Minimum complexity guidance, navigation, and control for an autonomous parafoil payload delivery system

Branden James Rademacher

Iowa State University

Follow this and additional works at: https://lib.dr.iastate.edu/rtd

Recommended Citation

This Thesis is brought to you for free and open access by the Iowa State University Capstones, Theses and Dissertations at Iowa State University Digital Repository. It has been accepted for inclusion in Retrospective Theses and Dissertations by an authorized administrator of Iowa State University Digital Repository. For more information, please contact digirep@iastate.edu.
Minimum complexity guidance, navigation, and control for an autonomous parafoil payload delivery system

by

Branden James Rademacher

A thesis submitted to the graduate faculty
in partial fulfillment of the requirements for the degree of

MASTER OF SCIENCE

Major: Aerospace Engineering

Program of Study Committee:
Ping Lu, Major Professor
Jerald Vogel
John Basart

Iowa State University
Ames, Iowa
2005

Copyright © Branden James Rademacher, 2005. All rights reserved.
This is to certify that the master’s thesis of

Branden James Rademacher

has met the thesis requirements of Iowa State University

Signatures have been redacted for privacy
# TABLE OF CONTENTS

<table>
<thead>
<tr>
<th>Section</th>
<th>Pages</th>
</tr>
</thead>
<tbody>
<tr>
<td>ABSTRACT</td>
<td>ix</td>
</tr>
<tr>
<td><strong>1. INTRODUCTION</strong></td>
<td>1</td>
</tr>
<tr>
<td>The Recovery Guidance System</td>
<td>3</td>
</tr>
<tr>
<td>Mission Profile</td>
<td>4</td>
</tr>
<tr>
<td>Previous Research Sponsored by the SSOL</td>
<td>5</td>
</tr>
<tr>
<td>Current Research</td>
<td>6</td>
</tr>
<tr>
<td><strong>2. PARAFOIL DYNAMICS AND MODELING</strong></td>
<td>8</td>
</tr>
<tr>
<td>Modeling Issues and Qualitative Behavior</td>
<td>8</td>
</tr>
<tr>
<td>Axes Systems and Transformations</td>
<td>14</td>
</tr>
<tr>
<td>Apparent Mass Effects</td>
<td>16</td>
</tr>
<tr>
<td>Six DOF Parafoil Equations of Motion</td>
<td>20</td>
</tr>
<tr>
<td>Aerodynamic Model</td>
<td>24</td>
</tr>
<tr>
<td>Actuator Model</td>
<td>26</td>
</tr>
<tr>
<td><strong>3. NAVIGATION</strong></td>
<td>28</td>
</tr>
<tr>
<td>Navigation Sensors</td>
<td>28</td>
</tr>
<tr>
<td>Sensor Selection</td>
<td>30</td>
</tr>
<tr>
<td>On-board Wind Estimation</td>
<td>31</td>
</tr>
<tr>
<td><strong>4. CONTROL</strong></td>
<td>38</td>
</tr>
<tr>
<td>Model Linearization</td>
<td>40</td>
</tr>
</tbody>
</table>
Course Angle Dynamics 44
Transfer Function Models 45
Heading Angle Control Design 46
Course Heading Control Design 49

5. GUIDANCE 51
Guidance Algorithm Development 53

6. INTEGRATED GNC SIMULATIONS 58
Wind Profiles 58
Simulation Study 61

7. SUMMARY AND CONCLUSIONS 70

8. RECOMMENDATIONS 73

APPENDIX: AERODYNAMIC DERIVATIVES AND GEOMETRIC CONSTANTS 75

REFERENCES 77
LIST OF FIGURES

Figure 1.1: The Recovery Guidance System in flight during a low-altitude drop test. 4

Figure 2.1: X-38 Parafoil with large symmetric deflection of the rear flaps. 10

Figure 2.2: Longitudinal geometry of a parafoil in an equilibrium glide. 11

Figure 2.3: Comparing the feasible landing region for two values of L/D in calm conditions. In both cases, the parafoil was deployed at (0,0) and an altitude of 5000 ft. 13

Figure 2.4: Comparison of the feasible landing region for windy and non-windy conditions at constant L/D. In each case, the parafoil was deployed at (0,0) and an altitude of 5000 ft. 14

Figure 2.5: The body-axis coordinate frame, shown on NASA’s SpaceWedge vehicle. 15

Figure 2.6: Front view of the parafoil geometry showing the relative location of the pitch and roll centers, the confluence point, and the center of gravity. 18

Figure 2.7: Second-order nonlinear actuator model with rate and position limits. 27

Figure 3.1: Horizontal geometry used in the wind estimation algorithm. 31

Figure 3.2: Data flow for the nonlinear 6 DOF simulation to generate truth data for the modeled sensor data. 33

Figure 3.3: Ground track of the trajectory followed by the parafoil/payload system. 33

Figure 3.4: Data flow for generating the modeled sensor data. 34

Figure 3.5: Data flow for the wind estimation algorithm. 35

Figure 3.6: Estimated and actual wind velocity using modeled sensor data. 35

Figure 3.7: Estimated and actual wind heading using modeled sensor data. The discontinuities are due to the wind heading being resolved to ±180 degrees. 36
Figure 3.8: Heading and velocity estimation error magnitudes using modeled sensor data.

Figure 4.1: The relationship between the heading angle $\Psi$ and the course angle $\Psi_v$.

Figure 4.2: Root locus plot for the open loop transfer function of a proportional heading control design.

Figure 4.3: Root locus plot for the open loop transfer function of a proportional plus integral heading control design.

Figure 4.4: Comparison of the response to a step input in the heading angle command between the linear and nonlinear model.

Figure 4.5: Comparison of the response to a step input in the command of the course angle for various choices of the control gain.

Figure 4.6: Comparison of the actual and measured response to a step input in the course angle command for the design gain of 0.07.

Figure 5.1: An example of the trajectory generated by the guidance algorithm.

Figure 5.2: The relationship between the target position and the commanded and actual course angle.

Figure 6.1: Data flow for integrated guidance, navigation, and control simulation.

Figure 6.2: Wind profile velocity vs. altitude.

Figure 6.3: Wind profile heading vs. altitude.

Figure 6.4: Ground track of a trajectory generated by the guidance algorithm in calm conditions.

Figure 6.5: 3-D profile of a trajectory generated by the guidance algorithm in calm conditions.

Figure 6.6: Zoomed in view of the ground track of the energy management and final approach phases for a trajectory generated by the guidance algorithm under calm conditions.
Figure 6.7: Zoomed in view of the 3-D profile of the energy management and final approach phases for a trajectory generated by the guidance algorithm under calm conditions.

Figure 6.8: Polar scatter plot of landing sites relative to the target (center) for calm wind conditions.

Figure 6.9: Polar scatter plot of landing sites relative to the target (center) for wind profile 1.

Figure 6.10: Polar scatter plot of landing sites relative to the target (center) for wind profile 2.

Figure 6.11: Polar scatter plot of landing sites relative to the target (center) for wind profile 3.

Figure 6.12: Polar scatter plot of landing sites relative to the target (center) for wind profile 4.

Figure 6.13: Polar scatter plot of landing sites relative to the target (center) for wind profiles 1-4.

Figure 6.14: Ground track of a trajectory that enters the final approach phase immediately upon reaching the target for wind profile 4.
LIST OF TABLES

Table 3.1: Parameters used for modeling the sensors. 34
Table 4.1: Damping and natural frequency of the open-loop linear system model. 44
Table 6.1: Statistical analysis of the miss distance for multiple simulation cases. 65
Table A.1: Aerodynamic stability derivatives. 75
Table A.2: Geometric data and other constants. 76
ABSTRACT

A guided ram-air parachute, or parafoil, offers a lightweight and efficient means for the autonomous placement of payloads to specified ground coordinates. The technology has a wide variety of applications, including advanced precision airdrop operations and the safe return of high altitude balloon or sounding rocket payloads. Current systems rely on complicated and expensive sensor suites, and computationally intensive guidance and control algorithms. In the present work, an integrated guidance, navigation, and control scheme is presented which utilizes inexpensive “off-the-shelf” sensors and computationally simple guidance, navigation, and control (GNC) algorithms. The GNC scheme is intended for implementation on a Peripheral Interface Controller (PIC). The GNC scheme is intended to serve as the primary GNC scheme for a small-scale parachute system, but could equally well serve as a backup for a more complicated system in the event of CPU or sensor failure. Navigation data is collected from a GPS receiver and a 3-axis digital compass, sampled at 1 Hz and 2.5 Hz, respectively. The wind speed and heading are estimated using the supplied navigation data and an estimate of the vehicle airspeed. It is shown that an estimation accuracy of better than 3 ft/s for velocity and 10 degrees for heading can be achieved without additional filtering. A linear proportional feedback controller is designed to control the vehicle course angle. It is shown that the effect of the slow sample rate of the navigation sensor data acts as a filter in the forward loop and contributes non-minimum phase zeros to the closed-loop dynamics. The guidance algorithm generates a series of inertially fixed waypoints consisting of three distinct phases: initial homing, energy management, and final approach. The algorithm requires no a priori information about the wind profile. A six degree-of-freedom model is used to evaluate the performance of the integrated GNC algorithm. Monte-Carlo simulations were conducted for a number of wind profiles which were specifically chosen to test various design parameters. The combined results show a
circular error precision (CEP) landing accuracy of 115 ft. It is shown that the primary factors limiting landing accuracy are the maximum actuator deflection and the low system bandwidth.
1. INTRODUCTION

A guided ram-air parachute offers a lightweight and efficient means for the autonomous placement of payloads to specified ground coordinates. This technology has a wide variety of applications, including many that could support NASA’s new space exploration initiative and the Army’s precision airdrop requirements. This technology offers a relatively inexpensive means for the safe return of cosmic dust or extraterrestrial rock and soil samples, space experiments, or reusable supplies. These could be returned from the space shuttle or the new Crew Exploration Vehicle, the International Space Station, or probes similar to the Genesis probe. Closer to home, the technology could offer a more reliable means of returning high altitude balloon or sounding rocket payloads.

The ram-air parachute, or parafoil, belongs to the class of gliding parachutes. The gliding parachute offers many advantages over conventional vertically descending parachutes when used for the precise placement of airdropped payloads. Gliding parachutes have a nonzero lift-to-drag ratio which allows them to achieve horizontal velocities that may exceed their vertical velocity. Conventional parachute systems have little or no ability to be steered once deployed, while gliding parachutes can generate useful if not large turn rates. The combination of horizontal velocity and steerability allows the gliding parachute to minimize or eliminate landing accuracy errors due to wind drift, decreases the required size of the drop zone, and significantly increases the size of the launch envelope for a given target.

The low weight and small volume of the parafoil when packed lends its versatility and performance to many aerospace applications. This is especially true when light wing loading is desired only for a portion of the mission. These missions could include the return of sounding rocket or high altitude balloon payloads which would spend much of their return trajectory under a relatively highly loaded drogue chute, and deploy the parafoil at lower
altitudes to land at a predetermined landing site. The mission profile of the X-38 Crew
Return Vehicle used a relatively highly loaded lifting body for reentry through the upper
portions of the atmosphere and deployed a parafoil for the final descent and landing. The
Genesis probe was intended to deploy a parafoil and then autonomously maintain a circular
flight path to aid in aerial capture.

The use of autonomously guided parachutes for precise placement of payloads is not a
new concept. SSE Incorporated and ParaFlite developed the ParaPoint System in the 1960’s.
This system relied on a radio beacon placed at the target and used a pair of antennae to home
in on the target [1]. The U.S. Army Natick Research and Development Center began
development of technologies for autonomously guided parafoils in the early 1970’s [2]. In
the early 1990’s the Global Positioning System was opened to the public, offering a
relatively inexpensive and very reliable means of autonomous navigation. This spurred more
systems to begin development. In 1991 SSE Incorporated began development of the ORION
system which was later completed in 1995 [1]. In 1994, NASA began a program to
determine the feasibility of using a ram-air parachute for the autonomous recovery of
spacecraft during the final stages of reentry [3]. This program began as the Spacecraft
Autoland Project using the SpaceWedge vehicle and a sub-scale parafoil. The program
eventually grew into the full-scale X-38 Crew Recovery Vehicle program. In 2003, Atair
Aerospace demonstrated its Onyx system [4].

It is evident from the literature that the development of these programs can be time
consuming and expensive. The most effective means of data acquisition seems to be
repetitive drop testing. Theoretical methods and wind tunnel data are sparse. Added to that,
parafoil system parameters are just plain difficult to measure. The guidance program also
has to work with a number of system limitations. The vehicle airspeed is of the same order
as the speeds of the wind likely to be encountered. The maneuverability is limited in two
respects: First, the maximum turn rate is limited, and for larger parafoils this can be less than
12-15 degrees per second [5]. Second, the flight path angle can only be weakly controlled. The system is not powered, further limiting the trajectory options. Once these hurdles are overcome, the resulting systems are quite effective.

The Recovery Guidance System

The Spacecraft Systems and Operations Laboratory (SSOL) at Iowa State University (ISU) is currently developing a small-scale parafoil delivery system. Called the Recovery Guidance System, the project aims to return high altitude balloon payloads to safe landing zones. A picture of the system can be seen in Figure 1.1.

The High Altitude Balloon Experiments in Technology (HABET) program takes scientific and engineering payloads to altitudes of 65,000-120,000 ft, carried by helium-filled latex balloons. The SSOL allows students to design and build satellite systems and components. The HABET program provides a means to simulate the operation of those systems in a space-like environment. The current recovery system consists of a standard circular parachute. With this configuration, the landing site is largely determined by the prevailing winds. This poses two main problems. First, it complicates the recovery effort, as the possible range for the landing site covers a large area. Second, there is nothing to stop the payload from landing in population centers, restricted or undesirable areas, or in such a manor as to cause injury or damage to person or property. Landing in undesirable areas such as lakes or rivers or on major highways or railroad tracks can result in loss of or damage to the payload. The motivation for the Recovery Guidance System is to protect the payload, private property, and people while aiding the recovery effort in limiting the possible landing sites.
Mission Profile

HABET missions are typically launched from Howe Hall on the ISU campus. The ground track of the balloon varies from mission to mission, as it is dependent on the local winds. The balloon may burst more than 50 miles downrange from the launch site. The proposed mission profile calls for a standard circular parachute or a smaller drogue chute to take the payload from burst altitude to around 20,000 ft. The payload will drift further downrange during this initial descent. Near 20,000 ft the parafoil will be deployed and the RGS system will come online. From this altitude the maximum cross range distance that the system can travel will be less than 9.5 miles. This will require the system to have a database of acceptable landing sites on-board. When the system comes online it will choose the “best” landing site based on its current position and any information available about the wind. Once
the landing site is selected, the system will autonomously land at that site. The goal is to achieve a CEP (circular error probable) landing accuracy of 50 yards or less. The development in this thesis will not consider the creation of the landing site database, or the selection of the landing site. Further, it will be assumed that the landing site is known and within range of the system’s capabilities at the time of deployment.

Previous Research Sponsored by the SSOL

The Recovery Guidance System (RGS) is an ongoing research project in the SSOL, and has been studied by three previous graduate students. Their work has centered on investigating guidance and control concepts and developing an aerodynamic model of the RGS parafoil.

In 1997, the feasibility of implementing fuzzy-logic and bang-bang controllers with proportional-navigation (PN) guidance was studied by John Bendle [6]. Bendle’s simulations were based on a general parafoil model with a planform area of 30 ft$^2$ and a payload weight of 25 lb. The simulations were run with linearized equations of motion. His work showed acceptable results in calm air conditions, but extremely poor results in the presence of winds exceeding 10 mph. It is believed by the author that this was largely due to the use of linearized motion dynamics.

In 1999 a linear quadratic regulator (LQR) controller with proportion-navigation (PN) guidance was studied by H. Bruce Tsai [7]. Tsai’s simulations were based on parafoil parameters estimated using the Vortex-Lattice Method and the dynamics were based on the linearized aircraft equations of motion. Tsai’s simulations were only run for calm conditions.

In 2004 Joshua Bowman began the process of building the parameter model and determining the performance of the RGS parafoil [8]. Bowman conducted many uncontrolled flight tests, varying the parafoil rigging angle and string configuration. He was
able to estimate the steady-state lift and drag coefficients of the RGS parafoil/payload system.

**Current Research**

The previous research in the SSOL did not consider the navigation task. The sensors in the RGS hardware have limited sample rates and have an additional latency in the signal. As will be shown, this has a significant impact on the closed-loop dynamics.

The goal of the current research is to develop a complete guidance, navigation, and control (GNC) scheme that relies on inexpensive off-the-shelf navigation sensors and accounts for the effects of the quantized and delayed nature of the navigation data. The design will require minimal computation power, and minimize the amount of required information about the system. The final design will run on a peripheral interface controller (PIC), which has significantly less computing power and power consumption than an embedded central processing unit (CPU). The GNC concept proposed could be used as a backup GNC algorithm for a more complex system in the event of sensor or CPU failure.

The remainder of this thesis will show the development of the proposed guidance, navigation, and control algorithms. In Chapter 2, the issues with modeling and simulation of parafoil systems are described, and then a nonlinear six degree-of-freedom model is described. Chapter 3 begins with a discussion of navigation sensors, then describes the sensors used in the RGS system, and finishes with the development of an online wind estimation algorithm. In Chapter 4 a linear proportional controller is designed. The chapter begins with the tradeoffs associated with the choice of the controlled variable. Next, numerical linearization is discussed and the nonlinear model developed in Chapter 2 is trimmed and linearized. The chapter closes with the design of heading and course-angle control systems. Chapter 5 begins with a discussion on the guidance algorithms used by
other autonomous parafoil systems and then develops the proposed guidance algorithm. Chapter 6 details a rigorous simulation study of the combined GNC scheme. Concluding remarks and recommendations for further work are given in Chapters 7 and 8, respectively. The aerodynamic derivatives and geometric constants used in all simulations are given in the Appendix.
2. PARAFOIL DYNAMICS AND MODELING

Modeling Issues and Qualitative Behavior

A parafoil with a suspended payload is an extremely complex dynamical system. The parafoil itself is non-rigid and is in fact highly aeroelastic. The shape of the parafoil is determined by many factors including the rigging and control line tensions (i.e. stretch), the control line deflection, the spatially varying pressure forces acting on the canopy surface, and any stretching of the canopy fabric itself. Traditionally constant stability derivatives such as $C_{l_e}$, $C_{l_o}$ and $C_{M_0}$ can change with time as a result in fluctuations in those factors. In addition to these complications with the parafoil aerodynamics, the payload can move with respect to the parafoil. This changes the location of the center of gravity (c.g.) of the parafoil/payload system. In general the vector from the c.g. to the center of pressure, (the point where aerodynamic forces are applied) will vary, as well as the moments of inertia about the c.g. [9].

Accurate computational fluid dynamic (CFD) modeling of a parafoil requires powerful parallel multi-processor computers [10]. This high level of accuracy is often only needed for certain phases of the flight such as canopy deployment, stage de-reefing, and flare maneuvers. For steady gliding flight, the variation of the aerodynamic forces can be assumed small. Relatively accurate simulation results and flight reconstructions have been shown even when these aerodynamic variations are neglected completely [11]. Accurate simulation of canopy deployment and stage de-reefing is well outside the scope of this thesis, and from this point on we will assume that we are dealing with a fully deployed parafoil after the canopy opening transient effects have disappeared.
Many models of varying complexity have been developed to describe the motion of the parafoil and payload system. Point-mass models are useful for describing the spatial motion of the system and are often used to simulate and test guidance concepts. Rigid-body models either treat the parafoil and payload as a single rigid body or as two independent rigid bodies attached together at some point (the confluence point). These models are useful for describing the response of the vehicle to disturbances and control input and are often used to test control and navigation concepts. Models of 8 degree-of-freedom (DOF) and 9 DOF are common for multi-body models, and models of 3 DOF to 6 DOF are used for single-body models [12]. The navigation sensors and algorithm developed in this thesis (discussed in a later chapter) are independent of the relative motion between the parafoil and payload. Thus a 6 DOF model will capture enough detail of the motion and will be used for modeling and simulation purposes.

Longitudinal control is achieved by symmetric deflection of the rear flaps. Longitudinal control effectiveness is primarily reduced to velocity control. Large symmetric deflections of the flaps can increase the amount of lift significantly but seem to have little effect on the lift-to-drag ratio (LID) [13]. There is very limited control effectiveness on the flight path angle. More discussion on the effects of the LID ratio can be found below. A picture showing the X-38 parafoil with a large symmetric deflection can be seen in Figure 2.1. Note the curvature of the airfoil shape on the outer edge.
There are two methods of lateral control. The first involves pulling in the lines on the edge of the parafoil inboard of the desired turn. This produces a lateral pressure differential across the span and effectively tilts the aerodynamic force vector into the turn. The second method is an asymmetric deflection of the rear flaps. This creates a yaw moment and a sideslip in the direction of the desired turn. The sideslip generates a side force that banks the canopy into the turn [14]. This method of control is the most common and will be the one discussed in this thesis.

The turning behavior is dependent on the size of the parafoil. Small parafoils can generate large turn rates and do so at high roll angles. Large-scale parafoils such as those used on the X-38 generate small turn rates and at low roll angles. In both cases the roll angle is less than the roll angle for a coordinated turn, with small parafoils closer to that coordinated roll angle. A consequence of this turning behavior is that the sink rate for smaller parafoils is much more affected in a turn than that for larger parafoils as the sink rate is proportional to the cosine of the bank angle [15].
Equilibrium Gliding Flight: Effects of W/S and L/D

Consider the point-mass model of a parafoil in equilibrium flight. The geometry is shown in Figure 2.2. The conditions required to satisfy an equilibrium glide are:

\[ L \sin \gamma - D \cos \gamma = 0 \]  
\[ W - L \cos \gamma - D \sin \gamma = 0 \]

where \( L \) and \( D \) are the lift and drag forces, respectively, \( W \) is the system weight, and \( \gamma \) is the flight path angle. In Figure 2.2 \( V \) is the velocity of the vehicle. From the first equation it is immediately apparent that

\[ \tan \gamma = \frac{1}{L/D} \]  

If you move the lift and drag terms in the second equation to the other side, then square and combine both equations, you can see that

\[ W = \left( L^2 + D^2 \right)^{1/2} \]
In terms of non-dimensional coefficients,

\[ W = 0.5 \rho V^2 S \left( C_L^2 + C_D^2 \right)^{1/2} \]  \hspace{1cm} (2.5)

where \( C_L = \frac{L}{0.5 \rho V^2 S} \), \( C_D = \frac{D}{0.5 \rho V^2 S} \), \( \rho \) is the air density, and \( S \) is the wing area. Or in terms of the so-called tangent coefficient,

\[ W = 0.5 \rho V^2 S C_T \]  \hspace{1cm} (2.6)

where \( C_T^2 = C_L^2 + C_D^2 \). The velocity may therefore be written as

\[ V = \left( \frac{2 W}{\rho S C_T} \right)^{1/2} \]  \hspace{1cm} (2.7)

Equations (2.3) and (2.7) describe the flight path angle and velocity of a parafoil in steady gliding flight. In still-air conditions the flight path angle or gliding efficiency and, consequently, the horizontal distance that can be traversed is a function only of the lift-to-drag ratio. The velocity is influenced by the air density and the wing loading \((W/S)\), but not by \(L/D\). Thus, for a given opening altitude, a parafoil will cover the same horizontal distance regardless of density and wing loading variations, but the velocity will vary along the trajectory in still-air conditions [16]. In the absence of wind, the “feasible landing footprint” or region that the parafoil can actually land in is a circle centered on the position the parafoil was deployed. The size of the circle is dependent only on the opening altitude and the \(L/D\) ratio. Increasing \(L/D\) and increasing opening altitude will increase the diameter of the circle. This can be seen qualitatively in Figure 2.3.
Figure 2.3: Comparing the feasible landing region for two values of L/D in calm conditions. In both cases, the parafoil was deployed at (0,0) and an altitude of 5000 ft.

In the presence of wind, the landing footprint is still circular, but the circle is no longer centered on the opening position of the parafoil. The amount of shift of the center of the landing region circle is dependent on W/S for a given altitude-density profile. The shift will be smaller for large values of W/S because the time of flight is lower, lessening the amount of wind-induced drifting.

A corollary to the feasible landing footprint is the feasible drop zone. This is the area the parafoil can be deployed in order to reach a specified target on the ground. The analysis described above is the same for both cases.
Comparison of Feasible Landing Region Varying W/S

Figure 2.4: Comparison of the feasible landing region for windy and non-windy conditions at constant L/D. In each case, the parafoil was deployed at (0,0) and an altitude of 5000 ft.

Axes Systems and Transformations

The parafoil equations of motion will require the use of and transformations between three axes systems. The first axis system has its origin fixed on the surface of the Earth and the orientation is fixed with the x-axis pointing north, the y-axis pointing east, and the z-axis pointing vertically down. This is referred to as the NED coordinate system and will be used as the inertial reference frame. The second axis system has its origin at the confluence point of the parafoil/payload system with the x-, y-, and z-axis pointing out of the front, right, and bottom respectively (see Figure 2.5). This axis system is referred to as the body-frame and translates and rotates with the parafoil/payload system. The third axis system also has the origin at the confluence point and can rotate relative to the parafoil/payload system. The frame is defined by successively rotating the body-frame axes about the y-axis by the angle of attack and the z-axis by the sideslip angle. With these rotations, the x-axis is aligned with the velocity vector. This axis system is referred to as the wind-frame.
The position in the NED frame and orientation of the body-frame with respect to the NED-frame will be denoted by

\[
p_o = \begin{bmatrix} p_N \\ p_E \\ p_D \end{bmatrix}
\] (2.8)

\[
\phi = \begin{bmatrix} \phi \\ \theta \\ \psi \end{bmatrix}
\] (2.9)
where $\phi$, $\theta$, and $\psi$ are the Euler roll, pitch, and yaw angles respectively. Using a standard 3-2-1 rotation sequence, the transformation matrix from the NED-frame to the body-frame is given in [17] as

\[
C_{b/n} = \begin{bmatrix}
\cos \theta \cos \psi & \cos \theta \sin \psi & -\sin \theta \\
-\cos \phi \sin \psi + \sin \phi \sin \theta \cos \psi & \cos \phi \cos \psi + \sin \phi \sin \theta \sin \psi & \sin \phi \cos \theta \\
\sin \phi \sin \psi + \cos \phi \sin \theta \cos \psi & -\sin \phi \cos \psi + \cos \phi \sin \theta \sin \psi & \cos \phi \cos \theta 
\end{bmatrix}
\]

(2.10)

The rotation from the body-frame to the wind-frame is obtained by successively rotating by the angle of attack about the pitch axis and the sideslip angle about the yaw axis. The transformation matrix is given by

\[
C_{w/b} = \begin{bmatrix}
\cos \alpha \cos \beta & \sin \beta & \sin \alpha \cos \beta \\
-\cos \alpha \sin \beta & \cos \beta & -\sin \alpha \sin \beta \\
-\sin \alpha & 0 & \cos \alpha 
\end{bmatrix}
\]

(2.11)

The transformation matrices are orthogonal so that

\[
C_{n/w} = C_{w/b}^{-1} = C_{b/n}^{-1}
\]

(2.12)

Thus, the reverse transformation matrix can be found easily by taking the transpose of the original transformation matrix.

**Apparent Mass Effects**

A parafoil is a lightly loaded vehicle with the lifting surface significantly displaced from the center of real mass. This configuration is strongly influenced by apparent mass effects as
not only are the apparent mass terms large, they also act at the center of apparent mass which is displaced from the center of real mass. Lissaman [18] describes the origin of apparent mass effects,

“A moving body immersed in a fluid sets the fluid itself into motion. Changes in the body motion require changes in the external fluid motion and thus introduce pressure forces on the body that are called apparent mass pressures. For standard air vehicles of normal relative mass (mass of vehicle divided by mass of air volume of dimension related to the body) these apparent mass terms are relatively small, less than a few percent of the actual mass. For lightly loaded vehicles, with wing loadings on the order of less than [1 psf], the apparent mass terms can be very significant.”

The affects of the apparent mass terms will in general make it impossible to find a single point where the rotational and translational motions are decoupled. Furthermore, the apparent moments of inertia about the principle pitch and roll axes act at the pitch and roll centers respectively. About any other axis the inertia terms will be larger. The roll center is described in [19] as the point along the yaw axis that the roll axis must pass through for the roll inertia to assume its minimum value. The description of the pitch center is similar. Figure 2.6 shows a front view of the parafoil geometry.
Figure 2.6: Front view of the parafoil geometry showing the relative location of the pitch and roll centers (p and r), the confluence point (c), and the center of gravity (c.g.).

If the origin of the body-frame axes is placed at the confluence point c, and it is assumed the confluence point lies on the line connecting the pitch and roll centers, the vectors connecting the origin to these points have the first two components equal to zero when expressed in the body-frame. The values of the third components can be approximated by:

\[ a_{pc} = a_{po} = \frac{R \sin \Theta}{\Theta} \]  \hspace{1cm} (2.13)

\[ a_{rc} = a_{ro} = \frac{a_{pc} m_{f22}}{m_{f22} + I_{f11} / R^2} \]  \hspace{1cm} (2.14)

where \( R \) and \( \Theta \) are defined in Figure 2.6. The apparent mass terms for a flat wing given in [18] are
\begin{align}
\dot{m}_{f11} &= k \pi \left( t^2 b / 4 \right) \quad (2.15a) \\
\dot{m}_{f22} &= k \pi \left( t^2 c / 4 \right) \quad (2.15b) \\
\dot{m}_{f33} &= \left[ AR / (1 + AR) \right] \pi \left( c^2 b / 4 \right) \quad (2.15c) \\
I_{f11} &= 0.055 \left[ AR / (1 + AR) \right] b S^2 \quad (2.15d) \\
I_{f22} &= 0.0308 \left[ AR / (1 + AR) \right] c^3 S \quad (2.15e) \\
I_{f33} &= 0.055 b^3 t^2 \quad (2.15f)
\end{align}

where \( c \) is the chord length, \( t \) is the ratio of the thickness to the chord length, \( AR \) is the aspect ratio, \( b \) is the wing span, and \( S \) is the planform area. The \( f \) in the subscript specifies that the apparent mass term corresponds to a flat wing, and the numbers in the subscript indicates the component’s index in the apparent mass matrix. The equations for a cambered wing are given in [19] as

\begin{align}
\dot{m}_{11} &= k A \left[ 1 + \left( 8 / 3 \right) h^\ast \right] \pi \left( t^2 b / 4 \right) \quad (2.16a) \\
\dot{m}_{22} &= \left( R^2 m_{f22} + I_{f11} \right) / a_{pc}^2 \quad (2.16b) \\
\dot{m}_{33} &= m_{f33} \quad (2.16c) \\
I_{11} &= \left( a_{pc}^2 / A_{pc}^2 \right) R^2 m_{f22} + \left( A_{pc}^2 / a_{pc}^2 \right) I_{f11} \quad (2.16d) \\
I_{22} &= I_{f22} \quad (2.16e) \\
I_{33} &= 0.055 \left( 1 + 8 h^\ast \right) b^3 t^2 \quad (2.16f)
\end{align}

where

\[
h^\ast = \frac{h}{b} = \frac{R(1 - \cos \Theta)}{2 R \sin \Theta} \equiv \frac{\Theta}{4}
\]  

(2.17)

The apparent moments of inertia about the body-frame axes centered at the confluence point are
\[
\begin{align*}
J_{ao11} &= I_{11} + a^2_{ro} m_{22} \\
J_{ao22} &= I_{22} + a^2_{ro} m_{11} + a^2_{rp} m_{11} + 2a_{ro} a_{rp} m_{11} \\
J_{ao33} &= I_{33}
\end{align*}
\] (2.18a, 2.18b, 2.18c)

It is important to note that all apparent mass quantities need to be multiplied by the local air density to yield the actual apparent mass and inertia components. With the apparent mass terms now defined, the apparent mass and inertia matrices become

\[
M_a = \begin{bmatrix}
m_{11} & 0 & 0 \\
0 & m_{22} & 0 \\
0 & 0 & m_{33}
\end{bmatrix}
\] (2.19)

\[
J_{oa} = \begin{bmatrix}
J_{ao11} & 0 & 0 \\
0 & J_{ao22} & 0 \\
0 & 0 & J_{ao33}
\end{bmatrix}
\] (2.20)

**Six DOF Parafoil Equations of Motion**

In this section the following notation will be used: Lower case bold will represent 3×1 column vectors, uppercase bold will represent a matrix quantity, and lower case normal will represent scalar quantities. Vectors with a superscript cross will represent a cross product matrix:

\[
\mathbf{v}^\times = \begin{bmatrix}
0 & -v_3 & v_2 \\
v_3 & 0 & -v_1 \\
-v_2 & v_1 & 0
\end{bmatrix}
\] (2.21)

where the 1, 2, and 3 subscripts correspond to the x, y, and z axes respectively.
Assuming a flat, non-rotating Earth, the dynamic equations of motion given in [19] are:

\[
\begin{bmatrix}
\dot{v}_o \\
\omega
\end{bmatrix} = \left[ A_{nl} + A_{oa} \right]^{-1} \begin{bmatrix}
f_{ac} + f^{\text{ext}} + f_{Rnl} + f_{a nl} \\
g_{ac} + g_{ext} + g_{Rnl} + g_{a nl}
\end{bmatrix}
\]  

(2.22)

Each variable will be explained in detail below. The velocity of the body frame origin \(v_o\) and the angular velocity with respect to the inertial axis \(\omega\), both expressed in the body-frame, are given in component form as:

\[
v_o = \begin{bmatrix}
u \\
v \\
w
\end{bmatrix}
\]  

(2.23)

\[
\omega = \begin{bmatrix}
p \\
q \\
r
\end{bmatrix}
\]  

(2.24)

The aerodynamic force and moment vectors are defined by:

\[
f_{ac} = \begin{bmatrix}
x_{ac} \\
y_{ac} \\
z_{ac}
\end{bmatrix} = \begin{bmatrix}
-D \\
-C \\
-L
\end{bmatrix}
\]  

(2.25)

\[
g_{ac} = \begin{bmatrix}
f_{ac} \\
m_{ac} \\
n_{ac}
\end{bmatrix}
\]  

(2.26)

The external force and external moment can be given by:

\[
f_{ext} = g \begin{bmatrix}
-sin \theta \\
\sin \phi \cos \theta \\
\cos \phi \cos \theta
\end{bmatrix}
\]  

(2.27)
\[ g_{\text{ext}} = r^*m_R g \begin{bmatrix} -\sin \theta \\ \sin \phi \cos \theta \\ \cos \phi \cos \theta \end{bmatrix} \]  
(2.28)

where \( g \) is the gravitational acceleration. The real and apparent mass nonlinear forces are

\[ f_{R,nl} = \omega^* M_a \left( \mathbf{v}_a - r^*\omega \right) \]  
(2.29)
\[ f_{a,nl} = \omega^* M_a \left( \mathbf{v}_a - D\omega \right) \]  
(2.30)

where \( r \) is the vector from the confluence point to the c.g. and

\[ D = a_{ro}^* + a_{pr}^* S_2 \]  
(2.31)
\[ S_2 = \begin{bmatrix} 0 & 0 & 0 \\ 0 & 1 & 0 \\ 0 & 0 & 0 \end{bmatrix} \]  
(2.32)

The real and apparent mass nonlinear moments are

\[ g_{R,nl} = m_R^* \mathbf{v}_a^* \omega - \omega^* J_{ao}^* \omega \]  
(2.33)
\[ g_{a,nl} = \mathbf{v}_o^* M_a D\omega + \omega^* M_a D^T \mathbf{v}_o - \omega^* J_{oa}^* \omega \]  
(2.34)

Under the assumptions outlined in the previous section, the apparent mass matrix is

\[ A_{ao} = \begin{bmatrix} m_{11} & 0 & 0 & 0 & -m_{11} a_{po} & 0 \\ 0 & m_{22} & 0 & m_{22} a_{ro} & 0 & 0 \\ 0 & 0 & m_{33} & 0 & 0 & 0 \\ 0 & m_{22} a_{ro} & 0 & J_{oa11} & 0 & 0 \\ -m_{11} a_{po} & 0 & 0 & 0 & J_{oa22} & 0 \\ 0 & 0 & 0 & 0 & 0 & J_{oa33} \end{bmatrix} \]  
(2.35)
If we assume the body has right/left and fore/aft symmetry, $J_{oR12}$ and $J_{oR23}$ are zero. If it is further assumed that the center of gravity lies along the line containing the roll center, pitch center and origin, the first two components of $r$ expressed in the body frame are 0. Then the real mass matrix can be written as

$$
\mathbf{A}_{oR} = \begin{bmatrix}
  m_R & 0 & 0 & 0 & m_Rr_3 & 0 \\
  0 & m_R & 0 & -m_Rr_3 & 0 & 0 \\
  0 & 0 & m_R & 0 & 0 & 0 \\
  0 & -m_Rr_3 & 0 & J_{oR11} & 0 & -J_{oR13} \\
  m_Rr_3 & 0 & 0 & 0 & J_{oR22} & 0 \\
  0 & 0 & 0 & J_{oR13} & 0 & J_{oR33}
\end{bmatrix} \quad (2.36)
$$

The kinematic and navigation equations relating the orientation and position of the parafoil system in the inertial frame are

$$\mathbf{p} = C_{n\mathbf{b}} \mathbf{v}_o \quad (2.37)$$

$$\mathbf{\dot{\phi}} = H\mathbf{\omega} \quad (2.38)$$

where

$$
H = \begin{bmatrix}
  1 & \tan \theta \sin \phi & \tan \theta \cos \phi \\
  0 & \cos \phi & -\sin \phi \\
  0 & \sin \phi / \cos \theta & \cos \phi / \cos \theta
\end{bmatrix} \quad (2.39)
$$
Aerodynamic Model

The aerodynamic forces are dependent on the relative velocity between the parafoil/payload system and the air. Define the relative velocity vector in the body-frame as

\[
\begin{align*}
v_{rel} &= v_o - C_{b/o} v_W = \begin{bmatrix} u' \\ v' \\ w' \end{bmatrix} = C_{b/w} \begin{bmatrix} V_T \\ 0 \\ 0 \end{bmatrix} = \begin{bmatrix} V_T \cos \alpha \cos \beta \\ V_T \sin \beta \\ V_T \sin \alpha \cos \beta \end{bmatrix}
\end{align*}
\]

(2.40)

where \( v_w \) is the wind velocity vector defined in the NED frame and \( V_T = |v_{rel}| \). The aerodynamic angle of attack and sideslip angle are then defined by

\[
\begin{align*}
\tan(\alpha) &= \frac{w'}{u'} \\
\sin(\beta) &= \frac{v'}{V_T}
\end{align*}
\]

(2.41 and 2.42)

The dimensional forces and moments are given by

\[
\begin{align*}
L &= q_\infty S C_L \\
D &= q_\infty S C_D \\
C &= q_\infty S C_Y \\
\ell_{aero} &= q_\infty S b C_\ell \\
m_{aero} &= q_\infty S c C_m \\
n_{aero} &= q_\infty S b C_n
\end{align*}
\]

(2.43a - 2.43f)

where \( q_\infty \) is the dynamic pressure \( 0.5 \rho V_T^2 \). The non-dimensional force and moment coefficients are calculated from
where $\delta_e$ and $\delta_a$ are symmetric and asymmetric flap deflections respectively. The quadratic coefficient in the drag coefficient equation is given by

$$k = \frac{1}{\pi eAR}$$

(2.45)

where $e$ is the span efficiency factor and $AR$ is the aspect ratio of the parafoil [20]. The line drag coefficient assuming a small angle of attack is given in [16] as

$$C_{D_{line}} = \frac{nRd}{S}$$

(2.46)

where $n$ is the number of lines and $R$ and $d$ are the length and diameter of the lines respectively. The drag coefficient of the payload is given as
\[ C_{\text{payload}} = \frac{(C_D S)_S}{S} \]  

(2.47)

where \((C_D S)_S\) is the drag coefficient of the payload multiplied by the area of the payload.

The density variation with altitude is modeled as

\[ t_{\text{ref}} = 1 + 6.875 \left(10^{-6}\right) p_D \quad \text{(2.48a)} \]
\[ \rho_a = \rho_0 t_{\text{ref}}^{4.2561} \quad \text{(2.48b)} \]

and is valid to 36,000 ft.

**Actuator Model**

The actuator has rate and position limits. The transfer function for the servo model without the nonlinearities is given by:

\[ \frac{u(s)}{\delta_a(s)} = \frac{\omega_n^2}{s^2 + 2\zeta \omega_n s + \omega_n^2} \]  

(2.49)

The natural frequency \(\omega_n\) and the damping coefficient \(\zeta\) were chosen to be 10 Hz and 0.7, respectively. A block diagram of the model with nonlinearities is given in Figure 2.7.
Figure 2.7: Second-order nonlinear actuator model with rate and position limits.
3. NAVIGATION

The navigation system processes sensor information and provides usable data to the guidance and control systems. From a guidance standpoint, the state of the parafoil/payload system is completely specified by the position vector with respect to the target, the inertial velocity vector, and the airspeed vector [21]. Depending on the control system design, more data may be needed. The airdrop systems described in Chapter 1 use a variety of sensors to obtain all or some of this data. Common sensors or sensor suites utilized include GPS receivers, Inertial Navigation Systems (INS) which utilize rate gyros and accelerometers, integrated GPS/INS systems, digital compasses, radar/laser altimeters, pressure altimeters, and air data systems.

Navigation Sensors

GPS receivers can output position and velocity to a high degree of accuracy. The GPS architecture makes the errors in the GPS navigation solution drift free and bounded [22]. Typical position errors are less than 15 meters for standard GPS receivers, 3-5 meters for Differential GPS (DGPS) receivers, and less than 3 meters for Wide Area Augmentation System (WAAS) enabled GPS receivers. Typical velocity errors are less than 0.1 knots. The output rate of the navigation solution for most low- to mid-range receivers is 1 Hz, while more expensive receivers can have update rates of 10 Hz or more. Low-end receivers may have latency in the navigation solution of up to 1 second.

Inertial Navigation Systems can output position, velocity, attitude, and angular velocity at high output rates (10-100+ Hz). INS systems numerically integrate data from accelerometers and rate gyros to generate the navigation solution. Errors are introduced through sensor
alignment errors, signal noise, and sensor drift (bias and scale factor). Without correction, these errors will grow unbounded over time. “Loosely coupled” integrated GPS/INS systems use the GPS navigation solution as a reference to estimate and eliminate the error. With an integrated GPS/INS system you get the benefit of the high output rate of the INS and the drift-free characteristics of GPS. INS systems tend to be expensive and introduce operational complexity by requiring a complicated startup initialization process [23].

Digital compasses provide the heading orientation with respect to magnetic or true North or both. A single-axis compass must account for errors introduced by non-zero roll and pitch angles. Without this correction, the sensor error can be much greater than 30 degrees! Tilt compensated compasses appear to share this problem because they correct for the apparent vertical rather than the actual vertical. A three-axis digital compass eliminates this problem and, after calibration, can provide a fairly accurate estimate of the roll and pitch angles. Update rates are typically on the order of a few Hz.

Air data systems directly measure all or part of the airspeed vector. Systems typically include pitot-static tubes and angle-of-attack and sideslip angle wind vanes. These systems are typically heavy, difficult to calibrate, and expensive. They require special placement ahead of the body to get accurate measurements.

Ground-sensing altimeters such as radar, laser, or ultrasonic altimeters are helpful when the exact altitude of the target is not known or when the accuracy of the GPS altitude is not sufficient. These systems measure the relative altitude with respect to the ground. These are especially important when an altitude specific pre-landing maneuver (such as a landing flare or final turn into the wind) needs to be completed.
Sensor Selection

The navigation sensor selection is tightly coupled with the control and guidance system design. The quantities measured or computed by the navigation system determine the variables that can be controlled and the effectiveness and performance of the control. The sensors for the design proposed in this thesis include a low-end WAAS-enabled GPS receiver and a three-axis digital compass. Thus the information directly measured by the sensors are the inertial position and velocity and the vehicle heading. With some assumptions about the airspeed (discussed in the next section), the wind vector can be estimated. If the target position is known in three axes, the state of the parafoil/payload system is fully determined with this data.

The GPS receiver will be modeled after the Garmin GPS 15L. The update rate of the GPS receiver is 1 Hz and the signal latency is assumed to be 1 second. With appropriate inertial sensors the GPS data could be interpolated in-between updates at a much higher rate and reduce or eliminate the effect of the signal latency. This adds unnecessary complexity to the system and it will be shown in the next chapter that this is not required to get sufficient control performance. The errors in position data are small and will not be modeled in the simulation. The errors will not change the guidance error signal significantly unless the system is very close to a waypoint. The error in the landing solution can be assumed to be within the error bounds for the GPS position solution. The velocity error will be assumed to be ±0.1 knots which translates to 0.17 ft/s. The output of the GPS receiver is ±0.33 ft/s, which is more restrictive than the measurement error. The digital compass will be modeled after the PNICorp MicroMag3. The update rate is 2.5 Hz and the error is ±0.3 degrees.
On-Board Wind Estimation

If the magnitude of the horizontal airspeed and sink rate are known or approximately known, knowledge of the vehicle heading along with the inertial velocity (provided by GPS, INS, or other means) can be used to estimate the wind velocity vector. If the payload shape and weight will not change from mission to mission, the horizontal airspeed can be determined through extensive flight testing and stored on-board. If the payload will change from mission to mission, the airspeed can be estimated by flying a few circles at the beginning of the flight and averaging the inertial velocity. From this point on, it is assumed that the horizontal airspeed is known or approximately known.

The wind vector can be determined by the subtraction of the horizontal airspeed from the inertial velocity (see Figure 3.1).

\[
\mathbf{v}_{\text{wind}} = \mathbf{v}_{\text{GPS}} - \mathbf{v}_{\text{airspeed}}
\]  

(3.1)

Figure 3.1: Horizontal geometry used in the wind estimation algorithm.

The airspeed vector can be found from the heading angle and assumed airspeed magnitude \( v_a \) as

\[
\mathbf{v}_{\text{airspeed}} = \begin{bmatrix} v_a \cos \psi \\ v_a \sin \psi \end{bmatrix}
\]

(3.2)
Without filtering, this method of wind estimation is most susceptible to errors during an extended turn. This method assumes no sideslip angle is present which, during an extended turn in windy conditions, will typically not be the case. Also, the glide speed increases over the non-turning glide speed.

A test of this wind estimation algorithm will be conducted through simulation. A wind profile that has significant variation in the magnitude and direction of the wind vector can be given by

\[ w_x = 10 \sin\left( (0.14 h + 19.5) \pi / 180 \right) \]  
\[ w_y = 6.67 \sin\left( (0.12 h + 90) \pi / 180 \right) \]

where \( w_x \) and \( w_y \) are the x and y components of the wind vector, respectively, and \( h \) is the altitude. This profile comes from [2] and was used because it presents great variations in magnitude and direction, not because it matched any realistic wind profile.

The nonlinear 6 degree-of-freedom model developed in Chapter 2 was used to generate truth data from which modeled sensor data could be extracted. The simulated sensor data was used to evaluate the performance of the wind estimation algorithm. The nonlinear parafoil model was initialized with the trimmed state values corresponding to a straight-line equilibrium glide at an altitude of 5000 ft. The simulated flight consisted of three phases, each with a specified constant control input. The flow of data for this simulation is shown in Figure 3.2. The wind model was supplied by Equations (3.3a) and (3.3b). The resulting trajectory consisted of two extended turns separated by a low sideslip arc. The ground track of the trajectory as well as the local wind direction along the trajectory can be seen in Figure 3.3.
Figure 3.2: Data flow for the nonlinear 6 DOF simulation to generate truth data for the modeled sensor data.

\[ \delta_a = \begin{cases} 
-0.2 & 3500 \text{ ft} < h \leq 5000 \text{ ft} \\
0.0 & 1500 \text{ ft} < h \leq 3500 \text{ ft} \\
0.2 & 0 \text{ ft} < h \leq 1500 \text{ ft} 
\end{cases} \]

Figure 3.3: Ground track of the trajectory followed by the parafoil/payload system.

The sensor models were chosen to accurately reflect the signals generated by the GPS receiver and digital compass mentioned earlier. A zero-mean error was added to the truth signal provided by the nonlinear simulation. The sensor sample rate was modeled by taking discrete samples of the truth data and placing a zero-order hold on the sampled value for the duration of the sample period. The computation lag was modeled by placing a delay on the signal. Finally, resolution limits were enforced by rounding the signal to the least significant
bit. The flow of data for generating the modeled sensor signal can be seen in Figure 3.4. The parameters used in the sensor models can be seen in Table 3.1.

![Figure 3.4: Data flow for generating the modeled sensor data.](image)

<table>
<thead>
<tr>
<th>Parameter</th>
<th>GPS Position</th>
<th>GPS Velocity</th>
<th>Digital Compass</th>
</tr>
</thead>
<tbody>
<tr>
<td>Sample Period</td>
<td>1 Hz</td>
<td>1 Hz</td>
<td>2.5 Hz</td>
</tr>
<tr>
<td>Signal Latency</td>
<td>1.0 s</td>
<td>1.0 s</td>
<td>0.0 s</td>
</tr>
<tr>
<td>Zero-Mean Error</td>
<td>0.0 ft</td>
<td>0.0 ft/s</td>
<td>0.3 deg (2σ)</td>
</tr>
<tr>
<td>Resolution</td>
<td>0.0 ft</td>
<td>0.33 ft/s</td>
<td>0.5 deg</td>
</tr>
</tbody>
</table>

Table 3.1: Parameters used for modeling the sensors.

The modeled sensor data was used to estimate the wind profile used in the nonlinear simulation. The GPS velocity data was used for $v_{gps}$ in Equation (3.1). An assumed value for the vehicle airspeed $v_a$ was used in Equation (3.2). This assumed value was held constant throughout the flight, resulting in an average error of 2 ft/s with the actual vehicle airspeed. The airspeed vector $v_{airspeed}$ in Equation (3.1) was generated with the digital
compass signal and the assumed airspeed using Equation (3.2). The flow of data for the wind estimation can be seen in Figure 3.5.

![Diagram](image)

**Figure 3.5:** Data flow for the wind estimation algorithm.

The wind vector estimate was performed once every two seconds so the compass and GPS measurements were synchronized. The actual and estimated wind velocity and heading can be seen in Figures 3.6 and 3.7. The estimation errors for both the velocity magnitude and heading are shown in Figure 3.8.

![Graph](image)

**Figure 3.6:** Estimated and actual wind velocity using modeled sensor data.
Figure 3.7: Estimated and actual wind heading using modeled sensor data. The discontinuities are due to the wind heading being resolved to ±180 degrees.

Figure 3.8: Heading and velocity estimation error magnitudes using modeled sensor data.

As predicted, the accuracy of the estimation is best under non-turning flight for estimations of both velocity and heading. The small errors during the non-turning flight
between 1500 and 3500 ft are due to a combination of a non-zero sideslip angle, error in the airspeed estimate, and sensor resolution effects. The oscillations in the error magnitudes are come from turning into and then away from the wind. The velocity estimation error for this test case is less than 0.5 ft/s during non-turning flight and less than 2 ft/s during turning flight. The heading error is less than 7 degrees for non-turning flight and typically less than 18 degrees for turning flight. In this simulation, there is one instance where the estimation error spikes to 100 degrees and does not re-converge for 12 estimation cycles (24 seconds). The reason for the divergence is not known, but the divergence does not show up in more realistic flight profiles with limited periods of turning.

The wind profile and flight profile used in this test case were chosen to test a “worst case scenario” for the estimation algorithm. Other flight profiles and wind profiles were tested, and this test case did present the largest errors. The divergence problem was not observed in any of the other cases. If only the non-turning portions of the flights are considered, the error magnitudes in both heading and velocity are approximately the same as for the simulation test shown here. Thus it is a reasonable assumption that if the estimations are only made when the craft is not turning, the velocity estimation can be assumed accurate within 3 ft/s and the heading estimation accurate within 10 degrees.
4. CONTROL

Many control schemes have been proposed and/or implemented on autonomous guided parafoil systems. These range from linear and nonlinear model predictive control [24], fuzzy-logic [6], linear output-feedback regulator [7], and adaptive-neural network control [25]. The controlled variables typically include the heading angle, the course angle, or the heading rate (see Figure 4.1). Each of these variables has its own advantages and disadvantages to the overall GNC performance.

![Diagram showing the relationship between the heading angle $\Psi$ and the course angle $\Psi_v$.](image)

**Figure 4.1: The relationship between the heading angle $\Psi$ and the course angle $\Psi_v$.**

Heading rate (turn rate) control offers the highest bandwidth (quickest response) for the closed-loop system. If the measurement is made by an inertial sensor such as a rate gyro, the measurement is very sensitive to movement of the payload with respect to the parafoil, and typically requires filtering. If the measurement is made by discrete differentiation of the heading as measured by a compass, the bandwidth is reduced, and some degree of error is brought into the control signal. This ‘averaged’ heading rate is susceptible also to computation delay. To have stable closed-loop performance requires small gains, and thus yields a low bandwidth.
Heading angle control offers the second highest bandwidth. The heading angle is typically available at lower sample rates than heading rate data from an inertial sensor, but at higher sample rates than course angle data. Thus heading angle control offers moderate response time without the added complexity of filtering. However, if heading control is to be the sole control method used, the guidance computer must accurately know the wind speed and direction as well as the vehicle speed and direction. Without this information, the vehicle may not travel the desired course.

Course angle control yields the lowest system bandwidth. The course angle is most easily obtained from GPS data, which often comes at 1 Hz for inexpensive receivers. On the other hand, no additional data on the wind or vehicle velocity is required to track a desired trajectory.

Both course angle and heading angle control systems will be developed. The course control system will be the primary system used. The heading angle control will be used when the wind speed is known to be greater than the vehicle airspeed, which is the case when the difference between the heading and course angles is greater than 90 degrees.

As will be shown, the computation lag and sample time of the navigation data dominate the closed-loop system performance, significantly slowing down the response speed. The open-loop system is stable throughout the entire flight envelope, and only two control surfaces are available. The combination of stable system dynamics, slow closed-loop performance, and minimal actuators implies that the demand placed on the control system will be minimal and a simple control scheme should be sufficient. Linear proportional or proportional-integral (PI) controllers will be considered.
Model Linearization

Many control analysis and design tools require a linear model of the plant. It is especially instructive to see the effects of the zero-order hold on the control signal and the computation lag of the navigation sensors. The nonlinear dynamic model described in Chapter 2 using the linear aerodynamic model described in Appendix 1 will be numerically trimmed and linearized. The linearization procedure will follow that given in [26].

Using the traditional notation of \( x \) and \( u \) as the state and control vectors, respectively, and \( y \) as the output vector, consider the autonomous nonlinear state-space model

\[
\dot{x} = f(x, u) \quad \text{(4.1a)}
\]
\[
y = g(x, u) \quad \text{(4.1b)}
\]

and define the perturbation variables

\[
\delta x = x - x_e \quad \text{(4.2a)}
\]
\[
\delta u = u - u_e \quad \text{(4.2b)}
\]
\[
\delta y = y - y_e \quad \text{(4.2c)}
\]

where \((x_e, y_e, u_e)\) represent an equilibrium or trim solution such that

\[
\dot{x}_e = f(x_e, u_e) = 0 \quad \text{(4.3a)}
\]
\[
y_e = g(x_e, u_e) \quad \text{(4.3b)}
\]

Expand \( f \) and \( g \) about this point in multivariable Taylor series:
\[ \dot{x}_e + \delta \dot{x} = f(x_e, u_e) + \left( \frac{\partial f}{\partial x} \right)_e \delta x + \left( \frac{\partial f}{\partial u} \right)_e \delta u + \ldots \] 

\[ y_e + \delta y = g(x_e, u_e) + \left( \frac{\partial g}{\partial x} \right)_e \delta x + \left( \frac{\partial g}{\partial u} \right)_e \delta u + \ldots \]

Then the approximate linear state-space model becomes

\[ \delta \dot{x} = \left( \frac{\partial f}{\partial x} \right)_e \delta x + \left( \frac{\partial f}{\partial u} \right)_e \delta u \]  

\[ \delta y = \left( \frac{\partial g}{\partial x} \right)_e \delta x + \left( \frac{\partial g}{\partial u} \right)_e \delta u \]

which is of the standard state-space form:

\[ \dot{x} = Ax + Bu \]

\[ y = Cx + Du \]

The partial derivatives in the Jacobian matrices are traditionally solved analytically and evaluated at the trim point. However, the parafoil dynamic equations do not readily lend themselves to analytic derivatives due to the matrix inversions. The partial derivatives will be approximated with a second order finite-difference equation. For a scalar function, the finite-difference equation is

\[ \frac{df}{dx}(x_e) = \frac{f^+ - f^-}{2h} \]

where \( f^+ = f(x_e + h) \) and \( f^- = f(x_e - h) \). For the multi-variable case, the \( i^{th} \) column of \( \frac{\partial f}{\partial x} \) can be found by perturbing the elements of \( x \) about \( x_e \) as
\[
\frac{1}{2h}\left[f(x_{i}, \ldots, x_{i+h}, x_{i-1}, \ldots, x_{i}, u_{i}, \ldots, u_{a}) - f(x_{i}, \ldots, x_{i-h}, x_{i-1}, \ldots, x_{i}, u_{i}, \ldots, u_{a})\right]
\]

(4.8)

The perturbation \( h \) must be chosen small enough such that the truncated Taylor series is valid, but not so small that the rounding errors due to machine precision become an issue. In general, the value of the perturbation can be different for each element of the state and control vectors.

Let the state and control vectors be

\[
x = [u \ w \ q \ \theta \ v \ p \ \phi \ \psi]^T
\]

(4.9)

\[
u = [\delta_a \ \delta_c]^T
\]

(4.10)

The body-frame velocity components \( u, v, \) and \( w \) are given in ft/s. The components of the angular velocity of the body-frame with respect to the inertial frame expressed in body-frame coordinates, \( p, q, \) and \( r, \) are given in rad/s. The Euler roll, pitch and yaw angles \( \phi, \ \theta, \) and \( \psi \) are given in rad. The asymmetric and symmetric control deflections \( \delta_a \) and \( \delta_c \) are given in rad. The perturbation for velocities, angles, and angular velocities were chosen to be 0.1 ft/s, 0.01 rad and 0.01 rad/s, respectively.

The nonlinear dynamic model described in Chapter 2 using the aerodynamic model described in Appendix 1 was trimmed for a straight-line, steady, non-sideslipping equilibrium glide at an altitude of 5000 ft. The trimmed state values satisfying Equation 4.3a are

\[
x_c = [25.81 \ 7.37 \ 0 \ -0.0835 \ 0 \ 0 \ 0 \ 0 \ 0]^T
\]

(4.11)
The Jacobian matrices for this trim condition are

\[
A = \begin{bmatrix}
-0.4096 & 0.7055 & -7.3703 & -32.0872 & 0 & 0 & 0 & 0 & 0 \\
-1.3674 & -3.9183 & 25.8124 & 2.6862 & 0 & 0 & 0 & 0 & 0 \\
3.6412 & 12.7526 & -55.0265 & 0 & 0 & 0 & 0 & 0 & 0 \\
0 & 0 & 1.0000 & 0 & 0 & 0 & 0 & 0 & 0 \\
0 & 0 & 0 & 0 & -0.5773 & 7.3703 & -25.8363 & 32.0872 & 0 \\
0 & 0 & 0 & 0 & -2.2030 & 13.8032 & 1.0378 & 0 & 0 \\
0 & 0 & 0 & 0 & 1.8681 & -3.1731 & -8.5429 & 0 & 0 \\
0 & 0 & 0 & 0 & 0 & 1.0000 & -0.0837 & 0 & 0 \\
0 & 0 & 0 & 0 & 0 & 0 & 1.0035 & 0 & 0 \\
\end{bmatrix}
\]

\[
(4.12a)
\]

\[
B = \begin{bmatrix}
-2.8370 & -5.6418 \\
-7.7823 & -15.4631 \\
116.4907 & 11.8075 \\
0 & 0 \\
3.9009 & 0 \\
-4.6804 & 0 \\
27.0824 & 0 \\
0 & 0 \\
0 & 0 \\
\end{bmatrix}
\]

\[
(4.12b)
\]

The dynamics decouple into the traditional longitudinal and lateral dynamics. The damping and natural frequency of the characteristic modes are given in Table 4.1. Note that the short period is degenerate, with both poles on the real axis. The natural frequencies of these poles are quite high. All poles are stable except, of course, for the pole corresponding to the heading angle which is at the origin. One final note is the effect of the lateral control variable on the longitudinal dynamics as can be seen in the first column of the B matrix. The asymmetric control deflection creates forces and moments about all axes, thus influencing all of the dynamic states directly.
<table>
<thead>
<tr>
<th>Mode</th>
<th>Damping</th>
<th>Frequency (rad/s)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Short Period</td>
<td>1.000</td>
<td>11.6</td>
</tr>
<tr>
<td></td>
<td></td>
<td>46.9</td>
</tr>
<tr>
<td>Phugoid</td>
<td>0.325</td>
<td>1.37</td>
</tr>
<tr>
<td>Dutch Roll</td>
<td>0.479</td>
<td>8.37</td>
</tr>
<tr>
<td>Roll</td>
<td>1.000</td>
<td>14.3</td>
</tr>
<tr>
<td>Spiral</td>
<td>1.000</td>
<td>0.629</td>
</tr>
</tbody>
</table>

Table 4.1: Damping and natural frequency of the open-loop linear system model.

Course Angle Dynamics

The course angle is defined by

\[
\psi_V = \tan^{-1} \frac{\dot{p}_E}{\dot{p}_N}
\]  

(4.13)

and the time rate of change of the course angle is given by

\[
\ddot{\psi}_V = \frac{1}{1 + \dot{p}_E / \dot{p}_N \left( \frac{\dot{p}_N \dot{p}_E - \dot{p}_E \dot{p}_N}{\dot{p}_N^2} \right)}
\]  

(4.14)

This equation is highly nonlinear due to the multiple products of trigonometric functions of the Euler orientation angles. Details of the dynamics are lost in the linearization. Therefore, the course angle control design will be performed on the nonlinear model after an understanding of the effects of the navigation data sample period and delay have on closed loop control of the heading angle.
**Transfer Function Models**

Transfer function models of the plant can be extracted from the linear state-space models by taking the Laplace transform of Equation 4.6a:

\[ G(s) = C(sI - A)^{-1} B \]  \hspace{1cm} (4.15)

where \( I \) is the identity matrix and \( G(s) = x(s)/u(s) \). Using \( A \) and \( B \) as given in Eqs. 4.12 and selecting \( C \) to select \( \psi \) as the output, the transfer function that relates the heading angle to asymmetric control input is

\[
\frac{\psi(s)}{\delta_\alpha(s)} = \frac{27.18s^3 + 413s^2 + 730.1s + 1640}{s\left(s^4 + 22.92s^3 + 198.6s^2 + 1117s + 629.7\right)}
\]  \hspace{1cm} (4.16)

The computation delay or signal latency \( \Delta \) has a transfer function

\[ G_{\text{comp}}(s) = e^{-s\Delta} \]  \hspace{1cm} (4.17)

This transfer function has a magnitude of one and a phase of \(-\omega\Delta\) radians. The combined transfer function of the computation delay and the plant is not rational, and therefore difficult to work with. Therefore a Padé approximant, which matches the first few terms in a Taylor series expansion, will be used. The Padé approximant accurate to five Taylor series terms is given by:

\[
G_{\text{comp}}(s) \approx \frac{1 - s\Delta/2 + (s\Delta)^2/12}{1 + s\Delta/2 + (s\Delta)^2/12}
\]  \hspace{1cm} (4.18)

In essence, the computation delay acts as a filter in the forward loop and contributes a pair of unstable zeros.
The sampler plus zero-order hold on the control signal has a transfer function of

\[ G_{0o}(s) = \frac{1 - e^{-sT}}{sT} \]  \hspace{1cm} (4.19)

where \( T \) is the sample period. Using a Padé approximant of \( e^{sT} \), the transfer function of the zero-order hold accurate to four terms of the Taylor series is

\[ G_{0o}(s) \approx \frac{1 - sT/10 + (sT)^2/60}{1 + 2sT/5 + (sT)^2/20} \]  \hspace{1cm} (4.20)

Thus, like the computation delay, the zero-order hold acts as a filter in the forward loop with a pair of unstable zeros.

Finally, the nonlinear actuator model will be approximated with a transfer function of

\[ \frac{\delta_a(s)}{u(s)} = \frac{36}{s^2 + 9.6s + 36} \]  \hspace{1cm} (4.21)

**Heading Angle Control Design**

The plant transfer function model is type one, i.e. an \( s \) can be factored out of the denominator. Thus we expect to see zero steady-state error with proportional control for constant input. The design will be stable and have zero steady-state error if the pole at the origin is pulled into the left half plane by adjusting the forward loop gain, and not towards the unstable zeros contributed by the signal latency and zero-order hold. The root locus zoomed in to the origin for the closed-loop transfer function with a hold and latency of one second is shown in Figure 4.2. The pole at the origin is indeed pulled into the left half plane for small gains. Proportional control, then, should work for this system. PI control, on the
other hand, will not result in a stable system. The additional pole at the origin results in both poles heading immediately into the right half plane as seen in Figure 4.3.

Figure 4.2: Root locus plot for the open loop transfer function of a proportional heading control design.

Figure 4.3: Root locus plot for the open loop transfer function of a proportional plus integral control design.
The proportional gain is chosen as a compromise between time response and the phase margin. The final choice of gain was 0.08 which yielded a phase margin of 52.7 degrees and a gain margin of 10.6 dB. The step responses of the linear and nonlinear models are shown in Figure 4.4. The differences arise from two main factors. The first is that the approximations for the computation delay and the zero-order hold are valid for small delays and sample times. With a sample time and a delay of one second, many terms from the Taylor series expansion need to be retained to yield an accurate approximation. The higher-order approximations given in [17] all had the same trend: more stable poles and zeros were added as the order increased. In all approximations, only one pair of unstable zeros was contributed. The second factor is that the linearization underestimates the turn response. Recall from Chapter 2 that the heading rate depended on the pitch rate and the yaw rate. The linearization of the heading rate depends only on the yaw rate. As it turns out, the choice of gain for the linear model still offers the best compromise of rise time and overshoot for the nonlinear model.

![Step Response](image)

**Figure 4.4**: Comparison of the response to a step input in the heading angle command between the linear and nonlinear model.
Course Heading Control Design

Looking at equation (4.14) and noting that the derivative of a dynamic variable (i.e. $u, v, w, p, q, r$) is not present anywhere, we can assume that the transfer function relating course heading to asymmetric control input will also be of type one. Thus, as was the case with heading control, for constant input we can expect zero steady-state error from a proportional control scheme, and an unstable system from a PI control scheme. The gain for the proportional control will be found by trial-and-error, and will be selected based on the time-domain performance parameters of peak overshoot and rise time, with the emphasis on rise time. The step response is plotted for gains from 0.06-.09 in increments of 0.01 in Figure 4.5.

![Step Response](image)

**Figure 4.5:** Comparison of the response to a step input in the command of the course angle for various choices of the control gain.

The gain was chosen to be 0.07. This offered a compromise between rise time, overshoot, and settling time. A comparison of the actual and measured course angle can be seen in Figure 4.6. Note the error contributions due to the signal latency and the quantization.
Figure 4.6: Comparison of the actual and measured response to a step input in the course angle command for the design gain of 0.07.
5. GUIDANCE

The guidance task is to generate a trajectory that will take the system to the desired landing point, and to generate steering commands to track that trajectory. There are many challenges in the guidance of a lightly loaded parafoil/payload system. The system is unpowered, thus only one attempt at landing is possible. The vehicle airspeed is on the order of the winds that can be expected in the lower atmosphere, thus the wind strongly influences the motion of the system. The guidance system must overcome the drift induced by the wind, and do so without a priori information on the wind profile or with an estimation of the wind profile that may be subject to large errors.

Many guidance concepts have been implemented or proposed for autonomous parafoil systems. In the early 1970’s, the Army proposed various radial homing guidance laws [2]. These guidance laws were created so the parafoil system would home in on a radio beacon placed at the desired landing site.

A concept from the Naval Postgraduate School applies an optimal control synthesis to rapidly generate time-optimal trajectories for a known wind profile [27]. The trajectory consisted of three phases: a straight-line glide, a spiral descent, and a final straight-line glide. The location of the spiral descent could be arbitrarily placed near the beginning or the end of the trajectory. The final straight-line segment begins when just enough altitude (potential energy) is left to reach the landing site.

The Atair Aerospace Onyx system uses an adaptive guidance law [25]. Right after deployment, the system turns towards the target and estimates the glide angle relative to the ground and the velocity relative to the target. From this data, a desired glide-slope is generated. The system then enters a rapid spiral descent until it is just above the desired glide-slope. The system then turns towards the target and performs weaving maneuvers to
maintain the glide-slope. In this way, the variation of the wind is accounted for. Once at a specified range from the target, the system enters a terminal spiral.

The DLR German Aerospace Center has developed a trajectory planning approach called the ‘T-Approach’ [28]. The trajectory is continuously updated until landing, accounting for uncertainties in the wind and system parameters. Instead of using circular spiral descents for energy management, the waypoints are distributed in a T-shaped pattern with the ‘crossbar’ of the T perpendicular to the wind direction and the bottom of the ‘post’ pointing into the wind at the target. The initial homing phase terminates at the junction of the ‘post’ and ‘crossbar’ portions of the T. The next phase consists of a racetrack pattern running up and down the ‘crossbar’ of the T. When a certain decision altitude is reached, the system turns towards the bottom of the ‘post’. Once the system reaches this point, it turns directly into the wind and maintains that heading until landing. The decision altitude is chosen such that the landing point coincides with the target. The waypoints that make up the T are not fixed with respect to the target; rather they are fixed with respect to the wind.

While the guidance algorithms described above are quite varied, there are some common similarities. With the exception of the radial homing technique, each algorithm requires some knowledge of the wind. The information may either be provided before the mission, estimated in real-time, or some combination of the two. Also, three phases seem to be present in most of these guidance algorithms. The first is a homing phase that gets the parafoil system to or near the target. The second is an energy management phase that efficiently dissipates the excess altitude at the end of the homing phase. The final phase involves deciding when to exit the second phase and again home in on the target until landing. Most of these systems also have a constraint that the final heading must be pointed into the wind.
Guidance Algorithm Development

The guidance algorithm developed here will be done under two critical assumptions: the wind speed does not exceed the vehicle airspeed, and the system was deployed within the feasible drop zone. Under these assumptions, the target is reachable regardless of the amount of variation in both magnitude and heading found in the wind profile, and the course angle can be arbitrarily chosen. The guidance algorithm will make minimal assumptions about the performance of the system and require minimal computational power. The guidance algorithm will require no assumptions or a priori information about the wind profile. Although no constraint will be imposed on the final heading, the architecture of the guidance algorithm will allow for this to be incorporated at a later time. The only goal of the guidance system in this development is to land as close to the target as possible.

The trajectory generated by the guidance algorithm consists of a series of waypoints fixed in inertial space. These waypoints will correspond to the three phases described in the previous section: homing, energy management, and final approach. The first waypoint will be located at the desired target. This corresponds to the initial homing phase. The second and third waypoints will be located an equal distance upwind and downwind of the target. A racetrack pattern will be flown between these waypoints until a specified altitude is reached. This corresponds to the energy management phase. At the completion of each circuit of the ‘racetrack’, provision will be made to realign the waypoints with the wind based on the wind heading estimated by the navigation algorithm. The final waypoint will be located at the desired target. When the system reaches a certain altitude, the system immediately exits the racetrack pattern and homes in on this final waypoint. An example of a trajectory generated by this algorithm can be seen in Figure 5.1
The guidance command is a course heading equal to the heading of the line-of-sight between the current position of the parafoil and the current waypoint (see Figure 5.2). The guidance law is:

\[
\psi_{v_{COH}} = \psi_{LOS} = \tan^{-1}\left(\frac{y_{wp} - y_{par}}{x_{wp} - x_{par}}\right)
\]

(5.1)

where \((x_{wp}, y_{wp})\) and \((x_{par}, y_{par})\) are the coordinates of the waypoint and parafoil, respectively. The position data used for this computation is supplied by the GPS position solution, and is thus subject to a signal latency of one second. The course angle command is resolved to be within \(\pm 180\) degrees. Care is taken in computing the arctangent to eliminate quadrant ambiguity.
This direct course guidance was chosen over a proportion-navigation scheme because it minimizes the distance traveled and maximizes the altitude reserve at each waypoint. This also keeps the feasible drop zone discussed in Chapter 2 circular. With a proportion navigation scheme, the drop zone becomes elliptical, with the semi-major axis aligned with the mean wind direction [2]. The racetrack pattern for the energy management phase was chosen over the circular spiral descent to minimize the control effort and simplify the trajectory generation. Maintaining the circular flight path in the presence of varying winds seemed overly complicated when the same results could be achieved with much less computation and control effort.

The origin of the inertial coordinate frame is located at the position of the target. A waypoint is considered achieved and the next waypoint made active when the range to the waypoint is less than 75 ft. The range is calculated by:

\[
R = \left[ (x_{wpt} - x_{par})^2 + (y_{wpt} - y_{par})^2 \right]^{1/2}
\]  

(5.2)
The racetrack waypoints for the energy management phase are selected to be 500 ft upwind and downwind of the target landing site. This gives approximately 1000 ft of altitude decrease per circuit of the racetrack under calm conditions. The waypoints are determined at the entrance to the energy management phase and held constant until the final approach phase. The location of the waypoints is based on the estimated wind direction at the entrance to this phase. If there is no appreciable wind, the default is due north and south of the landing site. The waypoints are calculated as

\[ P_{N_{opt}} = \pm 500 \cos \Psi_{wind} \]  \hspace{1cm} (5.3a)
\[ P_{E_{opt}} = \pm 500 \sin \Psi_{wind} \]  \hspace{1cm} (5.3b)

Finally, the energy management phase is exited at the altitude where the system would have just enough time to get to the landing site if it was located at the downwind waypoint given the estimated airspeed of the system, the estimated wind speed, and the vertical velocity as measured by GPS. Note that the magnitude of the wind vector is used in this computation, not the component along the flight path. This selection of the “decision altitude” should allow the system to land very near the desired landing site under most conditions. This approach assumes the “worst-case” scenario regardless of the actual conditions, i.e. the system must fight upwind from the point furthest from the target in the racetrack pattern. In general, the system will not be at the downwind waypoint in the racetrack pattern, thus the system should be able to make it to the target with altitude to spare, even if the wind speed increases as the system approaches the ground. This approach mirrors the Min-Max approach in that it may not be the “best” solution, but it is robust to the most varied conditions [29].

If the airspeed is \( V_{AS} \) and the wind speed is \( V_W \), then the time required to travel the 500 ft from the downwind waypoint is given by:
Likewise, for a given vertical descent rate $V_h$, the time to descend from a given altitude $h$ is:

$$t = \frac{h}{V_h}$$  \hspace{1cm} (5.5)

By equating these two times, the decision altitude is given by:

$$h = \frac{500V_h}{(V_{AS} - V_w)}$$  \hspace{1cm} (5.6)
6. INTEGRATED GNC SIMULATIONS

A series of simulations were performed to evaluate the performance of the combined guidance, navigation and control algorithms. The nonlinear 6 degree-of-freedom parafoil model developed in Chapter 2 was combined with the wind estimation algorithm and sensor models developed in Chapter 3, the control algorithm developed in Chapter 4, and the guidance algorithm developed in Chapter 5. The parameters used in the sensor models for all of the simulations discussed in this chapter are given in Table 3.1. The flow of data of the combined simulation is shown in Figure 6.1.

Figure 6.1: Data flow for integrated guidance, navigation, and control simulation.

Wind Profiles

The following wind profiles will be used to evaluate the performance of the combined guidance, navigation, and control system.
Profile 1:

\[ V_w = 10 \]
\[ \psi_w = \sin \left( \frac{h}{5000} \pi \right) \]  

Profile 2:

\[ V_w = 20 \sin \left( \frac{h}{5000} \pi \right) \]  
\[ \psi_w = 0 \]  

Profile 3:

\[ V_w = 20 \]
\[ \psi_w = 0 \]  

Profile 4:

\[ V_w = 20 \cos \left( \frac{h}{5000} \pi \right) \]  
\[ \psi_w = \sin \left( \frac{h}{5000} \pi - \frac{\pi}{4} \right) \]

The velocity and heading of profiles 1-4 are plotted against altitude in Figures 6.2 and 6.3 respectively.
Profile 1 has a moderate constant velocity. The heading varies approximately 60 degrees from 5000 ft to the ground. This singles out the performance of the energy management waypoint selection mechanism. It should identify any sensitivity to the alignment of the waypoints with the wind. Profile 2 has a constant heading and the velocity varies by 20 ft/s
from 5000 ft to the ground, with the velocity equal to zero at the ground. This singles out the performance of the exit criteria for the energy management phase in the presence of wind variation. This profile will cause the system to leave the energy management phase early. Profile 3 has a constant high velocity and a constant heading. This singles out the performance of the energy management phase exit criteria when the wind velocity is near the system airspeed. Profile 4 has moderate variations in both heading and velocity, with the velocity magnitude highest at the ground. This evaluates the overall performance of the GNC algorithm.

Simulation study

Multiple simulations were run for each wind profile described in the previous section. A set of 25 starting points were generated, varying both altitude and position relative to the target. These 25 simulation cases were repeated for each wind profile described in the previous section and also for a calm wind condition. The stop condition for each simulation was when the parafoil reached the ground (h=0). The parafoil and the combined GNC algorithm were simulated in the Matlab/Simulink environment. Figures 6.4 and 6.5 show the ground track and 3-D profile of a typical trajectory generated by the guidance algorithm. The initial homing and energy management phases can be clearly seen. The final approach phase can be seen as the final loop over the target in Figure 6.4 and the hooked end of the trajectory in Figure 6.5. The radius of the final loop is determined by the maximum actuator deflection. Recall that we defined the origin of the inertial coordinate frame to coincide with the target position, thus the target coordinates are (0,0,0).
Figure 6.4: Ground track of a trajectory generated by the guidance algorithm in calm conditions.

Figure 6.5: 3-D profile of a trajectory generated by the guidance algorithm in calm conditions.

Figures 6.6 and 6.7 show a zoomed in view of the last 1400 ft of another simulation run using a calm wind profile. This details the exit of the energy management phase and
transition to the final approach phase. This shows that the system did leave the energy management phase early resulting in a terminal spiral about the target. The radius of the spiral in this case is again determined mainly by the limited maximum actuator deflection.

Figure 6.6: Zoomed in view of the ground track of the energy management and final approach phases for a trajectory generated by the guidance algorithm under calm conditions.
Figure 6.7: Zoomed in view of the 3-D profile of the energy management and final approach phases for a trajectory generated by the guidance algorithm under calm conditions.

The combined results of all simulation runs are shown in Table 6.1. The average miss distance, standard deviation, and circular error probable (CEP) for each wind profile are tabulated. The CEP is defined as the mean plus twice the standard deviation. Thus, the miss distance will be less than the CEP 95% of the time. Polar scatter plots showing the landing sites for each wind profile are shown in Figures 6.8-6.12. In these scatter plots, the bold circle represents the calculated CEP for the respective data set. The results of all simulations for wind profiles 1-4 are shown in Figure 6.13.

It is evident in studying Table 6.1 that the wind profile has little effect on the landing precision. The value of the mean miss distance is nearly constant. It is believed by the author that if more simulations were run for each profile, the variation of the means would further decrease. The trend in the standard deviations is similar.

The landing errors for the cases using wind profile 3 are somewhat larger due to the large velocity of the wind. The wind speed is near the airspeed of the modeled system. The system will typically leave the energy management phase early, resulting in excess altitude
when it reaches the target. The resulting flight path as the system continuously homes in on
the target is similar to that shown in Figure 6.14, but more elongated along the wind vector.
The larger landing error for this case comes from the fact that the system can potentially
touch the ground at any point along the weaving curve, and with the large wind velocity, the
deviations of the curve from the target for this wind profile are larger than those for other
wind profiles. Compare the terminal flight path in Figure 6.14 to that in Figure 6.6. Another
effect of the large wind speed in profile 3, as can be seen in Figure 6.11, is that all landing
sites resulting from this profile are downwind of the target. The same is true for wind profile
4 as shown in Figure 6.12.

<table>
<thead>
<tr>
<th>Wind Profile</th>
<th>Mean (ft)</th>
<th>Std. Dev. (ft)</th>
<th>CEP (ft)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Calm</td>
<td>58.8</td>
<td>17.7</td>
<td>94.2</td>
</tr>
<tr>
<td>Profile 1</td>
<td>54.7</td>
<td>30.2</td>
<td>115.1</td>
</tr>
<tr>
<td>Profile 2</td>
<td>54.0</td>
<td>21.4</td>
<td>96.7</td>
</tr>
<tr>
<td>Profile 3</td>
<td>77.5</td>
<td>28.5</td>
<td>134.6</td>
</tr>
<tr>
<td>Profile 4</td>
<td>57.3</td>
<td>21.2</td>
<td>99.6</td>
</tr>
<tr>
<td>Profiles 1-4</td>
<td>60.9</td>
<td>27.2</td>
<td>115.0</td>
</tr>
</tbody>
</table>

Table 6.1: Statistical analysis of the miss distance for multiple simulation cases.
Figure 6.8: Polar scatter plot of landing sites relative to the target (center) for calm wind conditions.

Figure 6.9: Polar scatter plot of landing sites relative to the target (center) for wind profile 1.
Figure 6.10: Polar scatter plot of the landing sites relative to the target (center) for wind profile 2.

Figure 6.11: Polar scatter plot of the landing sites relative to the target (center) for wind profile 3.
Figure 6.12: Polar scatter plot of the landing sites relative to the target (center) for wind profile 4.

Figure 6.13: Polar scatter plot of the landing sites relative to the target (center) for wind profiles 1-4.
Figure 6.14: Ground track of a trajectory that enters the final approach phase immediately upon reaching the target for wind profile 4.
7. SUMMARY AND CONCLUSIONS

This thesis presents the details of a complete guidance, navigation, and control concept for a small-scale autonomous parafoil. The design requirement was to keep the algorithm as simple as possible, requiring minimal computation effort and maximum flexibility to add additional capabilities. In those respects, the design is a success. The only parameter in the design that is sensitive to payload or canopy changes is the choice of the proportional control gain.

Navigation data is collected from a GPS receiver and a 3-axis digital compass. This sensor suite provides inertial position, inertial velocity, and heading azimuth. The GPS data is updated at 1 Hz and has a signal latency of 1 second. A zero-order hold is placed on the signal in between updates. The digital compass data is updated at 2.5 Hz and the signal latency is considered to be negligible. Errors in the GPS position and velocity solutions were neglected as they were shown to have minimal effect on the system performance. Errors in the digital compass solution were assumed to be ± 0.3 degrees with zero-mean and a Gaussian distribution.

A wind estimation algorithm was presented that used the inertial velocity provided by the GPS solution, the heading from the digital compass, and an assumed airspeed of the system. The estimation was computed at 0.5 Hz so the compass and GPS data were synchronized. It was shown that without additional filtering the accuracy of the solution was within 3 ft/s for velocity and 10 degrees in heading during non-turning flight. This accuracy was achieved in the presence of an average 2 ft/s error in the vehicle airspeed estimate.

Two linear proportional controllers were designed to control the course angle and the heading angle. The control system places a zero-order hold on the discrete sampled data. It was shown that the contributions from signal latency and the zero-order hold were two pairs
of unstable zeros to the open-loop transfer function, as well as slow complex poles close to the origin. The effect of these poles was to limit the range of acceptable gains and significantly reduce the bandwidth of the closed-loop system.

The control design was tested with the aid of a nonlinear 6 degree-of-freedom model. The dynamic model used a set of constant linear aerodynamic derivatives. It was shown that acceptable tracking could be achieved with the linear proportional control design in the presence of significant navigation signal latency and limited sample rate.

A guidance algorithm was developed that generates a trajectory consisting of a series of inertially-fixed waypoints. The trajectory consists of three phases: target homing, energy management, and final approach. The target homing phase takes the parafoil from the initial deployment position to the target. The energy management phase bleeds the excess potential energy of the system while staying close to the target. The energy management phase was chosen to be a racetrack pattern aligned with the wind vector when the system exits the initial homing phase. In this study, the racetrack was not realigned with the wind as successive circuits of the racetrack pattern were completed. Finally, when a specified altitude is reached, the system exits the energy management phase and continuously homes in on the target.

Monte-Carlo simulations were run to test the combined guidance, navigation, and control concept. The system was simulated for a number of wind profiles which were specifically chosen to test certain design parameters. It was shown that the landing accuracy was determined primarily by the maximum turn rate which was limited by the maximum actuator deflection. The landing accuracy was not sensitive to variations in the wind speed or direction, or misalignment of the racetrack pattern of the energy management phase.

The combined results of the simulation tests showed a landing accuracy of 115 ft or better 95% of the time. This accuracy is achieved with no a priori knowledge of the wind profile
and with the use of low-cost navigation sensors. These results show that the GNC concept has great promise.

There are two primary means to increase the landing accuracy. The first would be to use a better method of determining the decision altitude for exiting the energy management phase. This would minimize or eliminate the excess altitude of the system when it reaches the target. Thus the minimum turn rate of the system would not be the determining factor of the landing accuracy. The second would be to use a sensor suite that has a higher sample rate and lower computation lag. This would increase the bandwidth of the system by moving the poles contributed by the zero-order hold and signal latency transfer functions further into the left-half plane. The best results would be obtained if these two improvements were combined.
A baseline GNC concept has been developed, and there are many possible extensions to this design that should be considered. The present concept did not require the system to complete the landing while heading into the wind. There are advantages to requiring this terminal condition. First, it minimizes the ground speed of the system, minimizing the impact load on the payload. Second, it minimizes the lateral velocity of the payload with respect to the ground, and this may be desired to limit the structural requirements of the payload.

The addition of a landing flare should also be considered. This maneuver consists of a large symmetric flap deflection prior to touchdown that minimizes the vertical velocity on impact. This would likely require additional sensors to detect the relative distance between the payload and the ground. The maneuver has to be timed correctly to maximize the effect.

Though it was shown in this study that the landing accuracy was not significantly affected by the alignment of the racetrack pattern in the energy management phase, it would be interesting to add that capability to the system. This may prove to be beneficial as the wind speed approaches the vehicle airspeed.

In this study, the racetrack pattern of the energy management phase was arbitrarily centered over the landing site. It is possible to center the pattern such that the time to travel from each waypoint back to the target is the same. It may also prove beneficial to exit the energy management phase based on the time to return to the target from the current position in the racetrack pattern rather than from the downwind waypoint. This may eliminate the terminal spiral over the target and increase the landing accuracy.

Another extension to the design would be to use an adaptive algorithm to adjust the proportional control gain. This would allow the system to maintain the same step response
profile throughout the entire flight envelope and also would allow the payload to vary from mission to mission without the need to reprogram the flight computer. This would be a significant step towards “drop and forget” operations.

It may be desirable to exclude certain airspace from the trajectory and to be able to detect and avoid obstacles or landing hazards. The airspace may be restricted or have geographical obstructions. It may also be necessary to coordinate multiple airdrop systems to be in the same airspace and midair collisions must be avoided.

A final recommendation would be to develop an efficient guidance algorithm for the case when the wind speed is greater than the vehicle airspeed. Simply pointing the vehicle azimuth into the wind will not, in general, be the best solution. There are two distinct cases to consider. The first is when the wind speed is greater than the vehicle airspeed only for short periods of time. The second is when the wind speed is greater than the vehicle airspeed for significant portions of the flight. The first case requires a heading to be chosen to minimize the effects of the wind. The wind profile itself does not necessarily need to be completely known. The vehicle heading should be chosen such that the horizontal line-of-sight distance decreases if possible, or increases at the slowest possible rate. If the wind speed exceeds the vehicle airspeed for significant portions of the flight, it is believed that more information about the wind profile will be needed.
APPENDIX: AERODYNAMIC DERIVATIVES AND GEOMETRIC CONSTANTS

The stability derivatives come from [15] and are shown in Table A.1. It is important to note that these stability derivatives are for one operating point only. The value given for \( C_{m_a} \) seemed excessively large and was reduced by an order of magnitude. Other constants and the approximate geometry of the RGS parafoil are given in Table A.2.

<table>
<thead>
<tr>
<th>( C_{la} )</th>
<th>2.8756</th>
<th>( C_{ya} )</th>
<th>-0.5435</th>
</tr>
</thead>
<tbody>
<tr>
<td>( C_{lb} )</td>
<td>0.4670</td>
<td>( C_{yb} )</td>
<td>0.1368</td>
</tr>
<tr>
<td>( C_{ld} )</td>
<td>0.2350</td>
<td>( C_{yk} )</td>
<td>-0.0060</td>
</tr>
<tr>
<td>( C_{lmu} )</td>
<td>0.4000</td>
<td>( C_{f^b} )</td>
<td>-0.0796</td>
</tr>
<tr>
<td>( C_{Dh} )</td>
<td>0.0905</td>
<td>( C_{fr} )</td>
<td>-0.1330</td>
</tr>
<tr>
<td>( C_{Dr} )</td>
<td>0.1900</td>
<td>( C_{fg} )</td>
<td>0.0100</td>
</tr>
<tr>
<td>( C_{Du} )</td>
<td>0.0957</td>
<td>( C_{ng} )</td>
<td>0.0287</td>
</tr>
<tr>
<td>( C_{Dmp} )</td>
<td>0.0040</td>
<td>( C_{ng^b} )</td>
<td>0.0155</td>
</tr>
<tr>
<td>( C_{Dmu} )</td>
<td>0.0365</td>
<td>( C_{n_{e}e} )</td>
<td>-0.0130</td>
</tr>
<tr>
<td>( (C_{fj}S)_S )</td>
<td>1.8870</td>
<td>( C_{nf} )</td>
<td>-0.0350</td>
</tr>
<tr>
<td>( C_{m_j} )</td>
<td>0.2499</td>
<td>( C_{m_f} )</td>
<td>-1.8640</td>
</tr>
<tr>
<td>( C_{m_a} )</td>
<td>-0.8985</td>
<td>( C_{m_q} )</td>
<td>0.0294</td>
</tr>
<tr>
<td>( C_{m_{ha}} )</td>
<td>0.0298</td>
<td>( C_{m_{ha}} )</td>
<td>0.0294</td>
</tr>
</tbody>
</table>

Table A.1: Aerodynamic stability derivatives.
<table>
<thead>
<tr>
<th>Parameter</th>
<th>Value</th>
<th>Unit</th>
</tr>
</thead>
<tbody>
<tr>
<td>$S$</td>
<td>30.0</td>
<td>ft</td>
</tr>
<tr>
<td>$b$</td>
<td>7.5</td>
<td>ft</td>
</tr>
<tr>
<td>$c$</td>
<td>4.0</td>
<td>ft</td>
</tr>
<tr>
<td>$a$</td>
<td>0.15</td>
<td>ft</td>
</tr>
<tr>
<td>$t$</td>
<td>0.6</td>
<td></td>
</tr>
<tr>
<td>$AR$</td>
<td>1.875</td>
<td></td>
</tr>
<tr>
<td>$\alpha_0$</td>
<td>-5.1085</td>
<td>deg</td>
</tr>
<tr>
<td>$n$</td>
<td>28</td>
<td></td>
</tr>
<tr>
<td>$R$</td>
<td>5.0</td>
<td>ft</td>
</tr>
<tr>
<td>$\delta_{u_{\text{max}}}$</td>
<td>0.25</td>
<td>rad</td>
</tr>
<tr>
<td>$e$</td>
<td>0.7</td>
<td></td>
</tr>
<tr>
<td>$w$</td>
<td>25</td>
<td>lb</td>
</tr>
<tr>
<td>$g$</td>
<td>32.2</td>
<td>ft/s²</td>
</tr>
<tr>
<td>$m_R$</td>
<td>0.7764</td>
<td>slug</td>
</tr>
<tr>
<td>$J_x$</td>
<td>0.2235</td>
<td>slug/ft⁴</td>
</tr>
<tr>
<td>$J_y$</td>
<td>0.2235</td>
<td>slug/ft⁴</td>
</tr>
<tr>
<td>$J_z$</td>
<td>0.0950</td>
<td>slug/ft⁴</td>
</tr>
<tr>
<td>$\rho_0$</td>
<td>0.00238</td>
<td>slug/ft³</td>
</tr>
<tr>
<td>$h$</td>
<td>0.5</td>
<td>ft</td>
</tr>
</tbody>
</table>

Table A.2: Geometric data and other constants.
1. REFERENCES


