of the rib, the stress magnitude is $10^5$Pa. Based on the maximum stress failure criteria of the composite material and the strength of the plywood, the structure is safe.

![Figure 4.24 Maximum principal stress of the flexible rib](image)
The maximum stress in this structure is 2.62 MPa and appears at the position shown in Figure 4.26. From the figure, it can also be seen that the stress level around the pins is higher than other areas in the flexible wing. This stress concentration is caused by the contact and force transmission between the units and discontinuity in the structure. Even though the stress is higher in this area, it is still smaller than the strength of the plywood with the safety factor of around 4. Therefore, we can conclude that the flexible rib is safe when undertaking the most severe aerodynamic load and it complies with FAR23 standards.
Since the required force for actuation of the morphing wing is now known (1.72N), the required torque for the servo motor can be determined. From the morphing wing design, it is known that there are totally seven flexible ribs and that three driving rockers will be utilized. Therefore, the minimum required torsion output of the servo motor must be 0.059N·m. To meet this requirement, but also considering the reliability and weight of the servo motor, the ideal choice is offered by Hitec Co. Ltd, the HS82MG standard metal gear micro servo motor [65]. The weight of the servo motor is only 18.71 g and it can give the output of 0.2744N·m. Compared with the moment output needed, it can offer the safety factor of 4.65, which is ample to drive the entire morphing wing.

4.5.4 Conclusion

According to the FEA, it is concluded that the design of the most critical components of the morphing wing (the flexible rib mechanism) is safe based on the load outlined by FAR23. Additionally, from the design it was possible to determine the required force that the servo motor would have to provide to actuate the morphing wing at the maximum aerodynamic load. The Hitec HS82MG servo motor is chosen as the driving servo motor of the morphing for its lightweight and high torque output characteristics.
Chapter 5
Functional Prototyping and Testing

5.1 Introduction

In this chapter, we outline the detailed manufacturing as well as the development of a functional prototype so as to enable the test flight of the newly developed morphed wing design. The Chapter is divided into four main topics. The first deals with the manufacturing of the newly devised flexible rib morphing wing. Second, a discussion on the static load test conducted is presented. The third topic addresses preliminary aerodynamic tests to demonstrate the possible aerodynamic improvement resulting from the newly devised morphing wing. Finally, flight tests of a functional prototype to demonstrate the success of the entire research project.

5.2 Prototype Manufacture of FRMWS

Once the design and analysis of the FRMWS was completed validated, the developed designs were prepared and sent to an externally contracted manufacturer. All of the components consisting of the FRMWS were manufactured through the use of laser-cutting. The assembly of the components was done by the author. Figure 5.1 shows the assembled flexible rib that will be used for the morphing portion of the wing. Figure 5.2 shows the structure of the morphing wing including the flexible ribs, the servo-motor-rocker driving system and the spars. Figure 5.3 shows in greater detail the servo-motor-rocker driving system. Finally, in Figure 5.4, the complete morphing wing is shown.

![Figure 5.1 Assembled flexible rib](image-url)
The weight of each constructed morphing wing is measured to be 1100 g, which is within the weight tolerance of the proposed design as specified earlier. The total weight of the test plane
including all of the components, measurement equipment and power supplies is 5.8 kg. The appearance of the test plane with the installed morphing wings is shown in Figure 5.5.

![Test UAV of FRMWS](image)

**Figure 5.5 Test UAV of FRMWS**

### 5.3 Static Load Testing

In order to ensure that the design and manufacturing of the structure adhered to FAR23 standards, the static load testing was conducted before the flight test. The objective of the static load test is to determine whether the morphing wing can actually withstand more than 3.8G of the gross weight, which simulates the maximum aerodynamic loads that are expected during the test flight. Additionally, it must be shown that the mechanism can function under these loads to reassure that it can perform appropriately for the entire flight envelope.

As previously stated, the gross weight of the final test UAV is 5.8kg, therefore the wings must be able to undertake 22.04 kg of load. To perform this test, bags containing 0.5 kg of water are distributed equally over the wing as shown in Figure 5.6. In total 40 bags (20 kg) are applied on the wing. Including the weight of the wing itself (2.2kg), the total load applied is 22.2kg, which is 102% of the value stated in FAR23 requirement. Based on this observation it confirms that the manufactured FRMWS can undertake the necessary aerodynamic loads.
Additionally to ensure that the flexible-rib morphing mechanism could withstand the aerodynamic loads, the actuation of the wing was conducted while under the static load. By utilizing the embedded servo-motor-rocker mechanism to actuate the wing, it was able to fully extend (Figure 5.7 (a)) and fully retract (Figure 5.7 (b)). Therefore, the test provided confidence that the morphing mechanism can function appropriately during under the static load. Additionally, it further verifies that all of the morphing wing components including the driving servo motor, transmitting rocker and the flexible rib, work appropriately under static loading in the whole flight envelope.

Figure 5.6 Static load test

![Static load test](image1)

Water Bag

Figure 5.7 Morphing test during the load test

![Morphing test](image2)

(a) (b)
5.4 Wind Tunnel Testing

Since the actual airflow over the wing is much more complex than the predicted through the numerical methods, wind tunnel testing is validating the predicted results. Therefore, wind-tunnel testing is conducted to validate the simulated performance results of the morphing wing. Two scaled test specimens of the morphing airfoil were manufactured by the author as shown in Figure 5.8. The first scaled specimen has the airfoil of the smallest camber and the second has the biggest camber. For those two samples, the span and chord length was 154mm and 90mm respectively.

![Figure 5.8 Test sample](image)

A rudimentary wind tunnel was designed and constructed as shown in Figure 5.9. It consists of three main parts: the airflow source, the flow guidance tube and the aerodynamic balance. A NEIKO WFB180 portable fan blower provides airflow to the guidance tube. The cross-section of guidance the tube is rectangle and it is designed to be able to contain the test samples. Honeycomb airflow guides are placed at the airflow inlet area in order to a stable laminar flow. The velocity of the airflow was measured to be 10 m/s using a SMART SENSOR Anemometer. To measure the change in the lift and drag forces that the airfoil experienced, an aerodynamic balance was constructed. An electrical weight balance was used to measure the resulting forces from the aerodynamic balance. AOA of the specimen can be varied to allow testing for the different test configurations.
Four test configurations for each specimen are conducted to measure the lift and drag forces generated by the airflow. For the biggest camber airfoil the AOA considered are -2, 0, 2, 4 degrees, respectively. For smallest camber considered, the AOA considered are 0, 2, 4 and 6 degrees. The resulting drag and lift forces were determined from the aerodynamic balance, and using the results the respective aerodynamic coefficients $C_L$ and $C_D$ were determined. The $C_L$-$C_D$ curves of both test samples are shown in Figure 5.10.

![Figure 5.9 Wind tunnel equipment](image)

![Figure 5.10 $C_L$-$C_D$ curves of both test samples (wind-tunnel testing result)](image)
It can be seen in Figure 5.10 that two curves intersect at the position of $C_L=0.54$. On the left side of the intersection point morphing airfoil with smallest camber has bigger lift-to-drag ratio. On the right side of the intersection point morphing airfoil with biggest camber has bigger lift-to-drag ratio. These results are in line with the predicted numerical results. It further demonstrates that comparing with fixed airfoil wing morphing wing can always keep the highest lift-to-drag ratio by adjusting its airfoil based on actual flying request. The result of the wind tunnel testing proves this advantage of FRMWS in actual flow situation.

5.5 Flight Test

The main objective of performing flight testing is to examine the performance and reliability of the FRMWS during actual flight. The test plane was flown in a radio controlled flying club of Toronto. During flight, the FRMWS will be morphed to different configurations ranging from fully retracted to fully extend; these configurations were pre-programmed to the on-board control system. The radio control would measure the inputs given by the pilot to the plane, which would give an indication to the change in performance for the different configurations. To further examine the performance during flight, the aircraft was fitted with onboard video recording equipment to record the entire flight and allow for analysis of the performance afterwards. To minimize the risk while performing the flight testing, the control system of the plane and the control system on FRMWS were isolated from each other. Picture taken during the flight test is shown in Figure 5.11.

![Figure 5.11 Flight test](image)
A sample of the image stills from the movie taken by the camera on the plane is shown in Figure 5.12, and clearly shows the different configurations of the wing during the flight test. Additionally, the images verify that the FRMWS performed properly during the actual flight and the morphed configuration of the wing was stable and did not fluctuate significantly due to the aerodynamic loads. The system operates very reliably and the morphing wing undertakes dynamic load and the vibration of the plane. In conclusion, the result of the flight test gives convincible evidence that FRMWS is a suitable and reliable system to be equipped on UAVs.

![Morphing wing working status with different camber](image)

**Figure 5.12** Morphing wing working status with different camber during the flight
Chapter 6
Conclusions and Future Work

6.1 Statement of the Problem

Unmanned aerial vehicle (UAV) system must be able to perform many different maneuvers during a typical mission. Accordingly, to improve the overall performance of the UAV it is ideal to have the optimal wing shape for each different flight stage, thereby employing a wing that can morph to the required configuration. A morphing wing can be defined as the design and development of a flexible wing system that can adjust its configuration during varied flight missions in order to attain the best flying performance for these varied missions. Compared with fixed wing designs currently utilized, the advantages of morphing are numerous. It can increase the lift-to-drag ratio of the wing during both high-speed and low-speed cruise, increase the maneuverability of the plane, even expand the plane envelop [66]. Specifically, morphing wing can offer the following [67]: improved aircraft performance that leads to expansion the flight envelope, replacing conventional control surfaces for flight control thus reducing weight and design complexities and provide stealth capability [68], reducing parasite drag and vibration and flutter [69][70]. In addition, the control of the local flow by making small morphing adjustments to the wing surface can also help improve the aerodynamic force distribution [71].

6.2 Objectives

The goals of this study are to develop a novel morphing wing mechanism to be adopted in wing system. Specifically, the aims of the study are to:

(1) develop novel conceptual design of morphing wing systems in order to improve the flight performance of UAV without incurring large weight penalty,

(2) carry out structural and aerodynamic analyse to study the effect of the new morphing wing design upon the integrity and performance of the morphed wing and the adherence to aircraft design standards such as FAR23,

(3) design, build and test a functional prototype with the new morphing wing system to examine its function, reliability, and weight penalty, and
(4) implement the designed morphing wing to a small UAV and conduct the necessary flight test to test the reliability and the performance of the new design.

6.3 General Conclusions

In the following, a summary of the conclusions from the study are presented:

(1) the numerical predictions and the experimental results indicate that a significant improvement in the aerodynamic characteristics has been achieved by the proposed novel morphing wing system,

(2) wind tunnel tests of the morphing wing and flight tests of the complete system demonstrated the reliability and suitability of the proposed morphing wing system for uses in UAV,

(3) the designed flexible-rib mechanism modifies the airfoil and camber of the wing with sufficient accuracy, efficiency and flexibility worthy of future consideration, and

(4) proposed, designed, analysed, manufactured, and tested a novel flexible rib morphing wing system (FRMWS), for small UAVs in accordance with the safety requirements specified by FAR23 standards.

6.4 Contribution of the Thesis

The contributions of the presented work are:

(1) developed novel flexible-rib morphing wing system that adjusts the airfoil and camber configurations,

(2) developed an aerodynamic model to examine the performance of the newly morphing wing in the project,

(3) conducted FE structural analysis to validate the integrity of the design and its adherence to FAR23,

(4) built a functional prototype of FRMWS and conducted static load testing, wind tunnel testing and flight tests to demonstrate the adaptability of the novel morphing wing concept.
6.5 Future Work

The following areas are worthy of further research:

1. expand the wind-tunnel tests by examining the aerodynamic performance of the full morphing-wing system to better characterize the performance improvements,

2. implement a feedback control system to allow the morphing wing to provide rolling control for the UAVs by varying the configurations of the opposite wings to attain unbalanced lift forces,

3. investigate the feasibility of employing smart-materials to actuate FRMWS instead of servo motors, and

4. conduct optimization studies of FRMWS to achieve higher efficiency, lower weight, and better performance.
Appendix A Calibration of $C_l$ and $C_d$ from 2-D data to 3-D data

1. $C_l$ calibration

After CFD calculation, we can get $C_l$ value of the 2-D airfoil. Also the slope of $C_l$ with the angle of attack ($C_{l\alpha}$) can be acquired. Because of the limited aspect ratio, $C_{l\alpha}$ of the 3-D wing will be reduced. Reference [72] gives a semi-empirical formula of calibrating wing lift curve slope at low airspeed and subsonic airspeed.

\[
C_{l\alpha} = \frac{2\pi A}{2 + \sqrt{4 + \frac{A^2\beta^2}{\eta^2}(1 + \frac{\tan^2\Lambda_{max}}{\beta^2})}} \left(\frac{S_{exposed}}{S_{ref}}\right)(F)
\]  

\text{(A.1)}

where

\[
\beta^2 = 1 - M^2, \quad \eta = \frac{C_{l\alpha}}{2\pi / \beta}
\]  

\text{(A.1)}

$A$ is the aspect ratio, $M$ is Mach number, $S_{exposed}$ and $S_{ref}$ are the respective areas of the exposed and reference wing planform, $F$ is fuselage lift factor. $\Lambda_{max}$ is the sweep angle of the wing at the chord location where the airfoil is thickest. The slope in the expression is in radians.

In this project, the test plane of morphing wing will employ a straight wing configuration. Because of the relatively small fuselage cross-section area and its shape, $S_{exposed}$ is considered equal to $S_{ref}$ and the fuselage lift factor is considered as 1. Furthermore, there is no sweep angle of the wing, so $\Lambda_{max} = 0$. The plane will fly at around 20m/s, and we can thus ignore the compression factor of air. So the Mach number $M=0$. In this way, the expression of the wing lift curve slope can be simplified as

\[
C_{l\alpha} = \frac{\pi A}{1 + \sqrt{1 + \frac{A^2\pi^2}{C_{l\alpha}^2}}}
\]  

\text{(A.2)}

In this project, $A=6.9$. 
$C_{\alpha}$ can be reached directly from Fluent CFD analysis. After calibration from former equation, we can get $C_{L\alpha}$ of the 3-D wing. Then we can get the calibrated $C_L$ of the 3-D wing in linear slope area before reaching the edge of stall.

2. $C_d$ calibration

The drag force of the 3-D wing is composed of two parts. The first part is the so-called ‘parasite drag’ or ‘zero lift drag’ that comes without the influence from the lift. The parasite drag is mainly produced by viscous effects of the wing surface, the pressure distribution and the friction when the air flows over the surface. There is also a corresponding coefficient $C_{d0}$ which can be directly acquired through 2-D Fluent CFD analysis. In the following analysis, $C_{d0}$ will be directly used as the parasite drag part of the wing, i.e.,

$$C_{d0} = C_{p0} \quad (A.3)$$

The other part of the drag force is called ‘induced drag’. It is related to the lift directly. Accounting for the influence of 3-D effects, the phenomenon of downwash exists with the induced lift force of the wing. Downwash will produce extra drag force of the wing when the lift force exists ‘drag-due-to-lift’. Reference [72] gives the expression of the coefficient of induced drag $C_{Di}$ as,

$$C_{Di} = \frac{C_L^2}{\pi A e} \quad (A.4)$$

$C_L$ is the lift coefficient of the 3-D wing and $A$ is aspect ratio. Noting that $e$ is a constant called span efficiency factor or Oswald efficiency factor. The lift-dependent drag [73] is composed of three parts: an inviscid part that is caused by induced velocities from the wake, and a viscous part that is caused by the increase in skin friction and pressure drag because of the redistribution with the variation of the angle of attack. In this way, the Oswald efficiency factor can be expressed in the following formula [73]:

$$e = \frac{1}{Q + P \pi A} \quad (A.5)$$

$Q$ covers the inviscid part of the induced drag and it can be expressed as
\[ Q = \frac{1}{us} \]  

(A.6)

\( u \) can be taken as 0.99 in this equation. The factor \( s \) accounts for the extra lift-dependent drag caused by the change of the span loading with the influence of the fuselage. It is a function of the fuselage diameter \( d_f \) and the wingspan \( b \).

\[ s = 1 - 2\left( \frac{d_f}{b} \right)^2 \]  

(A.7)

In this project, the ratio between \( d_f \) and \( b \) was taken to be 0.1.

Reference [73] also offers a diagram to determine the value of \( s \) based on the ratio of \( d_f \) and \( b \).

\( P \) covers the viscous part and is expressed as

\[ P = KC_{d_0} \]  

(A.8)

The constant \( K \) can be determined from flight test area. Also according to Reference [73], based on different kinds of planes, \( K \) can be determined as 0.38.

In conclusion, the final expression of the Oswald factor for the whole aircraft is

\[ e = \frac{1}{\frac{1}{us} + KC_{d_0} \pi A} \]  

(A.9)

Finally the expression of the coefficient of drag of the 3-D wing is composed by the parasite drag coefficient and induced drag coefficient, which is

\[ C_D = C_{d_0} + C_{d_i} \]  

(A.10)

(3) Morphing wing calibration process

The following was used in the calibration of the aerodynamic analysis results of the newly devised morphing wing
\[ C_{La} = \frac{\pi A}{1 + \sqrt{1 + \frac{A^2 \pi^2 \alpha^2}{C_{La}^2}}} = \frac{21.67}{1 + \sqrt{1 + \frac{469.4}{C_{La}^2}}} \]  

(A.11)

\[ C_L = C_{La} \alpha \]  

(A.12)

\[ C_D = C_{D0} + C_{Di} = C_{D0} + \frac{C_L^2}{\pi A e} = C_{D0} + \frac{C_L^2}{\pi A} \left( \frac{1}{\mu s} + KC_{D0} \pi A \right) = C_{D0} + \frac{C_L^2 (1.03 + 8.23 C_{D0})}{21.67} \]  

(A.13)

\( C_{La} \) can be acquired by solving \( C_1 \) of the airfoil with different angles of attack within the linear area. \( C_{D0} \) can be directly calculated which is determined equal to \( C_{D0} \). The corresponding \( C_L \) and \( C_D \) at different angles of attack for the morphing wing, as well as the data of the same wing with NACA0012 and NACA6412 airfoils, are given and the result is used for further analysis and decision making. Details are provided in Chapter 4.
Reference


[59] Han Zhanzhong, 2009, FLUENT fluid engineering simulation and calculation, examples and analysis, Beijing Institute of Technology Press.


