MEASUREMENT OF AERODYNAMIC CHARACTERISTICS OF MAVS USING MOTION TRACKING

BY

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THESIS

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Abstract

Aerodynamic characteristics of Micro Air Vehicles (MAVs) is not well addressed in aeronautics literature, and more experimental data are required to help better understand the behavior of fixed-wing MAVs. In the current research, aerodynamic and stability characteristics of two MAVs were measured using a motion tracking system. Both the aircraft position and propeller rotation were tracked. The aircraft used were flat foam surface, highly aerobatic RC models. Tests were conducted for unpowered and powered flight conditions to assess the aerodynamic performance of a commercially manufactured Extra 300 3D and a custom-built Extra 260 with wingspans of 42.67 cm (16.8 in) and 41.27 cm (16.25 in), respectively. A comparison of theoretical calculations with experimental results was made for parameters such as the lift curve and neutral point location. The aircraft used in this research operated in a Reynolds number range of 20,000 to 30,000. Low Reynolds numbers effects that influence the aerodynamic characteristics and performance of the aircraft are discussed in this thesis. The behavior of MAVs in powered flight was analyzed by testing the Extra 300 3D over a range of propeller advance ratios ($J = 0.3–1.0$) with the propeller running at pre-determined speeds (RPM). A thrust model was developed for a commercially manufactured propeller, and the lift and drag characteristics of the aircraft was analyzed. The results indicate that the propeller induced flow influences the aerodynamic performance of the aircraft. Results indicate a delay in the onset of stall, an increase in the lift curve slope, and reduction in drag with decreasing advance ratios ($J$). The results show a myriad of interesting aerodynamic trends that are discussed.
To my parents for raising me right
Acknowledgments

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## Nomenclature

### Symbols

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<th>Description</th>
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<tbody>
<tr>
<td>$a_x, a_y, a_z$</td>
<td>body-axis translational acceleration</td>
</tr>
<tr>
<td>$\mathcal{A}$</td>
<td>aspect ratio</td>
</tr>
<tr>
<td>$b$</td>
<td>wingspan</td>
</tr>
<tr>
<td>$c$</td>
<td>wing chord</td>
</tr>
<tr>
<td>$C_D$</td>
<td>drag coefficient ($D/\frac{1}{2} \rho V^2 S_{ref}$)</td>
</tr>
<tr>
<td>$C_{D_p}$</td>
<td>parasite drag coefficient</td>
</tr>
<tr>
<td>$C_{D_i}$</td>
<td>induced drag coefficient</td>
</tr>
<tr>
<td>$C_L$</td>
<td>lift coefficient ($L/\frac{1}{2} \rho V^2 S_{ref}$)</td>
</tr>
<tr>
<td>$C_{L_{\alpha}}$</td>
<td>lift curve slope</td>
</tr>
<tr>
<td>$C_l, C_M, C_N$</td>
<td>roll, pitch, and yaw coefficients</td>
</tr>
<tr>
<td>$C_T$</td>
<td>thrust coefficient ($T/\rho n^2 D_p^4$)</td>
</tr>
<tr>
<td>$D$</td>
<td>drag</td>
</tr>
<tr>
<td>$D_p$</td>
<td>propeller diameter</td>
</tr>
<tr>
<td>$F$</td>
<td>force</td>
</tr>
<tr>
<td>$I_{xx}, I_{yy}, \ldots$</td>
<td>mass moments of inertia</td>
</tr>
<tr>
<td>$J$</td>
<td>advance ratio</td>
</tr>
<tr>
<td>$L$</td>
<td>lift</td>
</tr>
<tr>
<td>$m$</td>
<td>airplane mass</td>
</tr>
<tr>
<td>$n$</td>
<td>propeller rotational speed (rev/sec)</td>
</tr>
<tr>
<td>$p, q, r$</td>
<td>roll, pitch and yaw rates</td>
</tr>
<tr>
<td>$R$</td>
<td>transformation or rotation matrix</td>
</tr>
<tr>
<td>$Re$</td>
<td>Reynolds number based on mean aerodynamic chord ($Vc/\nu$)</td>
</tr>
<tr>
<td>$S_f$</td>
<td>fuselage area</td>
</tr>
</tbody>
</table>
\( S_{\text{prop}} \) = propeller area
\( S_{\text{ref}} \) = reference area (wing area + fuselage area)
\( S_t \) = tail area
\( S_w \) = wing area
\( u, v, w \) = body-fixed translational velocity
\( V \) = inertial speed
\( \alpha \) = angle of attack
\( \dot{\alpha} \) = angle of attack rate
\( \beta \) = sideslip angle
\( \dot{\beta} \) = rate of change of sideslip angle
\( \phi, \theta, \psi \) = roll, pitch and heading angles
\( \rho \) = density of air
\( \nu \) = kinematic viscosity

**Abbreviations**

LE leading edge
MAV micro air vehicle
RPM revolutions per minute
SM static margin
UIUC University of Illinois at Urbana-Champaign
Chapter 1

Introduction

1.1 Literature Review

There is a concerted effort to design aircraft that are capable of operating in special, limited-duration military and civil missions. These aircraft, called micro air vehicles (MAVs), are of interest because electronic surveillance equipment can be installed such that the payload mass is low. MAVs have advantages such as rapid deployment, real-time data acquisition capability, low radar cross section, and low noise. The primary missions of interest for fixed-wing MAVs include surveillance, detection, and communication. Surveillance missions include infrared images of a battlefield (referred to as the “over the hill” problem) and urban areas (referred to as the “around the corner” problem). These real-time images can provide vital information during times of conflict, peace or natural disasters [1, 2].

Significant technical barriers must be overcome before MAV systems can be realized. These include small-scale power generation and storage, navigation, communication, aerodynamics, and control. One of the least understood aspects of small-scale flights is the aerodynamic characteristics. Figure 1.1 shows the gross mass of MAVs and other flying objects versus the operating Reynolds numbers. A combination of small length scale and low velocities results in MAVs operating outside the normal realm of conventional flight.

The aircraft used in MAV research operate at low Reynolds numbers due to their size. Wind tunnel studies have revealed the changes in aerodynamic properties of airfoils as the Reynolds numbers is reduced [3–10]. A study conducted by Mueller, et al. [2] showed that reduction in the Reynolds numbers resulted in poor performance due to a drastic reduction in $L/D$ for flat-plate wings. The deterioration in the aerodynamic performance of airfoils at low Reynolds numbers is attributed to the scaling effects. This reduction in performance is illustrated in Fig. 1.2. It is observed that the scaling effects become more pronounced as the Reynolds number decreases to 5,000–40,000 [11].

Spedding and McArthur [12] showed that the span efficiency factors for an airfoil and wing at low Reynolds numbers deviated significantly from the values that are well documented in aeronautics literature. The cause for the modification in the values was attributed to the change in the location of the separation point for an airfoil. Experiments were conducted using an E387 airfoil and a wing with an aspect ratio of six. Lift slope correction factors, at low Reynolds numbers, were suggested for the 2D and 3D flow cases. The authors proposed a change in theoretical
models for flow regimes in which viscous effects could not be neglected, even at low angles of attack.

A recent study by Uhlig, et al. [13] evaluated the performance of a commercially manufactured aircraft and a hand-built glider in a low Reynolds number range of 15,000 to 25,000. Tests were conducted for both aircraft over a wide range of angles of attack and different configurations. The study revealed that the experimental lift curve slopes were lower than theoretical results and that the parabolic fit over the low angular rate data fit the effect of induced drag until the onset of flow separation. The stability characteristics of the aircraft were obtained, and it was observed that the location of the neutral point was governed by a combination of factors that included interaction between the wake of the main wing and horizontal tail, the lift curve slopes, and the aerodynamic center of each surface.

The advent of improved electronics and materials used to build the current generation of micro-scale RC airplanes has resulted in a marked improvement in their performance [14]. MAVs fly over a large range of angles of attack to improve their maneuverability while operating in confined spaces. The low Reynolds number, low aspect ratio nature of MAVs creates complex flow conditions over the wing; specifically, the short wingspan is dominated by the effect of wing tip vortices. These tip vortices cause significant spanwise flow on the top surface of the wing and generate nonlinearities in the lift at moderate and high angles of attack [15].

In an effort to alleviate the problem of rising fuel costs, high-efficiency propellers are used in propulsion systems. However, inherent to the consideration of propellers as an aircraft propulsion system is the problem of interaction between the propeller and other aerodynamic surfaces of the aircraft. This interaction can affect the aerodynamic performance of the aircraft. Several wind tunnel and numerical studies have been dedicated to investigate the influence
of propeller wake on the aerodynamic performance of a wing [16, 17]. Catalno [18] conducted a study on the influence of propeller slipstream on the wing boundary layer. Tests were conducted in the pusher and tractor configurations. The results of the study indicated an increment in the lift and pressure drag due to the propeller slipstream. Witkowski, et al. [19] explained the influence of propeller wash on the aerodynamic performance of a wing, placed downstream. The study showed an increase in the slope of the lift curve with decreasing advance ratios. They also noted that an increase in the advance ratio resulted in a corresponding rise in the drag produced by the wing.

A study conducted by Null, et al. [20] investigated the influence of propulsive-induced flow on the aerodynamic performance of MAV’s. Wind tunnel tests were conducted at various Reynolds numbers and propeller speed settings and configurations. They found that the induced flow from the propulsion system had a positive effect in the lift and resulted in a delayed stall. However, the induced flow had a detrimental effect on the drag, resulting in a decrease in the lift-to-drag ratios at low angles of attack.

Owing to the small size and weight of MAVs, it is difficult to mount on-board sensors without altering the aerodynamic and handling characteristics of the aircraft. Thus, there is a need for off-board or external sensors to monitor the behavior of an MAV. Preliminary research regarding MAVs using an external sensor system was documented by De Blauwe, et al. [21]. The Vicon motion capture system has been used to track and control small and micro UAVs in an indoor environment [22–24]. The work performed by Cory [25] successfully demonstrated the use of motion tracking for the modeling and control of an autonomous glider to perform a perch maneuver. The researchers developed a lift and drag model for a wide range of angles of attack. Extensive research was conducted by Mettler [26, 27] to obtain the aerodynamic forces and coefficients for a hand-built glider. From the glider flight path data, the researchers were able to establish the lift, drag, and pitching moment coefficients. A single flight of the glider was presented for the purpose of understanding the behavior of the glider in the linear and post stall region of flight. Lift and drag hysteresis was observed by virtue of the rapid change in the pitch of the glider (see Figs. 1.3 and 1.4).

Figure 1.2: Maximum lift to drag ratio as a function of Reynolds number (taken from [11]).
The study emphasized the benefits of using a vision-based motion tracking system as it empowered the user to gather data that captured transient aerodynamic phenomena.

A study conducted by Uhlig, et al. demonstrated the ability to use an off-board motion tracking system to track an MAV [28] and obtain aerodynamic characteristics [29]. A commercially manufactured MAV was used for the research. The study made a comparison of the experimental lift curve slope with the result obtained from lifting line theory and observed that the experimentally determined slope was lower than that predicted by theory.

In the current research, a commercially manufactured Extra 300 3D and a custom-built Extra 260 were used to gather flight test data using a 16 camera motion tracking system. The lift and drag characteristics of both aircraft were obtained for free-flight conditions. In addition to lift and drag forces, the moments acting on both aircraft were determined and estimates of the longitudinal and lateral static stability derivatives were obtained.

Most of the work in the area of motion tracking of MAVs has been performed for glide flights. However, the current work explains the aerodynamic characteristics of an MAV in the powered flight configuration. The effect of propeller slipstream on the lift and drag characteristics are investigated and documented in this research.

A commercially manufactured E-flite™ propeller was mounted on the nose of the Extra 300 3D aircraft. Initially, a preliminary thrust estimate was made to “ball park” the performance of the propeller. As no experimental data were available for the E-flite propeller, a thrust model based on the blade element theory, was developed and implemented. It was critical to develop a model that closely resembled the propeller behavior in order to better predict the aerodynamic performance of the aircraft at various propeller speeds and advance ratios ($J$).

The results obtained from glide and powered flight configurations are presented in this thesis. The results and
associated aerodynamic phenomena are discussed in detail and aid in the better understanding of MAV operation and behavior.

1.2 Organization

Chapter 2 of this thesis explains the experimental methodology, data acquisition and reduction processes. The calculation of aerodynamic forces and moments are discussed. Chapter 3 explains the various tests that were conducted in order to validate the motion tracking system. The noise levels and accuracy of the measurements at various aircraft attitude are presented. Chapter 4 lays out the results obtained from the glide and powered flight tests of the two MAVs used in this research. Chapter 5 comprises of two sections. Section 5.1 summarizes the research that was conducted and important observations and conclusions are brought to the fore. Section 5.2 is dedicated to discussing the shortcomings, suggestions, and various research enhancements that could be pursued for work in the future.
Chapter 2

Experimental Procedure

This chapter describes the experimental setup (see Fig. 2.1) used in this research. All the experiments were performed using the motion tracking facility at the University of Illinois at Urbana-Champaign. A detailed description of the data acquisition and reduction processes, geometric and physical properties of the aircraft tested, and calculation of aerodynamic forces and moments are discussed in this chapter.

Figure 2.1: Motion tracking facility using the Vicon system at UIUC.
2.1 Experimental Setup

Figure 2.1 shows the motion tracking facility at UIUC. In the current research, 16 cameras with individual sources of infrared light were used. Figure 2.2 shows a Vicon T20-S camera that was used in the motion tracking system. The T20-S offers an impressive resolution of 2 megapixels, captures 10-bit grayscale using 1600×1280 pixels and can capture speeds of up to 2,000 frames per second. The camera consists of state-of-the-art sensors, freeze frame shutter, and a lens that is custom built for motion capture. The Vicon lens has a large image circle to ensure that the entire image is evenly illuminated, not just the center.

2.2 Aircraft Tested

2.2.1 Extra 260

Figures 2.3(a–b) shows the top and side views of a model of the aerobatic Extra 260 aircraft, that was custom-built for use with the motion tracking system [30]. The “profile” type aircraft was produced from 2-mm depron foam sheets using a high precision laser cutter. The aircraft had a wingspan of 41.27 cm (16.25 in), length of 39.37 cm (15.50 in) and weight of 33.62 g (1.18 oz).

The primary airframe and control surfaces were supported by carbon rods. Carbon push rods that were connected to the servos enabled the actuation of the ailerons, elevator, and rudder. A 3.7-V battery was used to power a receiver that controlled the servos and the throttle of the aircraft. Thrust was generated by an electric motor driving a propeller with a diameter of 13 cm (5.12 in) and a pitch of 7 cm/rev (2.75 in/rev). The geometric and physical properties for the aircraft are listed in Tables 2.1 and 2.2, respectively.
The second aircraft tested was a 32-g (1.13-oz) Extra 300 3D, as shown in Figs. 2.4(a–b) [31]. The aircraft had a wingspan of 42.67 cm (16.8 in) and an overall length of 49.27 cm (19.4 in). From the top and side views [see Figs. 2.4(a–b)] of the Extra 300 3D, a triangular shaped design pattern is observed on the underside of the wing, horizontal tail, and the right hand side of the vertical tail and fuselage. This unique design pattern helped alleviate the weight of the primary airframe and control surfaces without comprising on the structural integrity of the aircraft. The geometric and physical properties of the Extra 300 3D are listed in Tables 2.3 and 2.4, respectively.
Figure 2.4: Commercially manufactured Extra 300 3D.

<table>
<thead>
<tr>
<th>Table 2.3: Geometric Properties of the Extra 300 3D</th>
</tr>
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<tbody>
<tr>
<td>Mass</td>
</tr>
<tr>
<td>Wingspan</td>
</tr>
<tr>
<td>Wing area ($S_w$)</td>
</tr>
<tr>
<td>Reference area ($S_{ref}$)</td>
</tr>
<tr>
<td>Wing chord (at root)</td>
</tr>
<tr>
<td>Length</td>
</tr>
<tr>
<td>Horizontal tail area</td>
</tr>
<tr>
<td>Vertical tail area</td>
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</table>

<table>
<thead>
<tr>
<th>Table 2.4: Physical Properties of the Extra 300 3D</th>
</tr>
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<tbody>
<tr>
<td>$I_{xx}$</td>
</tr>
<tr>
<td>$I_{yy}$</td>
</tr>
<tr>
<td>$I_{zz}$</td>
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<tr>
<td>$I_{xz}$</td>
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<tr>
<td>$I_{yz}$</td>
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<tr>
<td>$I_{xy}$</td>
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</tbody>
</table>

2.3 Data Acquisition and Processing

The Vicon tracking system provided information on whether the object was visible to the motion capture system and if it was, the attitude and position of the object. Each object (fuselage and wings, propeller, control surfaces) was tracked independently. The data stream was in the Earth referenced coordinate system and needed to be transformed into the aircraft coordinate system. Aircraft attitude, angular rates, velocities, and accelerations were required to analyze the aerodynamic performance.
2.3.1 Calibration

Prior to using the camera system to acquire experimental measurements, calibration of the system was conducted. The process was initiated by setting the appropriate recording speed and starting-up the cameras through the interface software. A 5-marker wand, shown in Fig. 2.5, is provided for the purpose of calibration. As shown in Fig. 2.6, camera masks were setup to prevent the cameras from picking up any stray reflections in the test environment as it would lead to erroneous results. After creating the camera masks, a wand was used to ensure that each camera recorded the prerequisite number of frames. Figure 2.7 shows the process of calibration. The wand is moved in capture volume and is tracked by the cameras. The multi-colored bands (see Fig. 2.7) represent the position track of the wand. The wand was used to set the origin of the test volume. After setting the origin the system was ready for use. Reflective markers were attached to the surface of the aircraft to generate strong reflections. The cameras tracked the circular reflections in their field of view, and using multiple camera views, the Vicon software triangulated the reflections in three dimensions. The Vicon software returns both position and orientation of the aircraft in the Earth-referenced frame, and these data were analyzed to obtain velocities and acceleration in the body-fixed frame. Each part of the aircraft (fuselage, horizontal tail, vertical tail, propeller, etc.) was tracked as an individual object as shown in Fig. 2.8.

![Figure 2.5: The 5-marker wand used for calibration.](image-url)
Figure 2.6: Creation of camera masks prior to calibration.

Figure 2.7: Calibration of the motion tracking system using the 5-marker wand.

Figure 2.8: The Vicon markers built into separate objects with the propeller, the aircraft, and the two control surfaces (left to right) being tracked in the Vicon software.
2.3.2 Filtering

In post processing the raw position and attitude measurements are filtered and numerically differentiated. The voids where the system was unable to triangulate the objects were located and filled-in using linear interpolation based on the neighboring points in the trajectory time history. After filling in these few points that represent less than 0.7% of the time history, the raw measurements were smoothed. A number of smoothing filters that used the full time history were explored. A comparative study was conducted to determine the best filtering and differentiation technique, and the results of the study are presented in Fig. 2.9. In addition to the methods selected for discussion in this section, methods such as Kalman smoothing, a variety of low-pass filters, and higher-order finite difference methods were explored.

In the first method, a third-order polynomial fit was used for a shifting time window of data and is referred to as local polynomial regression smoothing. The resulting polynomial was used to find smoothed measurement as well as the first and second derivate at the center point. The Savitzky-Golay algorithm was used to efficiently calculated the polynomial as well as the derivatives for the position vector and the three Euler angles ($\phi$, $\theta$ and $\psi$) [32–34]. In Fig. 2.9, this result is labeled as polynomial regression.

The second method used was a simple finite difference scheme that was subsequently robustly smoothed. In Fig. 2.9, this method is referred to as robust smoothing. A first-order finite difference scheme that subtracted the
measurements from subsequent time steps was employed in this approach. The resulting derivative had scatter that often included outliers. Thus, a robust smoothing technique that had less emphasis on the outliers was used. After the smoothing, the second derivative was obtained by differentiating and smoothing the first derivative [32].

In addition, the entire range of frequency measurements could be low-pass filtered and downsampled to a lower frequency. Raw measurements were usually taken at 200 Hz and sometimes as high as 300 Hz depending on the nature of the test. It was observed that downsampling the measurement rate to 50 or 100 Hz decreased the noise in the differentiated data. While tracking the propeller, a high sampling frequency was required to prevent aliasing. Hence, downsampling the objects with slow dynamics, such as the aircraft, was particularly helpful.

### 2.3.3 Axis Transformation

The method of data acquisition is similar to that used in earlier research [29]. Each part of the aircraft (fuselage, horizontal tail, vertical tail, propeller) was tracked as an individual object. The data stream provided by the Vicon system included the Earth-referenced position and the Euler angles for each of the objects. The tracking system provided information on the visibility of the object to the camera system and if it was, the attitude and position of the object. The tracking data from each object was filtered to acquire useful measurements.

For the fuselage object, the position and attitude were used in the post processing. The attitude, angular rates, velocities and accelerations were required to analyze the aerodynamic performance of the aircraft. The first step was to transform the raw measured data from object-fixed reference frame as recorded by the tracking system to the center of gravity of the aircraft. By measuring the distance and rotation between the aircraft center of gravity and the object-fixed origin, the rotation and offset between the object measured frame and the aircraft center of gravity body-fixed frame were known. Transformation matrices were used to combine the measured offsets and the Earth-referenced tracking data. Each transformation matrix includes a set of angular offsets \((\theta_o, \phi_o, \psi_o)\) and a set of position offsets \((x_o, y_o, z_o)\) as shown

\[
R = \begin{bmatrix}
\cos \theta_o \cos \psi_o & \cos \theta_o \sin \psi_o & -\sin \theta_o & x_o \\
\sin \phi_o \sin \theta_o \cos \psi_o - \cos \phi_o \sin \psi_o & \sin \phi_o \sin \theta_o \sin \psi_o - \cos \phi_o \cos \psi_o & \sin \phi_o \cos \theta_o & y_o \\
\cos \phi_o \sin \theta_o \cos \psi_o - \sin \phi_o \sin \psi_o & \cos \phi_o \sin \theta_o \sin \psi_o - \sin \phi_o \cos \psi_o & \cos \phi_o \cos \theta_o & z_o \\
0 & 0 & 0 & 1
\end{bmatrix}
\] (2.1)

A matrix was first developed for the transformation from the aircraft object measurement frame to the aircraft center of gravity body-fixed frame. The resulting matrix was labeled \(R_{\text{measured to CG}}\). The second transformation matrix, \(R_{\text{inertial frame}}\), was from the Earth-fixed inertial reference frame to the tracking object center and was recorded at each time step. By combining these two rotations through the multiplication of \(R_{\text{measured to CG}}\) and \(R_{\text{inertial frame}}\).
the transformation from the Earth-fixed reference frame to the aircraft center of gravity was calculated.

\[ \mathbf{R}_{ac} = \mathbf{R}_{\text{measured to CG}} \cdot \mathbf{R}_{\text{inertial frame}} \]  

(2.2)

From the resulting \( \mathbf{R}_{ac} \) matrix, the Earth-referenced attitude and position at the aircraft center of gravity were determined by calculating \( \theta, \phi, \) and \( \psi \) as well as \( x, y, \) and \( z. \)

The second step was to find any time-steps in the data where the system had lost track of the object, which were usually only a few consecutive measurements during a flight. Out of the total flight time of 1–2 sec, typically no more than 0.005–0.025 sec of data were missing. In these regions, a linear fit was used between the measurements at either side of the missing data. After filling in these few points, the noise in the raw measurements was smoothed using the MATLAB\textsuperscript{®} implementation of the robust local regression with a second-order polynomial. In order to limit the effect of any measured points that were significantly off the general trend line, the robust method was chosen. To find the velocity and acceleration, the smoothed data was differentiated using a fourth-order local fitting method developed by Klein and Morelli [34]. For both the position and attitude, the same filtering techniques were used to estimate the aircraft track.

### 2.3.4 Lift and Drag Measurement

From the smoothed and differentiated data, the position of the aircraft along with the velocity and acceleration in the Earth-referenced frame as well as the Euler angles were known. To transform these quantities into a body-fixed reference frame, a rotation matrix (see Eq. 2.1) based on the Euler angles was used [35]. First, the velocity and acceleration were transformed to the body-fixed frame using

\[
\mathbf{V}_b = \begin{bmatrix} u \\ v \\ w \end{bmatrix}^T = \mathbf{R}_{\text{earth to body}} \begin{bmatrix} \dot{x}_E \\ \dot{y}_E \\ \dot{z}_E \end{bmatrix}^T + (\mathbf{\omega} \times \mathbf{r}) \tag{2.3a}
\]

\[
\mathbf{a}_b = \begin{bmatrix} a_x \\ a_y \\ a_z \end{bmatrix}^T = \mathbf{R}_{\text{earth to body}} \begin{bmatrix} \ddot{x}_E \\ \ddot{y}_E \\ \ddot{z}_E \end{bmatrix}^T + \mathbf{\dot{\omega}} \times \mathbf{r} + \mathbf{\omega} \times (\mathbf{\omega} \times \mathbf{r}) \tag{2.3b}
\]

The body-fixed axis velocity and acceleration were obtained by applying the transformations as shown in Eqs. 2.3(a–b). The Euler rates were calculated by transforming the Euler angular rates to the body-fixed angular rates.

\[
\begin{bmatrix} p \\ q \\ r \end{bmatrix} = \begin{bmatrix} 1 & 0 & -\sin \theta \\ 0 & \cos \phi & \sin \phi \cos \theta \\ 0 & -\sin \phi & \cos \phi \cos \theta \end{bmatrix} \begin{bmatrix} \dot{\phi} \\ \dot{\theta} \\ \dot{\psi} \end{bmatrix} \tag{2.4}
\]
and the angular acceleration was found using,

\[
\begin{bmatrix}
\dot{p} \\
\dot{q} \\
\dot{r}
\end{bmatrix} =
\begin{bmatrix}
1 & 0 & -\sin \theta \\
0 & \cos \phi & \sin \phi \cos \theta \\
0 & -\sin \phi & \cos \phi \cos \theta
\end{bmatrix}
\begin{bmatrix}
\ddot{\phi} \\
\ddot{\theta} \\
\ddot{\psi}
\end{bmatrix} +
\begin{bmatrix}
0 & 0 & -\dot{\theta} \cos \theta \\
0 & -\dot{\phi} \sin \phi & -\dot{\theta} \sin \phi \sin \theta + \dot{\phi} \cos \phi \cos \theta \\
0 & -\dot{\phi} \cos \phi & -\dot{\theta} \cos \phi \sin \theta - \dot{\phi} \sin \phi \cos \theta
\end{bmatrix}
\begin{bmatrix}
\dot{\phi} \\
\dot{\theta} \\
\dot{\psi}
\end{bmatrix}
\]

With all of these quantities known over the duration the flight, the analysis of the aerodynamic performance could be completed.

It is essential to define the axis system to calculate the aerodynamic forces acting on the aircraft. As shown in Fig. 2.10(a), the \( X_b \) and \( Z_b \) axes are in the body-fixed frame and are in the plane of symmetry, with the \( X_b \) pointing along the fuselage and the positive \( Y_b \) axis along the right wing. The components of velocity along the \( X_b \), \( Y_b \), and \( Z_b \) axes are \( u \), \( v \), and \( w \), respectively. It is observed from Fig. 2.10(b) that \( V_P \) is the projection of the velocity \( V \) on the \( X_bZ_b \) plane. The angle of attack \( (\alpha) \) and sideslip \( (\beta) \) can be defined in terms of the components of velocity as illustrated in Figs. 2.10(a–b) [36, 37].

To obtain the angle of attack and sideslip angle, the measured inertial speeds can be used if two assumptions were made. First, the air was assumed to be quiescent, and second the induced flow effects on the aircraft were negligible.
With these assumptions, a good estimate of the freestream flow angles was made using

\[
\alpha = \tan^{-1}\left(\frac{w}{u}\right) \quad (2.6a)
\]
\[
\beta = \sin^{-1}\left(\frac{v}{V}\right) \quad (2.6b)
\]

The forces acting on the aircraft were known since the mass of the aircraft was fixed, and the body-fixed axis accelerations \((a_x, a_y, a_z)\) were known from the position tracking data. The total external forces acting on the aircraft were calculated using

\[
\mathbf{F}_{\text{external}} = \begin{bmatrix} a_x & a_y & a_z \end{bmatrix}^T m \quad (2.7)
\]

By subtracting the force of gravity \((\mathbf{F}_G)\) and thrust force \((\mathbf{F}_T)\) from the total external forces, the aerodynamic forces acting on the aircraft were determined.

\[
\mathbf{F}_{\text{aero}} = \mathbf{F}_{\text{external}} - \mathbf{F}_G - \mathbf{F}_T \quad (2.8)
\]

where \(\mathbf{F}_G\) is given by the expression

\[
\mathbf{F}_G = mg \begin{bmatrix} -\sin \theta & \sin \phi \cos \theta & \cos \phi \cos \theta \end{bmatrix}^T \quad (2.9)
\]

Three resulting components of \(\mathbf{F}_{\text{aero}}\) were in the body-fixed axis frame and were due to aerodynamic loading on the aircraft. To calculate lift and drag, which are the force components in the wind axis, the forces in the body frame were transformed into the wind frame using

\[
L = -F_z \cos \alpha + F_x \sin \alpha \quad (2.10a)
\]
\[
D = -F_z \sin \alpha \cos \beta - F_x \cos \beta \cos \alpha - F_y \sin \beta \quad (2.10b)
\]

By not making a small angle approximation on the sideslip angle, \(\beta\), in the drag calculations, the result was more accurate for a maneuvering aircraft. The angle of attack and sideslip angle were calculated throughout the flight to understand the performance of the aircraft.
2.3.5 Moment Measurement

The calculation of the moments generated due to the aerodynamic loads on the aircraft followed a similar approach to that used in Section 2.3.4. Starting with the body-fixed angular rates \( (p, q, r) \) and the moments of inertia, the moments acting on the aircraft were determined using the rotational equation of motion [34] that is

\[
\frac{d^2(I\omega)}{dt^2} = M_{aero}
\]  

(2.11)

The aircraft moments of inertia were used, and the standard assumption to ignore \( I_{xy} \) and \( I_{yz} \) was applied. \( I_{xy} \) and \( I_{yz} \) could be ignored because the aircraft were almost symmetric about these axes, and the values were small. The moments acting on the aircraft were calculated with respect to the aircraft center of gravity.

\[
M_x = \dot{p}I_x - \dot{r}I_{xz} + qr(I_z - I_y) - qpI_{xz} \quad (2.12a)
\]

\[
M_y = \dot{q}I_y + pr(I_z - I_x) + (p^2 - r^2)I_{xz} \quad (2.12b)
\]

\[
M_z = \dot{r}I_z - \dot{p}I_{xz} + pq(I_y - I_x) + qrI_{xz} \quad (2.12c)
\]

Equations 2.12(a–c) represent the roll, pitch, and yaw moments being generated by the aerodynamic forces about the center of gravity. The moments are important to help understand the response of the aircraft to control surface deflections.

2.3.6 Thrust Measurement

A commercially manufactured E-flite™ propeller was used in the current study (see Fig. 2.11). The rotational speed of the propeller was calculated using the attitude time history of the propeller object. The object was tracked using flat markers that were placed on the hub and blades of the propeller. Preliminary challenges were encountered during the calculation of the propeller speed. Tracking the propeller was more difficult than slow moving objects due to loss of
Table 2.5: Parameters Defining the Aircraft State in the Experiments

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Symbols</th>
</tr>
</thead>
<tbody>
<tr>
<td>Time</td>
<td>$t$</td>
</tr>
<tr>
<td>Airspeed</td>
<td>$V$</td>
</tr>
<tr>
<td>Velocity</td>
<td>$u, v, w$</td>
</tr>
<tr>
<td>Acceleration</td>
<td>$a_x, a_y, a_z$</td>
</tr>
<tr>
<td>Dynamic pressure</td>
<td>$q$</td>
</tr>
<tr>
<td>Angle of attack</td>
<td>$\alpha$</td>
</tr>
<tr>
<td>Sideslip angle</td>
<td>$\beta$</td>
</tr>
<tr>
<td>Pitch, roll and yaw angles</td>
<td>$\theta, \phi, \psi$</td>
</tr>
<tr>
<td>Pitch, roll and yaw rates</td>
<td>$p, q, r$</td>
</tr>
<tr>
<td>Drag force</td>
<td>$D$</td>
</tr>
<tr>
<td>Lift force</td>
<td>$L$</td>
</tr>
<tr>
<td>Moments</td>
<td>$M_x, M_y, M_z$</td>
</tr>
<tr>
<td>Control surface deflection</td>
<td>$\delta_e, \delta_r, \delta_a$</td>
</tr>
<tr>
<td>Propeller attitude</td>
<td>$q_1, q_2$</td>
</tr>
<tr>
<td>Propeller rotational speed</td>
<td>$n$</td>
</tr>
<tr>
<td>Propeller advance ratio</td>
<td>$J$</td>
</tr>
</tbody>
</table>

track by the motion tracking system. Despite using fast sampling rates (above the Nyquist frequency) there could be as few as two measurements during each rotation of the propeller.

In order to overcome these challenges, the attitude time history of the propeller was differentiated [38]. and the attitude track of the propeller was represented by quaternions. The quaternions were used for calculating the angular change in propeller attitude ($\delta$).

$$\delta = \frac{2 \cos^{-1}(q_1 \cdot q_2)}{dt}$$  \hspace{1cm} (2.13)

After calculating the angular change in propeller attitude, the rotational speed was calculated by using the time interval between measurements ($dt$).

$$n = \frac{\delta}{2\pi}$$  \hspace{1cm} (2.14)

The thrust from the propeller was calculated from the relation [39]

$$F_T = C_T \rho n^2 D_p^4$$  \hspace{1cm} (2.15)

where $C_T$ is the coefficient of thrust at various advance ratios, $J$ (see Eq. 2.16), $\rho$ is the density of air, $n$ is the rotational speed in revolutions per second, and $D_p$ is the diameter of the propeller.

$$J = \frac{V}{nD_p}$$  \hspace{1cm} (2.16)

In order to perform a thorough analysis, additional parameters such as atmospheric density and mass properties were measured and used in the analysis. A list of all of the recorded and calculated variables is shown in Table 2.5.
2.3.7 Preliminary Thrust Estimate

Reliable experimental data were not readily available for the E-flite propeller. Thus, a well tested propeller with almost the same diameter-to-pitch ratio was chosen to help estimate the thrust produced. Thrust measurements for an APC sport propeller with a diameter of 22.86 cm (9 in) and pitch of 12.70 cm/rev (5 in/rev) were used as the baseline to estimate the thrust produced by the propeller in this study. Figure 2.12 shows the geometric characteristics of the APC Sport propeller. The experimentally determined thrust characteristics of the propeller are shown in Fig. 2.13.

Figures 2.14 and 2.15 show the experimentally measured velocity, advance ratio, and RPM for two flights in the low speed range. The results show the successful measurement of propeller RPM and resulting advance ratios for the
duration of two flights. The thrust produced by the propeller (see Eq. 2.15) was estimated using the measured RPM and the baseline $C_T$ shown in Fig. 2.13. The effect of an offset in the $C_T$ value is discussed in Section 4.2.1.

However, the thrust model discussed in this section is only a preliminary estimate to understand the aerodynamic performance of the aircraft with the propeller running at various speeds. A detailed thrust model will be discussed in the subsequent section of this thesis (see Section 2.3.8).

### 2.3.8 Thrust Model Development

As discussed in Section 2.3.7, experimentally measured thrust at various advance ratios were not available for the E-flite propeller. The absence of these values necessitated the development of an accurate thrust model for the propeller. The new thrust model would be more accurate and help in better understanding the aerodynamic characteristics of MAVs in various powered flight configurations. A step-by-step method for developing a thrust model for the E-flite propeller is discussed in this section.

The geometric characteristics of propellers can be calculated using various methods. Brandt [40, 41] and Tehrani [42] scanned or photographed propellers to obtain twist and chord distributions. Front and a side view images of the propeller served as the input for PropellerScanner [43], which generated a twist and chord distribution. However,
the accuracy of PropellerScanner was never investigated. The geometric characteristics of the E-flite propeller are as shown in Fig. 2.16.

Following the acquisition of the twist and chord distribution, the thrust characteristics of the propeller were determined using the blade element theory. The thrust produced at various advance ratios was calculated and is shown in Fig. 2.17. Figure 2.18 shows the thrust characteristics of the E-flite propeller co-plotted with that of the APC 9×5. The thrust characteristics are co-plotted to highlight the difference in values of $C_T$ at various $J$, between the two thrust models.
Figure 2.16: Geometric characteristics of the E-flite propeller.
Figure 2.17: Thrust characteristics of the E-flite propeller.

Figure 2.18: Comparison of thrust characteristics of the E-flite propeller and APC 9×5.
Chapter 3

System Verification

In order understand the accuracy of measurements from the system and the processing used to calculate the forces, tests were conducted to investigate the noise in the system. First, data from stationary tests were used to analyze the noise in the measurements. Second, a free-falling ball accelerating under gravity was used to verify the acceleration measurements. Finally, repeatability tests were conducted and the results from a hand-built glider were consistent along repeated launches. While noise exists in the final results, the tests show the capability of the system to measure aerodynamic forces.

3.1 Stationary Test Measurements

The uncertainty in measurements from the tracking system was estimated from stationary tests. The aircraft [see Fig. 2.4(b)], with markers attached, was placed on the ground in a fixed position that was easily observed by the Vicon cameras. A time history of the position and attitude was recorded and the standard deviation in each axis was calculated. Figures 3.1 and 3.2 show a histogram of the unfiltered position and attitude from a stationary test. The noise distribution in the data was close to Gaussian for most of the axes, but the measurements often lacked a well-defined peak at the mean. From the results, the noise in the raw measurements was calculated. The standard deviation in three dimensions for the position and attitude were $1.13 \times 10^{-4}$ ft and $9.88 \times 10^{-3}$ deg, respectively. Noise levels varied slightly between the individual axis ($X$, $Y$ and $Z$) and the details are shown in Table 3.1. The standard deviation of the position obtained in this research is consistent with the results reported in literature [27]. However, the accuracy of the attitude measurements is greater in the current research. The difference could be due to the larger dimensions of the aircraft which caused the markers to be placed farther apart, thus reducing noise is in the measurements.

3.2 Free-Fall Measurements

The motion tracking system was used to measure the acceleration of a 5.45-kg (0.37-slug), 5-in diameter shot put (cast iron sphere) in free-fall. The sphere was under the influence of the force due to gravity and aerodynamic drag,
Table 3.1: Standard Derivation of the 100-Hz Measurements and Derivatives

<table>
<thead>
<tr>
<th></th>
<th>X</th>
<th>Y</th>
<th>Z</th>
</tr>
</thead>
<tbody>
<tr>
<td>Position (ft)</td>
<td>$5.81 \times 10^{-5}$</td>
<td>$4.09 \times 10^{-4}$</td>
<td>$8.77 \times 10^{-5}$</td>
</tr>
<tr>
<td>Attitude (deg)</td>
<td>$8.02 \times 10^{-3}$</td>
<td>$4.95 \times 10^{-3}$</td>
<td>$2.97 \times 10^{-3}$</td>
</tr>
</tbody>
</table>

during the free-fall. From theoretical spherical drag calculations in Hoerner [44], the drag force was found to be less than 0.44 N (0.1 lb) which corresponds to a deceleration of 0.08 m/sec$^2$ (0.27 ft/sec$^2$) for the shot put. It is important to note that any offset while tracking the center of the rotating sphere in free-fall contributes to an uncertainty in the calculated velocity. For example, a 1.5-in offset from the center of the sphere can result in a velocity uncertainty of 0.26 ft/sec. The instantaneous accelerations for each time step was found in seven drop tests over two days. The average value of acceleration from these tests was $-9.808$ m/sec$^2$ ($-32.179$ ft/sec$^2$) which was in proximity to the local acceleration due to gravity [$-9.801$ m/sec$^2$ ($-32.157$ ft/sec$^2$)].

The uncertainty in acceleration has an effect on the lift and drag coefficient for an MAV. To better understand the effect, the uncertainty was represented in terms of a force and subsequently in terms of a coefficient for typical cruise flight conditions.

$$C_{L_u} = \frac{ma_u}{0.5\rho V^2 S_{ref}}$$

(3.1)

The uncertainty in the lift coefficient ($C_{L_u}$) is compiled for a range of acceleration uncertainties ($a_u$) as shown in Table 3.2. Even when the acceleration uncertainty is 1 ft/sec, the lift coefficient uncertainty is small. This error
Table 3.2: Effect of Acceleration Uncertainty on $C_L$

<table>
<thead>
<tr>
<th>Acceleration Uncertainty</th>
<th>$C_{L_{\text{uncertainty}}}$</th>
</tr>
</thead>
<tbody>
<tr>
<td>ft/sec$^2$</td>
<td>m/sec$^2$</td>
</tr>
<tr>
<td>1</td>
<td>0.31</td>
</tr>
<tr>
<td>2</td>
<td>0.61</td>
</tr>
<tr>
<td>3</td>
<td>0.91</td>
</tr>
<tr>
<td>4</td>
<td>1.22</td>
</tr>
</tbody>
</table>

Figure 3.3: Hand-launched free-flight glider.

3.3 Repeatability Measurements

A small hand-built glider (see Fig. 3.3) was used to capture repeatability data from the test setup. A rail launching system was used to launch the aircraft at consistent flight conditions. The rail was placed at an inclined angle and a cart accelerated down the track carrying the aircraft. At the end of the track, the cart fell down, while the aircraft started to glide. By adjusting the aircraft trim point, the acceleration of the cart, and the angle of the track, the desired analysis does not include uncertainty in the aircraft attitude and flight path angle which corresponds to how much of the aerodynamic force is lift as opposed to drag.
flight conditions could be achieved. From the motion track, the flight conditions and aerodynamic characteristics of the aircraft could be analyzed.

Figure 3.4 shows the altitude of the aircraft as a function of the distance traveled from the starting point on rail. The second green mark along the trajectory indicates the approximate location of the end of the rail where the aircraft was released from a height of little over 5 ft. It is observed from Fig. 3.4 that the aircraft glides over a distance of 20–25 ft for the 6 flights chosen. The flights end as the aircraft approaches a landing or flies outside of the capture volume. Figures 3.5 and 3.6 show the time history of position and attitude of the glider during acceleration down the rail, launch, and subsequent flight. Each flight had a slight variation in the yaw-axis. The variation was due to inconsistencies in the initial yaw angle between the tests. Each time the aircraft was placed on the cart, the initial position along the yaw axis varied significantly more than the pitch and roll axes. The pitch curve in Fig. 3.6 shows
the onset of a phugoid mode with an approximate period of 1.5 sec. However, due to the limited time window for data capture, the damping effect is not observed. Figure 3.7 shows that the velocity trend of the aircraft in each flight was similar. From further processing of these flights, the lift and drag coefficients were determined. Across the set of tests, both $C_L$ and $C_D$ have some variation within a limited range. Figure 3.8 shows the lift coefficient as a function of angle of attack and shows that some of the variation was due to angle of attack. However, the color pattern on the plot shows that unsteady effects from $\dot{\alpha}$ caused variation in the lift coefficient. The points with pitch-up effect had a larger value of $C_L$ while negative pitching caused a decrease in lift. This set of repeatability flights showed the whole trajectory and the repeatability of the aerodynamic forces being measured by the system.

Another set of rail launched repeatability tests compare results across slightly different launch and trim conditions. The aircraft was launched a number of times from the rail and then the trim conditions were marginally altered. In Fig. 3.9, the first set of flights is represented by red with each flight having a different marker. After the first set of flights, the trim point was adjusted by adding nose weight and adjusting the elevator. A steeper launch angle setting was chosen for the subsequent tests. As expected, the aircraft flew over a smaller range of angle of attack and these flights are shown in green in Fig. 3.9. In the final set, the cart was adjusted to attain higher launch speeds, and the flights were tightly clustered around the trim flight condition, as shown in blue. A set of the lift and drag results for over 50 hand launched flights are co-plotted with the repeatability results. The hand-launched results show good agreement with the repeatability tests for the same aircraft. Figure 3.10 shows a 3D histogram of measured data points as a function of angle of attack and $C_L$. While the flight measurements have variations, the histogram shows that a
vast majority of points clustered around the expected trend lines and scatter is due to outliers.

While the measurements from motioning tracking include noise and variations, the noise was relatively small, and the motion capture system measured the acceleration to within 0.07% of the expected value. Results indicate that measurements from system were repeatable and follow expected aircraft performance trends. In order to minimize the uncertainty in measurements, numerous flights should be used gather data.

Figure 3.7: Time history of the velocity of six repeatable flights.
Figure 3.8: Lift coefficient as a function of $\alpha$ and $\dot{\alpha}$ during the six repeatability flights.
Figure 3.9: Repeatability measurements for an MAV at various trim conditions co-plotted with flight measurements.
Figure 3.10: A 3D histogram showing repeatability measurements of various flights of an MAV.
Chapter 4

Results and Discussion

The results obtained can be classified into two principal sections. The aerodynamic characteristics of the Extra260 and Extra 300 3D in free-flight are presented in Section 4.1. Thrust model development and results from the powered flight tests of the Extra 300 3D are discussed in Section 4.2.

4.1 Unpowered Flight Tests

4.1.1 Lift and Drag Characteristics

The results presented in this section indicate the behavior of the aircraft during hand-launched glide flights. The lift and drag characteristics of the Extra 260 and Extra 300 3D are presented in Figs. 4.1 and 4.2. The data points plotted correspond to lower angular rates \( \dot{\alpha} < 20 \text{ deg/sec}; \dot{\beta}, p, q, r < 60 \text{ deg/sec} \). By limiting the angular rates, unsteady flight dynamics effects could be minimized [13]. The lift curve slope for each aircraft was obtained by performing a least-squares linear fit to the low rate data. A parabolic fit was employed for the low angular rate data in the drag polar.

Lifting line theory was used to calculate the theoretical lift curve slope (Eq. 4.1) for a given \( \mathcal{A} \) without consideration given to aircraft trim [45]

\[
C_{L_{\alpha}} = 2\pi \left( \frac{\mathcal{A}}{2 + \mathcal{A}} \right)
\]  

A linear least-squares fit was applied to the lift curves for the Extra 260 and Extra 300 3D. The experimental lift curve slope for the Extra 260 was 2.58/rad which was lower than the 4.04/rad predicted by lifting line theory (see Eq. 4.1). The experimental result was significantly lower than the value of calculated from lifting line theory. This decrease in value is not unexpected because the lift curve slope decreases for finite wings and airfoils at low Reynolds numbers [12]. A similar trend was observed for the Extra 300 3D that had an experimental lift curve slope of 2.51/rad while the value obtained from lifting line theory was 3.85/rad. It is noted that both aircraft tested had a finite “flat plate” fuselage and that the area of the fuselage \( S_f \) was added to the area of the wing \( S_w \) to obtain the reference area \( S_{ref} \). The area of the fuselage was taken into account while evaluating the \( \mathcal{A} \) and lift curve slopes.
The $\mathcal{R}$ is calculated using

$$\mathcal{R} = \frac{b^2}{S_{ref}}$$  \hspace{1cm} (4.2)

where $b$ is the wingspan and $S_{ref}$ is the reference area. From the wingspans ($b$) and reference areas ($S_{ref}$) provided in Tables 2.1 and 2.3, the $\mathcal{R}$ for the Extra 260 and Extra 300 3D were calculated to be 3.60 and 3.18, respectively.

A parabolic fit was applied to the low angular rate data shown in the drag polars of Figs. 4.1 and 4.2. The parabolic fit is of the form

$$C_D = C_{D_o} + KC_L^2$$  \hspace{1cm} (4.3)

where $C_{D_o}$ is the parasite drag when $C_L$ is zero and $KC_L^2$ is the induced drag due to lift where

$$K = \frac{1}{\pi e_o \mathcal{R}}$$  \hspace{1cm} (4.4)

In Eq. 4.4, $e_o$ is the Oswald efficiency factor [39]. The resulting curve fits for the Extra 260 and Extra 300 3D are shown in Eqs. 4.5 and 4.6, respectively.

$$C_D = 0.0554 + 0.5278C_L^2$$  \hspace{1cm} (4.5)

$$C_D = 0.1015 + 0.5349C_L^2$$  \hspace{1cm} (4.6)

From Eqs. 4.5 and 4.6, it is observed that the $C_{D_o}$ values for the Extra 260 and Extra 300 3D are 0.0554 and 0.1015, respectively. The Extra 300 3D has a higher value of $C_{D_o}$ due to the presence of the propeller and the wing mounts. In addition, the minimum drag value depends on the interaction of various parts of the aircraft with the flow. Reflective markers, nose weights, and wing attachments vary between the two aircraft and account for a portion of the interference and form drags.
Figure 4.1: Experimentally determined drag polar and lift curve for the Extra 260.
Figure 4.2: Experimentally determined drag polar and lift curve for the Extra 300 3D.
Table 4.1: Parameters Used to Calculate the Theoretical Neutral Point Location for Extra 260

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Wing root chord ($c_w$)</td>
<td>13.03 cm (5.13 in)</td>
</tr>
<tr>
<td>Wing area ($S_w$)</td>
<td>403.10 cm$^2$ (62.48 in$^2$)</td>
</tr>
<tr>
<td>Horizontal tail area ($S_t$)</td>
<td>102.77 cm$^2$ (15.93 in$^2$)</td>
</tr>
<tr>
<td>Wing aspect ratio ($A_w$)</td>
<td>4.22</td>
</tr>
<tr>
<td>Horizontal tail aspect ratio ($A_t$)</td>
<td>3.22</td>
</tr>
<tr>
<td>Wing lift curve slope ($C_{L_{\alpha},w}$)</td>
<td>4.26/rad</td>
</tr>
<tr>
<td>Horizontal tail lift curve slope ($C_{L_{\alpha},t}$)</td>
<td>3.87/rad</td>
</tr>
<tr>
<td>Distance from wing LE to wing AC ($l_{ac,w}$)</td>
<td>3.25 cm (1.28 in)</td>
</tr>
<tr>
<td>Distance from wing LE to tail AC ($l_{ac,t}$)</td>
<td>24.13 cm (9.5 in)</td>
</tr>
<tr>
<td>Downwash gradient ($\frac{\partial \epsilon}{\partial \alpha}$)</td>
<td>0.54</td>
</tr>
</tbody>
</table>

4.1.2 Longitudinal Stability

The criterion for longitudinal static stability for an aircraft is achieved when the value of $C_{M_{\alpha}}$ is negative. Thus, an aircraft with $C_{M_{\alpha}} < 0$ can produce a restoring moment when perturbed from the equilibrium condition. Figures 4.3 and 4.5 show three different cases of trim angles of attack with each case including a set of unpowered flights with approximately the same trim angle of attack. It is noted that the location of the aircraft center of gravity is constant for all flights. A linear least-squares fit was applied to find $C_{M_{\alpha}}$ from the pitching moment and angle of attack time history. Figures 4.3 and 4.5 show an increase in $C_{M_{\alpha}}$ with angle of attack, and the effect can be analyzed from the neutral point location.

The neutral point was found from $C_{M_{\alpha}}$, $C_{L_{\alpha}}$, and the measured location of the center of gravity. Using the experimental lift curve slopes determined earlier, the static margin and experimental neutral point can be found from

$$C_{M_{\alpha}} = C_{L_{\alpha}} (SM)$$ (4.7)

where the static margin, $SM$, is the nondimensional difference between the aircraft stick-fixed neutral point and the known center of gravity.

The location of the neutral point relative to the center of gravity determines the static margin and thus, the stability of the aircraft. The theoretical neutral point can be determined using [39, 45]

$$l_{np} = \frac{C_{L_{\alpha},w} l_{ac,w} + C_{L_{\alpha},t} l_{ac,t} \eta_t S_t}{C_{L_{\alpha},w} + C_{L_{\alpha},t} \eta_t S_t} \left( 1 - \frac{\partial \epsilon}{\partial \alpha} \right)$$ (4.8)

where $\frac{\partial \epsilon}{\partial \alpha}$ is the change in the downwash angle with angle of attack, $\eta_t$ is the ratio of the dynamic pressure at the tail to the freestream dynamic pressure, and $C_{L_{\alpha},w}$ and $C_{L_{\alpha},t}$ are the lift curve slopes of the wing and tail, respectively. The lift curve slopes of the wing ($C_{L_{\alpha},w}$) and tail ($C_{L_{\alpha},t}$) are calculated using Eqs. 4.1 and 4.2. However, the areas of the wing ($S_w$) and tail ($S_t$) should be used to calculate the respective aspect ratios ($R_w$, $R_t$), in the denominator.
of Eq. 4.2. The distance from the aerodynamic center of each surface to a reference point (leading edge of the wing) is \(l_{ac}\) and it is normalized by the wing root chord \(c_w\). The parameters used to calculate the location of the neutral point of both aircraft are presented in Tables 4.1 and 4.2. A 3-view CAD drawing of the Extra 260 and Extra 300 3D, illustrating key dimensions, are shown in Figs. 4.4 and 4.6. From Eq. 4.8, the theoretical location of the neutral point for the Extra 260 and Extra 300 3D were found to be 43% and 40% of the root chord, respectively. Figures 4.7 and 4.8 show the experimentally determined neutral point locations for both aircraft. Results show that at low angles of attack, the neutral point is farther forward. As the angle of attack increases, the neutral point moves aft which implies that the effect of the tail is significant with respect to the wing. The shift aft could be due to various effects that influence the performance, particularly the lift curve slope of the wing and/ or the horizontal tail.
Figure 4.4: CAD drawing of the Extra 260.
Table 4.2: Parameters Used to Calculate the Theoretical Neutral Point Location for Extra 300 3D

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Wing root chord ( (c_w) )</td>
<td>13.33 cm (5.25 in)</td>
</tr>
<tr>
<td>Wing area ( (S_w) )</td>
<td>467.41 cm(^2) (72.45 in(^2))</td>
</tr>
<tr>
<td>Horizontal tail area ( (S_t) )</td>
<td>159.67 cm(^2) (24.75 in(^2))</td>
</tr>
<tr>
<td>Wing aspect ratio ( (A_w) )</td>
<td>3.89</td>
</tr>
<tr>
<td>Horizontal tail aspect ratio ( (A_t) )</td>
<td>2.76</td>
</tr>
<tr>
<td>Wing lift curve slope ( (C_{L\alpha,w}) )</td>
<td>4.15/( \text{rad} )</td>
</tr>
<tr>
<td>Horizontal tail lift curve slope ( (C_{L\alpha,t}) )</td>
<td>3.64/( \text{rad} )</td>
</tr>
<tr>
<td>Distance from wing LE to wing AC ( (l_{ac,w}) )</td>
<td>3.32 cm (1.31 in)</td>
</tr>
<tr>
<td>Distance from wing LE to tail AC ( (l_{ac,t}) )</td>
<td>24.63 cm (9.7 in)</td>
</tr>
<tr>
<td>Downwash gradient ( (\partial\epsilon/\partial\alpha) )</td>
<td>0.56</td>
</tr>
</tbody>
</table>

Figure 4.5: Experimentally determined pitching moment versus angle of attack for the Extra 300 3D.
Figure 4.6: CAD drawing of the Extra 300 3D.
The lift curve slope ($C_{L,\alpha}$) decreases gradually at higher angles of attack and will cause the neutral point to shift aft. Generally, the tail operates at a lower local angle of attack owing to the downwash from the main wing. Assuming similar lift curve behavior for the horizontal tail, $C_{L,\alpha,t}$ will also decrease at higher local angles of attack. However, the influence of downwash causes $C_{L,\alpha,t}$ to decrease at a higher freestream angle of attack. Thus, as $C_{L,\alpha,w}$ decreases with respect to $C_{L,\alpha,t}$, the neutral point of the aircraft moves aft. As mentioned earlier, $C_{L,\alpha,w}$ and $C_{L,\alpha,t}$ are calculated using lifting line theory.

The induced flow effects from the wake of the main wing on the horizontal tail affects the neutral point location. Downwash and velocity deficit caused by the drag of the main wing changes the flow at the horizontal tail. The theoretical formulation includes a $\frac{\partial \epsilon}{\partial \alpha}$ term that assumes downwash effect is constant with angle of attack. As the flight conditions, particularly angle of attack, change the wake of the main wing can shift which in turn alters the effect on the horizontal tail. In order to account for the shift in the wake and subsequent effect on the horizontal tail, both $\frac{\partial \epsilon}{\partial \alpha}$ and $\eta_t$ should be changed with angle of attack in Eq. 4.8.

In the theoretical calculations, the aerodynamic center of each of the surfaces is assumed to be fixed at the quarter chord. However, wind tunnel studies have shown that at low Reynolds numbers the pitching moment about the quarter chord does not remain constant [3–6]. Thus, there will be a change in the aerodynamic center of each of the lifting surfaces resulting in a change in the static margin. The change in the aerodynamic center at low Reynolds numbers needs to be incorporated into the theoretical formulation.
Figure 4.7: Experimentally determined neutral point location versus trim angle of attack for the Extra 260.

Figure 4.8: Experimentally determined neutral point location versus trim angle of attack for the Extra 300 3D.
4.1.3 Lateral Stability

**Yaw Stability**

The stick fixed lateral motion of an airplane is a complex combination of roll, yaw, and sideslip. The yawing motion of an aircraft is stabilized by the presence of a vertical tail, and yaw stability is determined by the yawing moment due to the sideslip angle, $C_{N\beta}$. For an aircraft to be stable $C_{N\beta}$ should be positive, i.e. $C_{N\beta} > 0$. An aircraft with $C_{N\beta} > 0$ is said to have weathercock stability and restores the aircraft to a zero sideslip condition. In the current research, yaw stability derivatives are obtained for the Extra 260 and Extra 300 3D, in the unpowered flight condition, over a range of trim angles of attack. From Fig 4.9, it is observed that the Extra 260 has a $C_{N\beta}$ between 0 and 0.08/ rad. Figure 4.10 shows that the Extra 300 3D exhibited an increase in weathercock stability with angle of attack. In general, the Extra 260 had a superior weather vane stability when compared with the Extra 300 3D.

**Roll Stability**

Roll due to yaw, $C_{l\beta}$, is the other major lateral stability derivative for an aircraft and is often referred to as the dihedral effect. An aircraft is said to be stable in roll if, when it is disturbed from a wings-level attitude, a restoring moment is developed. The restoring rolling moment is a function of the sideslip angle $\beta$ and the dihedral of the wings. The requirement for roll stability is $C_{l\beta} < 0$. The position of the wing, wing dihedral, vertical tail, and horizontal stabilizer influence the roll stability. In the current research, roll stability derivatives are obtained for the Extra 260 and Extra 300 3D, in the unpowered flight condition. Figure 4.11 shows the roll stability of the Extra 260 over a range of trim angles of attack. The aircraft has a $C_{l\beta}$ ranging from 0.01 to $-0.02$/rad. It is observed that the aircraft stabilizes in roll at higher angles of attack. From Fig. 4.12, it is observed that the Extra 300 3D has negative $C_{l\beta}$ at high angles of attack. Some of the roll instability in both aircraft, at low angles of attack, could be due to the absence of dihedral, mid-body wings, and large elevators.
Figure 4.9: Experimentally determined weathercock stability ($C_{N_{\beta}}$) as a function of trim angle of attack for the Extra 260.

Figure 4.10: Experimentally determined weathercock stability ($C_{N_{\beta}}$) as a function of trim angle of attack for the Extra 300 3D.
Figure 4.11: Experimentally determined roll stability ($C_{l_{\beta}}$) as a function of trim angle of attack for the Extra 260.

Figure 4.12: Experimentally determined roll stability ($C_{l_{\beta}}$) as a function of trim angle of attack for the Extra 300 3D.
4.2 Powered Flight Tests

The propeller wake over the aircraft wing and fuselage has a significant impact on the lift, drag, and stability of the aircraft. The thrust has three effects: the direct moment from the thrust, the propeller normal force due to the turning of air, and the influence of the propwash upon the tail, wing, and aft fuselage. A propeller produces thrust by accelerating a large mass of air rearwards, and this air flows over a certain portion of the wing and fuselage. The total lift produced is a combination of the lift generated by the airframe area not in the wake of the propeller and the region influenced by the propeller slipstream. Thus, the lift can be altered by changing the rotational speed (RPM) of the propeller. In the current study, a commercially manufactured E-flite propeller with a diameter of 13 cm (5.12 in) and pitch of 7 cm/rev (2.75 in/rev) was mounted on the nose of the aircraft. The aircraft was tested with the propeller running within three speed (rotational) settings. The three speed ranges were low (3,000–4,000 RPM), moderate (4,000–5,000 RPM), and high (5,000–6,600 RPM). The lift and drag characteristics of the aircraft, at the low speed setting, is presented in this section.

4.2.1 Preliminary Thrust Results

Owing to the absence of experimental data for the E-flite propeller, a well tested propeller was chosen to estimate the thrust produced (see Section 2.3.6). Thrust measurements for an APC sport propeller (see Appendix B) with a diameter of 22.86 cm (9 in) and pitch of 12.70 cm/rev (5 in/rev) were used as the baseline to estimate the thrust produced by the propeller in this study.

The lift and drag characteristics of the Extra 300 3D, with the propeller running at the low speed range, are shown in Fig. 4.13, respectively. A least-squares fit was performed on the data points corresponding to low angular rates. Prior to choosing the baseline $C_T$ for thrust calculations, various thrust models of the same propeller were tested by providing a reasonable offset from the baseline value. The effect of this offset on the lift and drag characteristics of the aircraft is shown in Fig. 4.14, respectively. While the lift curve slope showed negligible change, a slight shift in the position of the drag polar was observed. Figure 4.15 shows a comparison of the lift and drag characteristics of the Extra 300 3D in glide and powered flight configurations. From lift curve in Fig. 4.15, it is observed that there is an increase in the slope of the lift curve and a delay in the onset of stall when the propeller is running. The increase in lift slope and delayed stall can be attributed to the propeller slipstream which energizes the flow over parts of the wing and fuselage, consequently delaying flow separation. The drag polar in Fig. 4.15 shows that glide and powered flight configurations of the Extra 300 3D exhibit the same drag characteristics at low angles of attack and deviate as the angle of attack increases.
Figure 4.13: Experimentally determined drag polar and lift curve of the Extra 300 3D in powered flight.
Figure 4.14: Variation in experimentally determined drag polar and lift curve for the Extra 300 3D with $C_T$ offset.
Figure 4.15: Comparison of experimentally determined drag polar and lift curve for the Extra 300 3D in glide and powered flight.
Table 4.3: Parameters Used to Calculate the Change in Drag Coefficient

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Reference area ($S_{ref}$)</td>
<td>571.99 cm$^2$</td>
</tr>
<tr>
<td>Propeller area ($S_{prop}$)</td>
<td>9.74 cm$^2$</td>
</tr>
<tr>
<td>Flat plate drag ($C_{D_{FP}}$)</td>
<td>2.0</td>
</tr>
<tr>
<td>Drag change ($\Delta C_D$)</td>
<td>0.034</td>
</tr>
</tbody>
</table>

4.2.2 Results after Thrust Model Implementation

As discussed in Section 2.3.8, a thrust model was developed for an E-flite propeller. The forces due to the propeller running at various speeds, were accounted for by obtaining the coefficient of thrust ($C_T$) at various advance ratios ($J$). Following this, the lift and drag forces acting on the aircraft were obtained. The aerodynamic performance of the aircraft is presented for three ranges of $J$: low (0.30–0.45), moderate (0.45–0.60), and high (0.60–1.0).

The aerodynamic performance of the Extra 300 3D at various $J$ is compared with the lift and drag characteristics during glide flight. In order to make this comparison the drag from the propeller needs to be taken into consideration. The change in drag coefficient ($\Delta C_D$) can be calculated using,

$$\Delta C_D = C_{D_{FP}} \frac{S_{prop}}{S_{ref}}$$

where $C_{D_{FP}}$ is the coefficient of drag for a flat plate, $S_{prop}$ is the frontal area of the propeller and $S_{ref}$ is the reference area. From Eq. 4.9, the $\Delta C_D$ was found to be approximately 0.034. This change in the value of $C_D$ is shown in Fig. 4.16. The parameters used in Eq. 4.9 are listed in Table 4.3

Figures 4.17–4.19 show the lift and drag characteristics of the Extra 300 3D at various ranges of $J$ and a comparison is made with glide flight behavior. Glide flight behavior is chosen as a reference to highlight the change in aerodynamic characteristics owing to propeller induced flow. It is observed from Figs. 4.17–4.19, that as the advance ratio decreases the lift curve gets steeper. The increase in the lift curve slope is a consequence of enhanced aerodynamic performance of the wing and fuselage immersed in the propeller wake. The propeller wake imparts an increased axial velocity that results in beneficial lift characteristics. The lift curve in Fig 4.20 clearly shows the trend of augmented lift with decreasing $J$ (increasing propeller RPM). A delay in the onset of stall is a phenomenon that is associated with wings/aircraft in the propeller induced flow. However, constraints in the test environment limit the angle of attack range that could be achieved in the powered flight configuration. Increased lift coefficient is a significant phenomenon, especially when it comes to low speed flight performance. The phenomenon of increased lift would permit an MAV to attain a higher angle of attack before aerodynamic stall and have improved aerodynamic performance that what would be thought possible from wing aerodynamics alone.

The drag polars in Figs. 4.17–4.19 show the drag characteristics of the Extra 300 3D at various ranges of $J$. The
drag polar plots clearly show a change in the aerodynamic performance of the aircraft at various $J$. From Fig. 4.20, it is observed that as the $J$ range progresses from high to low, the drag characteristics of the aircraft depart from that of glide flight. This effect could be as a result of the modification of the local angles of attack on the wing and fuselage by the tangential velocity in the propeller wake. The effect of the propeller slipstream on the laminar wing boundary layer could lead to the alternation of the boundary layer between laminar and turbulent states [46]. This alternation could be attributed to the periodic external flow turbulence of the propeller wake. The alternating laminar-turbulent boundary layer could result in lower drag.

During glide flight, data were collected uniformly over a large range of angles of attack. Owing to the uniformity in the distribution of data points, a least-squares linear and parabolic fit were applied to the glide flight data as shown in the lift curve and drag polar plots in Fig. 4.2. However, due to dimensional constraints imposed by the test environment, the amount of data that could be collected over the entire angle of attack range were limited in the powered flight configuration. Thus, the data point density was calculated for each powered flight configuration and the feasibility of using a least-squares fit was assessed. The data point density was obtained by creating a $5000 \times 5000$ square mesh. The mesh was refined and the best size was arrived at after taking into consideration the desired level of
accuracy and computation time. This mesh enabled the figure to be divided into a number of square “bins” containing data points. A “count” of the number of data points within each bin was obtained and a color scheme was chosen to show the variation in the count. From the color scheme, a varied distribution of data points could be easily visualized.

Figures 4.21 and 4.27 shows the drag and lift characteristics of the Extra 300 3D at the low $J$ range and Figs. 4.22 and 4.28 show the corresponding density of data points. From Fig. 4.22, it is observed that the maximum density of points occurs in the $C_L$ range of 0.2–0.4. A similar observation can be made from the lift curve in Fig. 4.28. Figures 4.23 and 4.29 show the drag and lift characteristics and Figs. 4.24 and 4.30 show the corresponding density of data points at the moderate $J$ range ($J = 0.45–0.60$). Figure 4.24 shows an aggregation of data points in the $C_L$ range of 0.25–0.35. However, Fig. 4.30 shows a more uniform spread of data points in the $C_L$ range of 0.15–0.35 with clumps of data points at $C_L = 0.15$ and 0.25. The drag and lift characteristics at the high $J$ range ($J = 0.6–1.0$) are shown in Figs. 4.25 and 4.31. The corresponding data point densities are shown in Figs. 4.26 and 4.32. From Fig. 4.26, it is observed that there is a concentration of data points in the $C_L$ range of 0.35–0.50. A similar observation can be made from Fig. 4.32 with the exception of a few high density spots at $C_L = 0.10$ and 0.75.

Figures 4.33–4.35 show the nature of the least-squares fit applied to the corresponding data sets. As seen from lift curve in Fig. 4.33, the least-squares method creates a linear fit through the region of maximum point density. The linear fit is not influenced by the data points in the $C_L$ range of 0.50–0.80 as they have low point density values. Thus, the slope from a linear least-squares fit is shallower than what is observed from the nature of the lift curve. From the lift curve in Fig. 4.34, it is observed that a linear least-squares fit is applied in the $C_L$ range of 0.15–0.50. The linear fit passes through regions having the highest data point density (see Fig. 4.30). The distribution of data points is more uniform in the $C_L$ range of 0.15–0.50, as shown in Fig. 4.30. However, owing to a lack of data points at the low and high angles of attack, an accurate prediction of the lift curve slope and occurrence of stall cannot be made. A similar conclusion can be drawn from Fig. 4.35, that the linear fit under predicts the slope based on the density of data points and does not follow the true nature of the curve. As mentioned earlier, flight data could not be obtained at the low (0–4 deg) and high (15–30 deg) angle of attack ranges owing to space limitations resulting in an incomplete lift curve. For the drag data (Figs. 4.33–4.35), a least-squares parabolic fit was applied. Though the fit follows the nature of the curve, the scarcity of flight data at low and high values of $C_L$ results in an inaccurate prediction of $C_{Do}$ and induced drag effects. Thus, for the purpose of comparison of the lift and drag characteristics of the aircraft at various powered flight configurations and relative to glide, the fit line is ignored.
Figure 4.17: Comparison of experimentally determined drag polar and lift curve for the Extra 300 3D in glide and powered flight at $J = 0.30 - 0.45$. 
Figure 4.18: Comparison of experimentally determined drag polar and lift curve for the Extra 300 3D in glide and powered flight at $J = 0.45–0.60$. 
Figure 4.19: Comparison of experimentally determined drag polar and lift curve for the Extra 300 3D in glide and powered flight at $J = 0.60 - 1.0$. 
Figure 4.20: Comparison of experimentally determined drag polar and lift curve for the Extra 300 3D in glide and powered flight at various $J$. 

Extra 300 3D
+ Glide
* $J = 0.30 − 0.45$
△ $J = 0.45 − 0.60$
 ○ $J = 0.60 − 1.0$
Figure 4.21: Experimentally determined drag polar for the Extra 300 3D at $J = 0.30-0.45$.

Figure 4.22: Data point density of experimentally determined drag polar for the Extra 300 3D at $J = 0.30-0.45$. 
Figure 4.23: Experimentally determined drag polar for the Extra 300 3D at $J = 0.45 - 0.60$.

Figure 4.24: Data point density of experimentally determined drag polar for the Extra 300 3D at $J = 0.45 - 0.60$. 
Figure 4.25: Experimentally determined drag polar for the Extra 300 3D at $J = 0.60 - 1.0$.

Figure 4.26: Data point density of experimentally determined drag polar for the Extra 300 3D at $J = 0.60 - 1.0$. 
Figure 4.27: Experimentally determined lift curve for the Extra 300 3D at $J = 0.30–0.45$.

Figure 4.28: Data point density of experimentally determined lift curve for the Extra 300 3D at $J = 0.30–0.45$. 
**Figure 4.29**: Experimentally determined lift curve for the Extra 300 3D at $J = 0.45–0.60$.

**Figure 4.30**: Data point density of experimentally determined lift curve for the Extra 300 3D at $J = 0.45–0.60$. 
Figure 4.31: Experimentally determined lift curve for the Extra 300 3D at $J = 0.60 - 1.0$.

Figure 4.32: Data point density of experimentally determined lift curve for the Extra 300 3D at $J = 0.60 - 1.0$. 
Figure 4.33: Experimentally determined drag polar and lift curve for the Extra 300 3D at $J = 0.30–0.45$. 
Figure 4.34: Experimentally determined drag polar and lift curve for the Extra 300 3D at $J = 0.45-0.60$. 
Figure 4.35: Experimentally determined drag polar and lift curve for the Extra 300 3D at $J = 0.60–1.0$. 
5.1 Conclusions

In order to better understand the aerodynamic performance of fixed-wing MAVs, a motion tracking system was used to obtain flight data. For this research tests were conducted using a custom built Extra 260 and commercially manufactured Extra 300 3D. The Extra 260 was tested in glide flight while the Extra 300 3D was tested in free and powered flight configurations. The results obtained show interesting aerodynamic characteristics of MAVs which warrant further studies.

Initial glide flight results indicated that the aerodynamic performance of fixed-wing MAVs is critically dependent on the operating Reynolds numbers and wing geometry. Lift and drag characteristics of the Extra 260 and Extra 300 3D were obtained for unpowered flight without any considerations given to aircraft trim. Theoretical lift curve slopes were obtained using lifting line theory. The experimental lift curve slopes were found to be 64–65% of the values obtained from lifting line theory. However, a detailed trim analysis would need to be conducted in order to obtain the lift curve slope of the wing and thus, make an accurate comparison of the experimental lift curve slope with the theoretical values. These results provided useful information regarding the aerodynamics of MAVs in free-flight.

Tests were conducted to obtain the longitudinal and lateral stability derivatives of both aircraft. The aircraft were configured to fly at pre-determined trim angles of attack and longitudinal stability characteristics of the aircraft were obtained from the pitching moment and angle of attack time history. Results showed that both aircraft were longitudinally stable ($C_{M_n} < 0$) and the location of the neutral point was experimentally determined. The yaw ($C_{N_{\beta}}$) and roll ($C_{l_{\beta}}$) stability derivatives were obtained as functions of the trim angle of attack. It was observed that the Extra 260 exhibited superior weathercock stability ($C_{N_{\beta}} > 0$) when compared to the Extra 300 3D. The results for roll stability showed that both aircraft exhibited increased stability at higher angles of attack.

Powered flight tests were conducted for the Extra 300 3D, with the propeller running at various speeds. A preliminary estimate thrust estimate was made to account for the forces from the propeller. A thrust model was developed due to the absence of performance data for a commercially manufactured E-flite propeller. The thrust model was implemented and aided in the accurate calculation of the lift and drag forces at various ranges of $J$. 
The results showed interesting aerodynamic trends such as an increase in the lift curve slope and reduction in drag with decrease in $J$.

5.2 Future Work

As this research draws to a close, many new ideas and enhancements come to mind. Some of these suggestions and ideas can be addressed in the future and are discussed in this section. A thorough investigation on the effects of low Reynolds numbers on the aerodynamic performance of fixed-wing MAVs could be conducted. Currently, the tested airplanes fly in a Reynolds number range of 15,000–30,000. Expanding this range to include lower Reynolds numbers, perhaps as low as 5,000 would show the effects at even smaller scales.

In the current work the control surfaces were pre-deflected to a desired angle before testing due to the inability to obtain clear aerodynamic performance trends when the control surfaces were deflected during flight. A technique could be devised such that the pilot is able to attain the same amount of control surface deflection throughout the flight. Controlled flight tests would aid in understanding the dynamic stability and control problem for MAVs.

More testing during maneuver and high angle of attack flight is needed to characterize the unsteady effects of maneuvering flight. The testing could be carried out with additional airplane configurations. In addition, testing techniques should investigate the possibility of trying to achieve stable, high angle of attack descent (deep stall). These conditions would allow steady state high angle of attack flight lift and drag to be measured. In addition, visualization techniques could be explored to see if any of the techniques can be applied to MAV in free-flight. Flow visualization would provide more of an understanding of how the flow interacts with the airplane during high angle of attack flight (as well as low angle of attack flight).

Powered flight data could be collected for a variety of MAVs and the trends observed could be compared. The effect of the change in $AR$ on the lift and drag characteristics could be obtained and compared with results from wind tunnel studies. A detailed study on the development of the thrust model would help establish the margin of error and thus, the sensitivity of the model to correction factors that could be applied. Another area of flight that is relatively unexplored is MAVs in hover. This flight regime demands superior piloting skills and a better track of the aircraft and propeller during hover and hover-to-level flight transition.

The system verification section of this thesis discussed the various data reduction and processing techniques. However, there is room for improvement and new filtering methods could be used to process the data. These methods might reduce the noise in the measurements and help arrive at more accurate results.
Appendix A

Tabulated Propeller Geometry

Appendix A details the geometric characteristics of the propellers used in the experiment. The chord and pitch distributions for both propellers were obtained by digitizing them with the PropellerScanner software. Propeller digitization was achieved by using both a top–view and side–view image of a propeller at high resolution. Inputs to the program included the thickness ratios and the true diameter of the propeller, while the sectional chord value at the 75% radial station was returned. The geometric characteristics of the propellers used are shown in Figs. 2.12 and 2.16.
### APC
Sport propeller
9 \times 5

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<th>c/R</th>
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### E-flite
E-flite propeller
5.11 \times 2.75

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Appendix B

Tabulated Propeller Performance Data

Appendix B contains the performance data for all the propellers used in this research. The thrust coefficient data for the APC 9×5 and E-flite propellers are listed over a range of advance ratios. The thrust characteristics of the APC 9×5 were obtained from experimental results while the characteristics of the E-flite propeller were determined using blade element theory.
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Sport propeller
9×5

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### E-flite
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5.11×2.75

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References


