The Guided Samara: Design and Development of a Controllable Single-Bladed Autorotating Vehicle

by

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B.S. Mechanical Engineering, Drexel University, 2005

Submitted to the Department of Aeronautics and Astronautics in partial fulfillment of the requirements for the degree of Master of Science in Aeronautics and Astronautics

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MASSACHUSETTS INSTITUTE OF TECHNOLOGY

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Abstract

Accurate delivery of cargo from air to ground is currently performed using autonomously guided parafoil systems. These parafoils offer limited maneuverability and accuracy, and are often relatively complex systems with significant deployment uncertainty. The author has proposed, designed, and developed a novel precision airdrop system in the form of a samara. Consisting of a single wing, payload, and control system, the guided samara is mechanically simple, and when correctly configured, is globally stable during deployment and steady descent. The proposed control mechanisms grant the vehicle omni-directional glide slope control during autorotating descent. This is the first documented effort to develop an actively controlled vehicle of this form factor.

The research presented in this thesis includes the conceptual design of the vehicle and several control schemes, development of a six degree-of-freedom computer simulation predicting the vehicle’s flight performance, and the design, fabrication, and flight testing of a guided samara prototype. Over the course of the development, numerous free-flights were conducted with unguided samara models. Flight results and simulation results in various configurations yielded the discovery of several mechanisms critical to understanding samara flight.

The free-flight simulation predicted descent and rotation rates within 10% of those observed during flight testing. Simulations of the guided samara predicted stability during control actuation as well as considerable control authority. Using a programmable microprocessor, hobby servo, and 2-axis electronic compass, a control system was developed. The guided samara prototype was flight tested at NASA Langley to attempt to demonstrate omni-directional glide slope control during descent. Due to weather conditions and test circumstances, the demonstration was inconclusive. The simulation results and favorable performance of the developed control system indicate that guidance of the samara during autorotation is feasible, and only awaits favorable conditions for demonstration of this capability.

Thesis Advisor: Mark Drela
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V velocity vector \([v_1, v_2, v_3]^{\top}\)

\(V_m\) Earth’s vertical magnetic field intensity (nanoTesla)

\(V_v\) Air velocity due to vehicle’s descent rate, always negative

\(W\) weight

\(X\) state vector \([\omega \quad \theta \quad V \quad R]^{\top}\)

\(X, Y, Z\) body frame coordinates used for aerodynamic analysis

\(a\) section lift-curve slope

\(b\) number of blades

\(\hat{b}_1, \hat{b}_2, \hat{b}_3\) body frame unit vectors

\(c\) section chord

\(c_f\) flap chord

\(\hat{i}_1, \hat{i}_2, \hat{i}_3\) inertial frame unit vectors

\(f\) external force

\(m\) mass

\(r\) radial distance from root to blade element

\(r_{cp}\) spanwise distance from center of mass to center of pressure

\(r_{cr}\) horizontal distance from center of mass to the point of action of \(C_R\)

\(r_{ct}\) horizontal distance from center of mass to the point of action of \(C_T\)

\(y\) non-dimensional radial distance from root to blade element \((y = r/R_i)\)

\(\beta\) Compass tilt angle from horizontal

\(\Theta\) orientation vector \([\phi \quad \theta \quad \psi]^{\top}\)

\(\alpha\) angle of attack

\(\delta\) flap deflection angle

\(\phi, \theta, \psi\) Euler angles (roll, pitch, and yaw)

\(\rho\) air density

\(u_i\) rotor induced velocity, always positive

\(u_1, u_2, u_3\) velocity in inertial frame

\(\omega\) angular velocity vector \([\omega_1, \omega_2, \omega_3]^{\top}\)

\(\omega_1, \omega_2, \omega_3\) angular velocity

(\(\times\)) cross product operation
Chapter 1

Introduction & Background

The scope of this research includes the design and development of a controllable vehicle closely resembling the naturally occurring samara. It was hypothesized that a vehicle of this form factor could be actively controlled during its autorotative descent. Verification of this hypothesis was pursued through conceptual design, analytical modeling, and experimental demonstration. The end goal was to investigate the feasibility of developing the aforementioned vehicle, and granting it the capability of accurately delivering a payload to a ground target.

1.1 Precision Airdrop

Since its conception in the early 20th century, the airplane has been used as a platform for delivering payloads to the ground. Airdrop as we know it today was developed during World War II to re-supply otherwise inaccessible troops, and was conducted by pushing small crates with attached parachutes out of an aircraft’s side cargo doors. Over the course of the past century, airdrop has progressed to include specialized cargo aircraft with rear access ramps and advanced maneuvers such as low-level extraction, in which tanks and other large payloads are pulled from the rear of an aircraft and delivered from extremely low altitude.

Some of the common applications for airdrop include the delivery of supplies and equipment to troops, delivery of food and medical supplies for peacekeeping or humanitarian operations, re-supply of forest fire fighters, and recently, the strategic deployment of sensors and robotic systems. In each of these cases, the accuracy with which the payload is delivered to the target is often critical to the success of the intended mission. For example, in the case of both military and humanitarian airdrop, there is a high risk that if the payload deviates from its target it may be unattainable by the intended recipients, or may be intercepted by unfriendly forces. In the case of forest fire fighting, the target area is often very small and surrounded by intense fire, and in the case of sensors and robotic systems, often these systems are sensitive to both their landing location as well as impact velocity.
From the need for accurate payload delivery has arisen the development of precision airdrop. The US Army includes in its definition of precision airdrop, “systems that enable safe and accurate delivery of supplies, equipment, vehicles, or personnel from high altitudes. These include autonomous parachute systems (capabilities exist over a wide range of payload weights), mission planning tools, weather forecasting and sensing systems, personnel navigation aids and related integrated communication systems [10].”

This is a rapidly emerging field, largely enabled by the recent reduction in cost of the sensing and processing technologies that facilitate precise and autonomous guidance. Some of the direct advantages presented by precision airdrop vs. non-precision airdrop include [3]:

*High altitude deployment* – The accuracy of non-precision airdrop is greatly affected by wind, and therefore deployment is limited to low altitudes. The ability of precision airdrop systems to be steered in flight allows for high altitude deployment. Cargo aircraft are at much greater risk from ground fire at low altitudes, thus the ability to drop from high altitudes is a great advantage.

*Increased standoff distance* – Standoff is the horizontal distance between the location at which the payload is deployed from the aircraft, and the location of the intended target. Due to the increase in deployment altitude and the ability to steer the system in a desired direction, standoff distance is greatly increased.

*Reduced detection signature* – The combination of increased deployment altitude and standoff distance increases the covertness of payload delivery. Above 20,000 feet it is difficult to hear or see the cargo aircraft that initiates the drop, and once released, guided systems quickly leave the range of visual identification and/or earshot of the aircraft. This means that even though the aircraft may be seen and/or heard, the exact position of the impact location remains ambiguous to an observer.

*Flexible aerial release point* – Because the system can steer to the desired target, it can be deployed from any flight vector within a relatively large area of sky, dependent on wind conditions and the system’s glide characteristics.

*Deployment of Critical Payloads* – Valuable payloads can be delivered with much greater certainty of arriving on target, and in a controllable manner.
1.2 Current Precision Airdrop Vehicles

As defined by the Army, current precision airdrop technologies fit within several categories. Most relevant to this research are the guided vehicles, such as parachutes, parafoils, gliders, autogyros and other un-powered aircraft. The current vehicle types in use or currently under development are presented in this section.

1.2.1 Guided Parafoil

Parafoils have a distinct advantage over parachutes in that their direction and to some extent their glide slope can be controlled. Featuring the ability to maintain glide ratios of up to 3:1, parafoils have the potential to deliver payloads while fulfilling many of the aforementioned benefits of precision airdrop [5]. In addition to their flight performance, parafoils offer high payload/vehicle mass ratios, and because of their non-rigid, deployable structure, they can be efficiently packaged prior to deployment.

Recently, several companies and government agencies have been pursuing the development of autonomous parafoils for the purpose of precision airdrop. Most of these systems are designed for heavy payloads (10,000 to 42,000 pounds) and use GPS based guidance, navigation and control systems to aspire towards target accuracies of 100 meters[5]. While these larger systems focus on the task of troop re-supply, there is a growing need for smaller, UAV-deployed payload delivery systems. One of the applications for this scenario is the deployment of ground based sensor networks. This task requires low weight, small size, and extremely high accuracy. The only current precision airdrop parafoil of this size is the Mosquito, produced by STARA Technologies, Inc [19]. This vehicle is rated for payloads ranging from 1 to 150 pounds, and is claimed to be the only UAV-deployable precision airdrop system. The accuracy of this system was demonstrated during the 2006 Precision Airdrop Conference, when a 5 pound payload was delivered from an altitude of 10,000 feet and a standoff distance of 1.3 miles to within 20 meters of the target [19].

Figure 1.2.1: The Mosquito(a), and Dragonfly(b) precision airdrop parafoils
Along with the benefits of a parafoil based payload delivery system, there are several drawbacks. Because it is a non-rigid system, a parafoil’s canopy must be kept inflated by incoming airflow. This dictates a minimum forward speed requirement to prevent the canopy from collapsing. This forward speed necessity eliminates the ability to descend purely vertically or to change direction of flight at will, and results in a horizontal sliding of the payload during impact. Additionally, due to its non-rigid nature, a parafoil canopy moves independently of its suspended payload, providing no stable reference point to which guidance sensors can be mounted. Under the influence of turbulent winds or aggressive maneuvering, the canopy can collapse on itself, with very little chance of returning to stable flight. These factors all combine to diminish the attainable accuracy of a parafoil system. In addition to this limited accuracy, the deployment of a parafoil system can be a tumultuous event, plagued by tangled lines and partially inflated canopies. Deployment success bears a strong dependence on correct packaging of the chute prior to deployment.

1.2.2 Parafoil Hybrid

In addition to parafoil systems, a parafoil/parachute hybrid has been developed for precision airdrop. The Screamer, developed by Strong Enterprises, has a 2,000 - 10,000 pound payload range, and features a small ram-air drogue parafoil. This parafoil allows for autonomous flight at a very high descent rate towards a pre-programmed target point. After descending to this target altitude and location, two circular cargo parachutes are deployed to arrest forward velocity and affect an uncontrolled standard ballistic descent of 22-28 feet per second [3]. Figure 1.2.2 shows the Screamer with drogue parafoil and cargo parachutes deployed.

![Figure 1.2.2: The Screamer parafoil hybrid precision airdrop system](image-url)
This system falls into the category of a High Altitude Low Opening (HALO) aerial delivery system. Although the system is unguided during the final descent phase, its high descent rate during the controlled phase of its flight makes it difficult to detect and track, and allows very rapid insertion of payloads from high altitudes. The rapid descent rate and low wing loading also make the Screamer less susceptible to course deviations by wind gusts.

The drawbacks of this system are its increased complexity over parafoil systems. The addition of multiple stages of parachute deployment add complexity and time to the pre-flight preparation, and greatly increase the probability of failure during the complex deployment process. Additionally, although the first phase of descent is guided, the last few hundred feet are not, and during this time the aerodynamic loading and descent rate are low, and there is a high susceptibility to deviation from vertical descent. This results in an average accuracy of only 50 to 100 meters.

1.2.3 Autogyro

Prior to proposing the development of the guided Samara, the author was involved in the development of an autogyro precision airdrop system at The Charles Stark Draper Laboratory. This system was designed as a UAV deployable vehicle capable of delivering sensor packages with accuracy of up to 1 meter. The autogyro potentially possessed full omni-directional glide slope control, and maneuverability and accuracy limited primarily by its guidance, navigation and control systems. An autogyro with variable-pitch blades has the ability to flare just prior to landing. This flare allows minute adjustments in position and orientation to be made prior to touchdown, and also allows for very low impact velocity. These benefits clearly place the autogyro concept ahead of parafoil-based systems for the delivery of relatively light payloads requiring extremely high accuracy and control over impact speed. Although the autogyro project did not progress past the conceptual stage, it illustrated the feasibility of an alternative to parafoil based systems when minute control of payload delivery is required.

The drawbacks of the autogyro are its complexity. Effectively the autogyro is a fully functional helicopter lacking only the tail rotor and engine powering its main rotor. This results in a heavy vehicle with limited payload capacity, usually with a payload/vehicle mass ratio not exceeding one. In addition to this low mass ratio, the autogyro’s volume is also significant compared to that of its payload, taking up considerable space prior to its deployment. Another drawback to the autogyro is the complexity of its deployment procedure. To be efficiently packaged, the autogyro must employ compound folding blades. During deployment, a drogue parachute must be released to slow the vehicle’s descent and correctly orient it. The blades must then unfold, and must begin rotating in the correct direction while the vehicle descends under the drogue chute.
1.3 The Samara in Nature

Samara is a Latin word meaning “seed of an elm”, and is a term applied to winged seeds in general [12]. The most recognizable samara is the maple seed, shown in figure 1.3.1 [6].

![Figure 1.3.1: Maple seed](image)

The biological function of the samara is to slow its descent as it falls from its host tree. This increases the horizontal distance it will travel due to wind, which in turn aids the dispersal potential of its species [8]. When a samara initially falls from a tree, the offset between the gravity vector acting at the center of mass (CM) and the resultant drag force creates a yawing moment. This results in a tilted equilibrium orientation, which combined with the yaw rate produces chordwise flow across the wing. The chordwise flow and the yaw rate induce aerodynamic and inertial forces that accelerate the samara’s spin (explained more fully in section 2.1.1). The spin is centered about a point near the CM, while the tip of the wing traces a helical shape. This self-stabilizing, rotating descent is known as autorotation, and is characterized by extraction of energy from the air passing through the rotating blade, and use of this energy to sustain the rotation and create thrust.

Although much research has been conducted to investigate the biological aspects of the samara, relatively little has been done to investigate its flight mechanics. The first major study to do so was conducted by Norberg in 1973 [14]. In this study, simple analysis coupled with experiments using thin, flat plate models were used to examine the autorotational descent of the samara. Some of the topics investigated included an application of momentum theory to describe the aerodynamic interactions of autorotation, an analysis of stability during autorotation, and a brief investigation of the effects of the samara’s form on its flight performance. As presented by Norberg, figure 1.3.2 shows a side view of a samara (looking at the leading edge), and illustrates the forces acting upon it during steady autorotation.
As the samara descends the angle its spanwise axis makes with the horizontal is \( \phi \), referred to as the coning or roll angle. For a particular descent rate and rotation rate, the moments due to lift and centrifugal force create an equilibrium coning angle, and the forces due to lift and centrifugal force dictate the center of rotation. The resultant lift force is shown acting at the blade’s overall center of pressure (CP). During steady descent, the vertical component of lift is equal in magnitude to the samara’s weight. The moment generated by the lift force acting through the moment arm \( r_{cp} \) tends to increase the coning angle. Two centrifugal forces act on either side of the center of rotation. The spanwise location of these two resultant forces, \( r_{cr} \) and \( r_{ct} \), combined with the coning angle, yield the moment arms \( p_{cr} \) ad \( p_{ct} \). The moments generated by the centrifugal forces oppose the moment generated by the lift, and yield a steady state coning angle. At equilibrium, the moments about the CM cancel out, as do the vertical forces. If the center of rotation coincides with the samara’s CM, the opposing centrifugal forces are of the same magnitude and cancel each other out. The horizontal component of lift remains, and tends to accelerate the samara towards the seed. However, the direction of this force varies rapidly, so its effect is negligible, moving the center of rotation only slightly outboard of the CM [14]. Norberg found that when configured correctly, the samara consistently achieves stable flight when dropped from various orientations.

In 1988 Azuma and Yasuda performed a study on the flight performance of rotary seeds. Along with performing a simple momentum theory analysis, the researchers used a vertical wind tunnel to compare the flight characteristics of ten different samara species. Additionally, they performed gliding flight tests using the samara wings to determine the airfoil section properties of the various species [2]. No specific insights were made to build upon Norberg’s study.

In 1991, Seter and Rosen conducted a detailed study of samara autorotation mechanics. They developed a detailed analytical model and numerical simulation, and conducted flight tests with a cardboard model. The numerical results were then compared to the flight test results [18].
1.4 Samara-like Vehicles

The samara has prompted the development of several biomimetic vehicles crafted in its image. The vehicles presented in this section possess similar asymmetric qualities to the natural samara.

1.4.1 Samara-Wing Decelerator

The most developed and documented of these vehicles is the Samara-Wing Decelerator, created by Textron Defense Systems and the US Army Armament Research and Development Center. This system is comprised of a cylindrical payload compartment with an attached rectangular piece of fabric. The fabric “wing” is formed into an aerodynamic surface by the centrifugal load imparted on a weight at its tip. This vehicle must be launched with an initial spin in order to tension the fabric wing, and once spinning causes a passive lunar rotation and steady descent of its payload [4]. Crimi presented a study outlining the development of a seven degree-of-freedom analytical model of this system.

The decelerator has been employed by Textron as the airframe for several munitions systems. The Selectively Targeted Skeet (STS) submunition is an air launched Samara-Wing Decelerator that houses a shaped charge within its payload canister [21]. Figure 1.4.1a shows the STS with its cloth wing deployed, while figure 1.4.1b shows the scan footprint as the STS spins and descends. A sensor scans the footprint in a tightening spiral. When a target is detected, the shaped charge is fired. Textron has produced several variants of this system, but all use the same fundamental design, and more notably, all are passive systems that lack any control once deployed.

![Figure 1.4.1: The Textron Selectively Targeted Skeet (a), and its scan footprint [21]](image)

Although no guided systems were fully developed, Aerojet, Textron, and DARPA have investigated the feasibility of an open-loop, impulse guided samara decelerator [15]. This system consists of a Samara-Wing Decelerator airframe with the addition of a large, single thruster utilizing ball powder propellant. At a specified time, this thruster was activated and resulted in a single triangular thrust pulse of 71,000 newtons (16,000 pounds) over 0.008 seconds. No sensors were used to determine the heading during the impulse,
and the resulting single lateral translation of the submunition occurred in whichever direction the thruster was pointing at the time of ignition. The primary objective of the tests was to demonstrate that the decelerator could be laterally diverted without becoming unstable and losing its scan stability. The results were favorable, and the impulse resulted in a translation of 69 meters (225 ft) during 150 meters (500 ft) of descent.

### 1.4.2 Maple Seed Nano-UAV

Recent articles have vaguely presented the development of a DARPA funded Nano Air Vehicle in the form of a maple seed. The vehicle is under development by Lockheed Martin, and the few technical details that are available indicate that it is to be rocket powered, controllable, and will employ a high-speed imaging sensor to sense heading and to navigate. Lockheed has stated that the vehicle will measure roughly 2 inches long, and will have a maximum takeoff weight of 0.35 oz [17]. This vehicle is illustrated in figure 1.4.2.

![Figure 1.4.2: Lockheed Martin’s maple seed nano air vehicle concept [17]](image)

### 1.5 The Proposed Vehicle

The proposed vehicle employs the form factor of the autorotating samara to create a guided precision airdrop system. In its passive state, if properly configured, the samara can deliver a payload to the ground in an extremely reliable manner, while maintaining a very simple form factor. By employing sensors to monitor the orientation of its blade, a samara airdrop vehicle could actively manipulate its aerodynamic characteristics or mass properties with the necessary timing to grant it control over its flight direction. Such a vehicle could achieve omni-directional glide slope control during its autorotative descent, and it may even be possible to guide the vehicle prior to autorotation, and to control exactly the moment at which the vehicle transitions from free-fall to autorotation.

A benefit of the proposed guided samara is its extremely simple form factor. Such a vehicle would consists of a lightweight wing with a payload attachment point, a sensor and processing package, and an
actuator that modifies the wing’s aerodynamic characteristics or mass properties. This wing could be rigid or non-rigid, fixed or stowable, and could offer flexibility for packaging and deploying the system. The simple form factor provides the capability of maintaining the high payload/vehicle mass ratio of parachute based systems. Another advantage over other precision airdrop vehicles is the simplicity of deployment. It has been observed in natural samaras that they possess self-stability when dropped from any orientation or initial velocity [14]. Additionally, the guided samara requires no lateral velocity to maintain lift, and hence can change directions quickly and can make precise maneuvering possible. For relatively small, valuable payloads, this system offers the potential to greatly exceed the reliability and performance offered by parafoil systems, while offering much lower complexity, cost, and weight than an autogyro system.

1.6 Achievements of this Research

Other than the literature documenting the Textron Samara-Wing Decelerator, there is very little publicly accessible information concerning the development of any samara-like vehicles. So far as the author could determine, the present thesis is the first documented research attempting to examine the feasibility of actively controlling such a vehicle. The guided samara is a novel concept, and through the process of its design and development, several unprecedented discoveries explaining samara flight behavior were made. The most important of these was the discovery that in addition to the center of mass location, the orientation of the principal axes of inertia is critical to a samara’s flight stability. This research consisted of the following milestones:

- Guided samara concept development
- Analytical model and computer simulation development
- Flight testing of passive samara models
- Validation of the computer simulation
- Testing of hardware to be used for the guided samara (actuators, sensors)
- Design and fabrication of guided samara airframe
- Design and fabrication of guided samara control system
- Flight testing of guided samara
- Analysis of flight behavior discoveries, including effect of principal axis of inertia orientation

1.7 Thesis Outline

This thesis consists of seven chapters. A brief description of each is presented:

**Chapter 1 : Introduction & Background** – A description of the scope of the work and the final goal. Presents a background of precision airdrop, and the current vehicles employed for this task. Introduces the natural samara and its autorotative flight, and provides a background on past research. Introduces other samara-like vehicles and introduces the guided samara concept.
Chapter 2: Vehicle Concept Development – This chapter introduces the desired attributes of the vehicle and the design process used to attain them. The design trades are then presented, including the control methodology, the actuator system, the guidance/control/navigation system, and the sensing system.

Chapter 3: Computer Simulation – Presents the development of a six degree-of-freedom analytical model describing the steady autorotative flight of a samara. A simulation was then developed to numerically solve this model in a fast and convenient manner. Two additional simulations were developed to predict the response of two different control schemes used to guide the vehicle.

Chapter 4: Developmental Testing and Simulation Results – To select an actuator and sensor for the guided samara, the performance of these two components were tested. Additionally, this chapter presents flight-testing of two passive samara models, and validation of the simulation results using this flight data.

Chapter 5: Final Vehicle Development and Testing – This chapter presents the design and fabrication of the guided samara’s airframe and control system, and the integration of these two systems. The guided Samara was then flight tested at NASA Langley in an attempt to demonstrate controlled flight.

Chapter 6: Samara Flight Behavior Discoveries – Over the course of the research, many observations about samara flight mechanics were made. These include a proposed mechanism for the enhancement of pitch stability, and the importance of the orientation of the principal axes of inertia.

Chapter 7: Conclusions and Future Work – Presents a summary of the lessons learned and the questions yet to be answered, and suggests future work to advance the guided samara concept.
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Chapter 2

Vehicle Concept Development

This chapter presents the high-level concept development undertaken during the design of the guided samara. During this development, systems were devised for controlling the vehicle both prior to and after its entrance into steady autorotation. Further consideration resulted in pursuing the demonstration of control only during steady autorotation.

The design philosophy revolved around creating an effective test article. All systems were chosen to maximize simplicity, minimize cost, and minimize development time. The design process did not focus on the optimization of the vehicle itself, but rather on developing a control system with which to demonstrate control feasibility. This dictated that the vehicle’s aerodynamic properties and mass distribution be kept as simple as possible, while still providing a stable platform for the demonstration of controlled flight.

2.1 Control Methodology

The first design trade investigated was the mechanism by which the vehicle was to be controlled. The control methodology can be categorized by the two flight regimes experienced by the samara. The first of these is the control of flight prior to autorotation, and the second is the control of flight during steady autorotation.

2.1.1 Prior to Autorotation: Dart-Mode Control

During the initial moments of a samara’s flight, the vehicle begins to descend vertically under the influence of gravity. Regardless of its initial orientation and velocity, its aerodynamic and mass properties dictate that the vehicle will eventually assume a vertical trajectory, with a payload-down, wingtip-up orientation. This descent is dynamically and aerodynamically similar to the flight of a dart or a missile, with the exception that these two objects are generally axisymmetric. Due to the lack of symmetry, the Samara will begin to rotate, and depending on its particular configuration it may enter steady autorotation.
It may be potentially very useful to control the trajectory of the Samara when it is in this “dart-mode” flight regime, and even more useful may be the ability to control when the transition from dart-mode to autorotation occurs. The primary mechanisms devised to control the trajectory and dart-mode transition operate by regulating the vehicle’s aerodynamic symmetry and its mass distribution.

**Mass Distribution Control**

Figure 2.1.1c illustrates the effect of shifting the chordwise center of mass (CM) to coincide with the resultant center of pressure (CP). Figure 2.1.1a presents the yawing moment generated due to the offset between the CM and the CP. Figure 2.1.1b shows the steady state orientation assumed by the samara due to this moment. This tilted orientation during mass first descent begins the transition into autorotation.

![Figure 2.1.1: Effect of chordwise CM location on transition into autorotation](image)

If the CM is in line with the resultant drag vector produced during dart-mode descent, no moments will be generated, and the samara will remain in this orientation. When the CM is shifted forward again, this will trigger the moments that begin autorotation. This CM movement is one mechanism that could be used to control the transition from dart-mode to autorotation.

During the course of this research, a novel discovery about samara flight mechanics was made. All prior research discusses the importance of overall CM location on the flight characteristics of the samara, however, little mention is made of the effect of the chordwise CM distribution. Norberg briefly touched upon the subject of this mass distribution. He segmented a natural samara into nine chordwise strips, and determined the CM of each of these strips. He observed that the chordwise CM location of each strip roughly coincided with the overall CM of the samara. The weighted average of the CM location of each strip formed a spanwise mass axis, which is equivalent to the spanwise principal axis of inertia.
Elaborating on this explanation, one samara could have a wingtip whose sectional CM is very close to the leading edge, while the sectional CM of the wing’s root is very close to the trailing edge. A second samara could have a wingtip CM close to the trailing edge, and a root CM close to the leading edge. Both samaras would share the same overall CM location, but have drastically different mass distributions. This differing mass distribution is characterized by the orientation of their spanwise principal axes of inertia. The first samara’s spanwise principal axis would be tilted forward (towards the leading edge) when looking down the span of the wing, while the second samara’s axis would be tilted back (towards the trailing edge). Figure 2.1.2 illustrates this point by presenting two rectangular “wings” with central CM locations. The point masses on the outside edges of the wings have differing locations, and this causes a different orientation of the principal axes.

Through the course of this research, it was observed during flight-testing and verified analytically, that the orientation of the principal axes causes a coupling between the pitch and rolling (coning) moments present during a samara’s flight. This discovery is presented in Chapter 6. The result of this coupling is that if the spanwise principal axis is parallel to the aerodynamic axis (no tilt), there is no rolling moment produced as a result of pitching moment. If the axis is tilted towards the leading edge, a pitch up moment will result in a rolling moment that tends to reduce the samara’s coning angle. The opposite occurs when the axis is tilted back towards the trailing edge.

Figure 2.1.2: Effect of point mass location on the orientation of the principal axis of inertia

This effect can be used to control the transition from dart-mode to autorotation. Figure 2.1.1b presented the tilted steady state orientation of a samara with a chordwise CM in front of the resultant CP. The pitching moment generated due to this tilted orientation aids the process of decreasing the coning angle and transitioning the samara from dart-mode into autorotation. However, because the rolling moment is coupled to this pitching moment through the principal axis orientation, the coning angle can be controlled. By minutely manipulating the angle of the principal axis, the rolling moment’s sign and magnitude can be adjusted to maintain the dart-mode’s vertical orientation. This can be done very efficiently, by moving the chordwise location of a small mass at the end of the wing. Figure 2.1.3 illustrates the rolling moment generated by moving this mass when the samara is subjected to a pitching moment. A detailed explanation of this behavior can be found in Chapter 6.
In addition to controlling the moment at which transition occurs, the principal axis orientation can also be used to control the vehicle’s trajectory when in dart-mode. The method for controlling transition relies on controlling the magnitude and direction of the rolling moment generated due to a pitching moment. This pitching moment occurs as a function of the chordwise flow component, the angle of attack of this flow, and the airfoil’s moment characteristics. A proposed scheme employs an aerodynamic flap at the wing’s trailing edge to manipulate the sign and magnitude of the pitching moment. By actively manipulating both the pitching moment and the principal axis orientation, control is granted over both pitching and rolling. This allows control of rotation about both the spanwise and chordwise axes, effectively granting full control over the direction and glide slope of the samara’s descent while in dart-mode.

**Aerodynamic Control**

Another method for controlling the guided samara while in dart-mode is to use aerodynamic control surfaces to generate forces and moments. A simple way of achieving this is by employing two orthogonal hinged surfaces at the wing tip. This configuration is presented in figure 2.1.4. In this configuration, one surface creates a moment about the chordwise axis (rolling moment), and the other creates a yawing moment. Figure 2.1.4a illustrates both surfaces in their neutral positions. In this position the vehicle will descend vertically and the moment presented in figure 2.1.1a will cause the autorotation process to begin, although yawing will be hampered by the drag of the control surfaces. Figure 2.1.4b shows typical deflection of the flaps when controlling the vehicle’s trajectory in dart-mode. The flap that controls yaw moment can be deflected alone to incite autorotation. Once the vehicle is in autorotation, this flap can be rotated by 90 degrees so that it then acts as a wingtip plate. Additionally, this surface can be used as the primary source of control during autorotating descent. This control concept is presented in the following section. In addition to these surfaces providing control over the yaw and roll moments, the yaw surface can be split into two opposable surfaces, acting as ailerons to provide pitch moment control. Ideally, the hinge line of this surface would be located at the aerodynamic center, to minimize the required actuator torque.
2.1.2 During Autorotation

The dart-mode control concepts promise to add very useful functionality to the guided samara. However, the problem of guiding the autorotational descent of the samara is both more challenging and potentially more rewarding. The low descent rate characterizing the autorotating flight regime potentially allows for much greater accuracy and a greater glide slope than the dart-mode regime. Consequently, this research focused solely on the development and demonstration of omni-directional glide slope control during autorotation. The three primary concepts considered used the mechanisms of manipulating either the vehicle’s mass distribution or its aerodynamic properties.

Wing Lift Manipulation

This control concept works by actively changing the magnitude of the samara wing’s lift during a portion of one full revolution. As outlined in Chapter 1, during steady autorotation, the moments due to lift and centrifugal force yield a steady state coning angle that remains constant as the samara rotates about a vertical axis. This constant coning angle ensures that the wing’s tip-path-plane remains horizontal (the tip-path forms a plane when observed from a vehicle-fixed reference frame). One method of achieving directional control is to tilt this tip path plane, and force the vehicle to sideslip towards the lowest point of the titled plane. One way of accomplishing this is to actively increase the lift of the wing as it subtends a
particular azimuthal angle, thereby increasing the coning angle within this arc. The subtended angle is here forth referred to as the “actuation angle”. Due to precession, the increase in lift within the actuation arc causes a pitch-up moment. The direction of the resulting sideslip is 180 degrees ahead of the mean angle subtended by the wing during the increase in lift. The magnitude of the sideslip is dictated by either the magnitude of the lift increase, or the magnitude of the actuation angle. Figure 2.1.5a illustrates the conical rotation of the samara, with the red shaded region indicating the actuation arc. Figure 2.1.5b shows the tilting of the samara due to this lift increase. In addition to increasing the lift over one portion of the full rotation, the lift on the opposing side can be decreased to gain greater control authority.

![Figure 2.1.5: Tilting of the samara due to an increase in lift within the shaded region](image)

There are several methods that could be used to implement the periodic lift variation necessary for this control scheme. The first is an aerodynamic control surface located at the trailing edge of the wing. This wing flap is shown in figure 2.1.6. The location of this flap near the tip ensures that it experiences airflow with the highest velocity and with an angle of attack nearest to zero. Both of these factors increase the effectiveness of the flap.

![Figure 2.1.6: Control concept utilizing flap at the trailing edge of the wing](image)
A potential flaw in this method arises from the effect on pitch. The deflection of the control surface will alter the wing’s net pitching moment, and during the concept development stage, the samara’s pitch stability characteristics were not well understood. To compensate for this potential problem, a variation of this design was developed. Two flaps, one inboard and one outboard, can be deflected opposite to one another. The effect of the two flaps would add to form a larger net rolling moment, but would negate one another and form a smaller net pitching moment. Although the inboard flap would have considerably less authority than the outboard flap, the size and deflection angle of the flaps could be configured to minimize the pitching moment as much as possible. Additionally, the inboard flap should not extend outboard of the CM, to ensure that the entire flap generates the desired cone-down moment. This double flap configuration is shown in figure 2.1.7.

![Figure 2.1.7: Control concept utilizing two trailing edge flaps actuated in opposite directions to minimize pitching moment](image)

Another method of varying the lift is by employing a system that changes the wing’s cross-sectional curvature. Assuming the wing consists of a flat membrane, as in a natural samara, it can be actively deformed from a flat to curved surface, effectively adding camber and increasing lift. This is illustrated in figure 2.1.8.

![Figure 2.1.8: Control concept utilizing wing curvature](image)
The final proposed method of varying lift is to actively employ twist about the longitudinal axis of the wing. This twist changes the angle of attack of the root of the wing relative to the tip of the wing. The majority of the lift produced by a rotor is done so in the outermost third of the blade, hence by increasing the angle of attack of this portion of the samara wing, the net lift should increase. This wing twist is illustrated in figure 2.1.9.

![Figure 2.1.9: Control concept utilizing wing twist](image)

**Lateral Force Generation**

It was unknown how variations in lift would affect the stability of the samara, and it was hypothesized that both the induced pitching moments and rolling moments could lie outside the realm of the vehicle’s stability. Until analytical predictions could be made, another control concept was developed and considered. Rather than tilting the vehicle and causing it to sideslip in the desired direction, a wingtip and rudder-like control surface would be used to generate a lateral force. Again, like the wing flap concept, the rudder would deflect during each revolution as the vehicle subtends a specified actuation angle. The resulting lateral force would cause the vehicle to translate towards the center of the actuation angle. This system potentially coincides very well with the aerodynamic dart-mode control concept presented in the previous section. The same control surface used to control the yawing moment during dart-mode could be used to generate lateral force during autorotation. The same configuration from figure 2.1.4c is shown in figure 2.1.10 as it would be used during autorotation.

**Mass Distribution Perturbation**

A control scheme employing the principal axis orientation phenomenon can also be used to control the samara during autorotating descent. Using the same concept employed to control the samara when in dart mode, a small mass can be actively moved to provide control of the spanwise principal axis orientation. The resulting control over the rolling moment can effectively be used in the same manner as an increase in lift, tilting the disk and resulting in a lateral translation.
Figure 2.1.10: Lateral force generation control scheme employing a rudder-like wingtip control surface

2.2 Actuator Systems

All of the control concepts for autorotational flight require periodic actuation matching the vehicle’s rotation rate. To evaluate potential actuation schemes, two main categories were formed. The first includes actuators that directly manipulate a mechanical component, and must be actively commanded to displace in a periodic fashion. An example of this type of actuator would be an electromechanical servo. This device is capable of controlling the position of a control surface, but must be actively commanded to move the control surface in a periodic manner matching the samara’s rotational frequency. The second category includes mechanical systems that convert rotational motion into periodic linear motion.

2.2.1 Direct Actuation

All of the control schemes presented in the previous section could potentially use direct actuation. The most readily accessible and easy to use actuator in this category is the hobby servo. This consists of a packaged gearbox, motor, potentiometer, and microcontroller providing control of output position. The pulse width modulation communication standard is well documented and easy to implement, and there is a large selection of servos with varying performance. Although the maximum slew rate of servos is specified, it was unknown if the servos could maintain this slew rate under load and during rapid periodic actuation matching the samara’s rotation rate.

Another actuator considered in lieu of a servo is a micro solenoid. These systems consist of wire looped around a metallic core, and when provided with current produce linear stroke of the core. Without the limitations of a gearbox and built in controller, it is possible that the dynamic response of a solenoid would surpass that of a servo, especially at very high actuation frequencies.
The last type of direct actuator that was considered was piezoelectric. These actuators have been used with success for ornithopter type vehicles to produce flapping motions. One benefit of piezoelectrics is extreme precision of displacement. Another advantage is that piezoelectrics are available as polymer films, whose displacement occurs as curvature. This is ideal for use with the aerodynamic control concepts presented in the last section. Varying the curvature of a sheet can be used to create an aerodynamic control surface, or to vary wing curvature or wing twist. These actuators have been shown to produce significant force and can actuate rapidly and at high cyclic frequencies. The major downside is cost, and the requirement of extremely high voltages, with displacement being proportional to this voltage.

2.2.2 Indirect Actuation Mechanism

An alternative to employing direct actuation is to use a periodic actuation mechanism. The benefits of such a mechanism are that phasing of actuation is achieved by the mechanism, rather than by the control system commanding a direct actuator. Two types of such a mechanism include the swashplate and the crankshaft. Both of these designs alleviate the need to command the position of an actuator to follow a sinusoid or triangle wave of a particular amplitude and frequency.

A major hurdle in implementing either of these mechanisms on the guided samara is the lack of a stationary reference point on the vehicle. The swashplate functions on a conventional helicopter by allowing one disk to rotate with the rotor, and one disk to remain stationary relative to the helicopter’s body. In this manner, actuators on the helicopter can tilt the disk and determine the amplitude and phasing of the oscillatory motion of the swashplate’s output. To implement either design on the guided samara, a control system, sensor and electric motor would need to be used to effectively create a stationary reference.

In the case of a swashplate, the design from a helicopter can be used as is, with the addition of a de-spin motor. This would require a closed loop controller and sensor to determine the vehicle’s rotation rate and to spin the swashplate disk at the same rate in the opposite direction. This would create the stationary reference required for the mechanism to function.

In the case of a crankshaft design, the same control loop and motor are required, but rather than holding a disk stationary, they would rotate the crankshaft at a rate matching the vehicle’s rotation rate, and hence the resulting linear oscillation would also match the rotation rate. An additional mechanism could vary the displacement of the linear oscillation, and lagging or advancing the phasing of the crankshaft would achieve directional control of the actuation arc. The control loop necessary to achieve this is known as a phase-locked loop, and is well documented and widely used in various applications.

2.3 Control, Guidance, and Navigation

The overall goal is to achieve command of the vehicle’s position, orientation, and velocity. There are three levels from which this must be achieved; control, guidance and navigation. These levels will be presented
and related to work conducted during this research. There are numerous interpretations of these three terms, but the following definitions were adopted for the purpose of this project.

2.3.1 Control

Control is provided by the system that actively creates a change in the vehicle’s position and orientation. In the case of the guided samara, this system consists of an azimuthal angle sensor, control software, and an actuator that manipulates a control surface or a mass. Control can be used to facilitate the desired change in a vehicle’s position, or can be used to add stability to a naturally unstable vehicle. When properly configured, the guided samara will be inherently stable during autorotation, so the control system will only be used to facilitate movement for guidance purposes. The control software implemented for the guided samara will be open-loop. A sensor will monitor the azimuthal angle of the wing, and based on this sensor, an actuator will re-configure the vehicle so that movement translation occurs in the desired heading.

2.3.2 Guidance

Guidance is the practice of moving the vehicle from a current state (position and velocity) to a desired state. Demonstrating guidance of the samara during autorotation is the end goal of this research. For this project, this will be an open-loop process. The guidance system will command the control system to create motion in the desired heading, but there will be no feedback to determine if the vehicle’s movement is correct. The guidance software will be created such that a human operator can use a remote transmitter to visually close the loop and control the direction of the vehicle.

2.3.3 Navigation

Navigation is the monitoring of the vehicle’s state, and it purpose is to facilitate guidance. The long-term vision of the guided samara includes autonomous guidance. For this to be feasible, various sensors (inertial, magnetic, GPS) would be used fully describe the vehicle’s state, and to allow for closed-loop guidance. However, this will not be a focus of the current research.

2.4 Azimuthal Angle Sensors

To achieve control and guidance of the samara, it is necessary for the control system to sense the azimuthal angle of the vehicle as it rotates. This sensing capability must occur with sufficient resolution to provide accurate heading control, and at a sufficient sampling rate to prevent aliasing. Another requirement is that the sensor maintains its accuracy during the constant rotation and tilting of the samara.

The first sensor considered was the 2-axis electronic compass. This system uses two magnetic sensors placed at a right angle to each other to measure the earth’s magnetic field. A compass made this
way determines heading by an arctan function of the output of each sensor and is accurate only when held
in a level orientation. When the compass is tilted, heading errors occur as a function of the tilt angle and the
decision of the earth’s magnetic field at the particular geographic location [16]. The heading error for a
2-axis system can be calculated as follows:

\[
Error = \tan^{-1}\left(\frac{V_m \sin \beta}{H_m}\right) \tag{2.1}
\]

Where \((H_m)\) and \((V_m)\) are the earth’s horizontal and vertical magnetic field intensities in nanoTesla. A
sample calculation for the Boston area indicates that for 15 degrees of tilt, the associated heading error
would be roughly 33 degrees.

The samara rotates about a vertical axis with non-zero steady state roll and pitch angles. Additionally, during control inputs, these angles may change, causing a 2-axis compass sensor to produce
considerable heading error. A potential solution to this problem is a 3-axis electronic compass sensor. This
sensor uses three individual magnetic sensors mounted orthogonally. An accelerometer is used to determine
the orientation of the sensor relative to the gravity vector. This configuration allows for the accurate
computation of heading regardless of the tilt applied [16]. Unfortunately, the constant rotation of the
samara will cause an error in the accelerometer reading, hence resulting in a tilt error of the compass.

The last sensor considered was a piezoelectric rate gyro, which utilizes the Coriolis effect to
measure rotation rate. The rate signal must then be integrated to yield angular position. Generally there is a
cumulative error associated with this integration. However, because the guided samara’s descending flight
is expected to last a relatively short time, the error will be negligible. Test results have indicated that a
representative gyro, the Memsense Trirate, produces three degrees of drift during one minute of integration.
This is well within the acceptable range of heading error for the purpose of demonstrating control of a
samara. A drawback of the rate gyro is that its position measurement is relative to the heading at which the
integration begins. This means that in order to measure heading in earth’s reference frame, the gyro must be
initialized in a particular orientation prior to each flight.

2.5 Design Selection

As stated at the beginning of the chapter, the design philosophy for the guided samara revolved around
creating an effective test article. All systems were chosen to maximize simplicity, minimize cost, and
minimize development time. Additionally, it was decided to only pursue the development of control during
the autorotative flight regime.

Prior to this research, the effect of the principal axis orientation was not known and therefore was
not considered as a potential control scheme. Among the aerodynamic schemes, the wing trailing edge flap
design intuitively was selected as the most simple to implement and potentially the most effective way of
creating lateral translation. When selecting a control scheme, many questions arose regarding the stability of the samara and its response to a periodic flapping and the resulting tilting. It was decided that the simulation would be used to predict and compare the performance of the wing flap control scheme to the wingtip rudder control scheme. This would provide some level of insight into the drawback and benefits of each scheme before a commitment to either one was made.

The obvious choice for an actuator was the hobby servo, however, it was not known if this system could provide adequate performance to be used for the guided samara. To test this performance, experimentation was conducted using four off-the-shelf servos. The results of these tests are presented in Chapter 4.

The 3-axis electronic compass was eliminated from consideration based on its inability to operate during continuous rotation. The ideal sensor from a perspective of ease of use and interfacing was the 2-axis electronic compass. It was decided that based on simulation results and tilt error experimentation, it would be determined how much tilt the compass would be subjected to, and how much error this tilt would cause. If this error were great enough to prevent demonstration of control of the guided samara, the rate gyro would be used as a second option.
Chapter 3

Dynamic Simulation

The samara is a freely rotating rigid body with no reaction forces to maintain its orientation or position. Therefore, during the design process there was great uncertainty about the vehicle’s potential reaction to changes in its lift characteristics or mass properties. To mitigate this uncertainty, an in depth analysis of the vehicle’s dynamics and aerodynamics was performed. The goals of this analysis were to better understand the steady state flight behavior of a samara and to predict the transient response during controlled flight. First, an analytical model describing the samara’s behavior was developed. Next, a time domain computer simulation was created to numerically solve this analytical model. In addition to predicting transient response, the simulation also served as a design tool for tailoring the vehicle’s characteristics in order to attain desired steady state performance.

3.1 Analytical Model

The first step in creating the simulation was the development of an analytical model to describe the vehicle’s dynamics and aerodynamics. This analytical model can be broken into two parts; the equations of motion describing the six-degree-of-freedom rigid body dynamics of the samara, and the equations governing the external forces and moments generated by aerodynamic interactions with the wing.

3.1.1 Equations of Motion

During flight the Samara is a fully unconstrained rigid body, and is free to rotate and translate in all six degrees of freedom. To describe this motion, a standard formulation for the rigid body motion of a spacecraft was used [9]. Two reference frames were first defined. The mass center of the vehicle was chosen as the origin of the body-fixed reference frame (body frame). An inertial frame defined the body frame’s location in space. The inertial and body frames are shown in figure 3.1.1.
Euler angles were used to perform three individual rotations to relate inertial frame quantities to body frame. To begin this transformation, the body frame and inertial frame are initially aligned with one another. The three individual rotations are then sequentially performed to re-orient the body frame:

1) Rotate by $\psi$ about $\hat{i}_3$. This results in new coordinate system ($\hat{b}_1', \hat{b}_2', \hat{b}_3'$)

2) Rotate the new coordinate system by $\theta$ about $\hat{b}_2'$. This results in the coordinate system ($\hat{b}_1'', \hat{b}_2'', \hat{b}_3''$)

3) Rotate the new coordinate system by $\phi$ about $\hat{b}_1''$. This results in the body frame ($\hat{b}_1, \hat{b}_2, \hat{b}_3$)

The Euler angles in this case are $\psi$, $\theta$, and $\phi$, which are respectively equivalent to yaw, pitch, and roll when using aircraft terminology. The individual rotations can be performed in a number of sequences. The chosen rotation order is known as the 3-2-1 sequence because of the labeling of the axes about which the rotations occur. The use of Euler angles results in a singularity when the rigid body reaches a particular orientation. The singularity associated with the 3-2-1 sequence occurs at $\theta = \pi/2$, where rotation about $\psi$ and $\phi$ cannot be discerned from one another. This singularity did not pose a problem in modeling the Samara, as there is no flight condition in which the pitch angle should approach $\pi/2$.

Each step of the sequence can be represented in matrix form. The three resulting matrices are referred to as the principal rotation matrices, $C_1$, $C_2$, and $C_3$, presented in the order of the sequence:

\[
\begin{bmatrix}
\hat{b}_1' \\
\hat{b}_2' \\
\hat{b}_3'
\end{bmatrix} = \begin{bmatrix}
\cos \psi & \sin \psi & 0 \\
\sin \psi & \cos \psi & 0 \\
0 & 0 & 1
\end{bmatrix} \begin{bmatrix}
\hat{i}_1 \\
\hat{i}_2 \\
\hat{i}_3
\end{bmatrix} = C_3(\psi) \begin{bmatrix}
\hat{i}_1 \\
\hat{i}_2 \\
\hat{i}_3
\end{bmatrix}
\] (3.1)

\[
\begin{bmatrix}
\hat{b}_1'' \\
\hat{b}_2'' \\
\hat{b}_3''
\end{bmatrix} = \begin{bmatrix}
\cos \theta & -\sin \theta & 0 \\
0 & 1 & 0 \\
\sin \theta & \cos \theta & 0
\end{bmatrix} \begin{bmatrix}
\hat{b}_1' \\
\hat{b}_2' \\
\hat{b}_3'
\end{bmatrix} = C_2(\theta) \begin{bmatrix}
\hat{b}_1' \\
\hat{b}_2' \\
\hat{b}_3'
\end{bmatrix}
\] (3.2)
\[
\begin{bmatrix}
\hat{b}_1 \\
\hat{b}_2 \\
\hat{b}_3
\end{bmatrix} = \begin{bmatrix} 1 & 0 & 0 \\
0 & \cos\phi & \sin\phi \\
0 & -\sin\phi & \cos\phi
\end{bmatrix} \begin{bmatrix}
\hat{i}_1 \\
\hat{i}_2 \\
\hat{i}_3
\end{bmatrix} = C_1(\phi) \hat{i}_1 = C_2(\theta) \hat{i}_2 = C_3(\psi) \hat{i}_1
\]

The three principal rotation matrices combine to yield the direction cosine matrix:

\[
C(\Theta) = C_1(\Theta_1)C_2(\Theta_2)C_3(\Theta_3) = C_1(\phi)C_2(\theta)C_3(\psi)
\]

Substituting the principal rotation matrices into equation (3.4):

\[
C(\Theta) = \begin{bmatrix} 1 & 0 & 0 \\
0 & \cos\theta & -\sin\theta \\
0 & \sin\theta & \cos\theta
\end{bmatrix} \begin{bmatrix} \cos\phi & \sin\phi & 0 \\
-\sin\phi & \cos\phi & 0 \\
0 & 0 & 1
\end{bmatrix}
\]

and performing matrix multiplication:

\[
C(\Theta) = \begin{bmatrix} 
\cos\phi \cos\psi & \cos\phi \sin\psi & \sin\phi \\
\cos\theta \cos\psi - \sin\theta \sin\phi \sin\psi & \cos\theta \sin\psi + \sin\theta \cos\phi \sin\psi & \sin\theta \cos\phi \\
\sin\phi \cos\psi + \cos\phi \sin\theta \sin\psi & \sin\phi \sin\psi - \cos\phi \cos\theta \sin\theta & \cos\phi \cos\theta
\end{bmatrix}
\]

yields the final matrix that represents the transformation from inertial to body frame coordinates. The complete transformation occurs using the following equation:

\[
\begin{bmatrix}
\hat{b}_1 \\
\hat{b}_2 \\
\hat{b}_3
\end{bmatrix} = C(\Theta) \begin{bmatrix}
\hat{i}_1 \\
\hat{i}_2 \\
\hat{i}_3
\end{bmatrix}
\]

The next step is to obtain the relationship between the angular velocity components and the time derivatives of the Euler angles. For the purpose of simplifying the derivation, rather than re-computing cross products, a pre-formed matrix is used as a multiplier to perform a cross operation. The notation \( \times \) is used to represent this pre-formed skew symmetric matrix [9]. The pre-formed matrix takes the form:

\[
u^\times = \begin{bmatrix} 0 & -u_3 & u_2 \\
u_3 & 0 & -u_1 \\
u_3 & u_1 & 0
\end{bmatrix}
\]

Using this notation a cross product is computed as follows:

\[
\vec{u} \times \vec{v} \rightarrow u^\times v
\]
To relate the angular velocity components to the Euler time derivatives, the following S matrix is defined [9]:

\[
S = \begin{bmatrix}
1 & 0 & -\sin \theta \\
0 & \cos \phi & \sin \phi \cos \theta \\
0 & -\sin \phi & \cos \phi \cos \theta
\end{bmatrix}
\]  

(3.10)

The angular velocity component is:

\[
\omega = S \dot{\Theta}
\]

(3.11)

so the Euler time derivates can be expressed as:

\[
\dot{\Theta} = S^{-1} \omega
\]

(3.12)

where:

\[
S^{-1} = \begin{bmatrix}
1 & \sin \phi \tan \theta & \cos \phi \tan \theta \\
0 & \cos \phi & -\sin \phi \\
0 & \sin \phi \sec \theta & \cos \phi \sec \theta
\end{bmatrix}
\]

(3.13)

The singularity associated with the 3-2-1 sequence makes itself apparent when the \( S^{-1} \) matrix is created. At \( \theta = \pi/2 \), where the singularity occurs, the \( S^{-1} \) matrix becomes undefined. Equation (3.12) may now be integrated to yield the yaw, pitch, and roll orientations. Before the scalar equations of motion are developed, the state vector \( \mathbf{X} \) is assembled:

\[
\mathbf{X} = \begin{bmatrix}
\omega \\
\Theta \\
\mathbf{V} \\
\mathbf{R}
\end{bmatrix}
\]

(3.14)

This vector fully describes the position, orientation, translational rate, and rotational rate of the vehicle. The state vector quantities \( \mathbf{V} \) and \( \mathbf{R} \) refer to the origin of the body frame, which is located at the mass center of the vehicle. The individual components of the state vector are as follows:

- \( \omega \): roll, pitch, and yaw rate of the vehicle in the body frame

\[
\omega = \begin{bmatrix}
\omega_1 \\
\omega_2 \\
\omega_3
\end{bmatrix}
\]

(3.15)

- \( \Theta \): roll, pitch, and yaw orientation of the vehicle in the body frame

\[
\Theta = \begin{bmatrix}
\phi \\
\theta \\
\psi
\end{bmatrix}
\]

(3.16)
• \( V \): velocity of the body frame origin relative to the inertial frame, expressed in body frame

\[
V = \begin{bmatrix} v_1 & v_2 & v_3 \end{bmatrix}^T
\]  

(3.17)

• \( R \): position of the body frame origin relative to the inertial frame, expressed in body frame

\[
R = \begin{bmatrix} R_1 & R_2 & R_3 \end{bmatrix}^T
\]  

(3.18)

In the formulation of the scalar equations of motion, external forces (which includes gravity) are represented by the variable \( f \), and external torques are represented by the variable \( M \). The equation of linear momentum is:

\[
\dot{P} = -\omega \times P + f
\]  

(3.19)

where:

\[
P = mV \quad \text{and} \quad \dot{P} = m \dot{V}
\]  

(3.20)

Substituting equation (3.20) into equation (3.19) yields:

\[
m \ddot{V} = -m\omega \times V + f
\]  

(3.21)

which simplifies to the final expression:

\[
\ddot{V} = \frac{f}{m} - \omega \times V
\]  

(3.22)

The angular equation of motion is:

\[
\dot{H} = -\omega \times H + M
\]  

(3.23)

where:

\[
H = I\omega \quad \text{and} \quad \dot{H} = I \dot{\omega}
\]  

(3.24)

Substituting equation (3.24) into equation (3.23) yields:

\[
I \ddot{\omega} = -I\omega \times I\omega + M
\]  

(3.25)

which simplifies to the final expression:

\[
\ddot{\omega} = I^{-1} \left( -\omega \times I\omega + M \right)
\]  

(3.26)

Through this substitution, the momentum variables have been eliminated. The final expressions, equation (3.22) and equation (3.26), may now be integrated to yield \( V(t) \) and \( \omega(t) \). The following is the kinematic equation:
\[ 0.7 R \cdot R = V \]  

Equation (3.28) is the final equation of motion. This expression is integrated to yield \( R(t) \), which is expressed in the body frame. \( R(t) \) can subsequently be transformed to inertial coordinates for final output. All of the equations needed to form the derivative of the state vector have been developed.

### 3.1.2 Aerodynamic Forces and Moments

To complete the analytical model of the samara, external forces and moments applied to the vehicle during flight had to be considered. These forces and moments consist of gravity and aerodynamic interactions with the surfaces of the vehicle. The rotation of a samara blade is very similar to that of a conventional helicopter rotor. The main differences are that the samara consists of a single blade, and the fact that this blade is mechanically fully unconstrained. Because of the aerodynamic similarities to a conventional rotor blade, classical blade element momentum theory (BEMT) was used to model the Samara.

As its name suggests, BEMT is a combination of momentum theory and blade element theory. Momentum theory was developed on the basis that a spinning rotor is composed of an infinite number of blades that can be represented as an actuator disk. This disk uniformly changes the speed of air passing through it, and the power required to produce thrust is represented only by the axial kinetic energy imparted to the air composing the slipstream [7]. The change in velocity of the air from its initial value to its value at the rotor disk is called the induced velocity (\( v_i \)).

Blade element theory takes into consideration the contribution of many 2-dimensional slices of each rotor blade. This provides a model that estimates the radial and azimuthal variations in aerodynamic loading over the entire rotor disk. The theory treats each element of the rotor as a quasi 2-D airfoil that produces forces and moments independently of adjoining elements. The thrust and torque of the rotor are obtained by integrating the individual contribution of each element along each blade’s length [11]. An example blade element and the resulting elemental lift and drag it creates are illustrated in figure 3.1.3.

As a hybrid of the two methods, BEMT invokes equivalence between the circulation and momentum theories of lift, and in doing so, allows the calculation of the distribution of induced flow along the blade [11]. The direct result of combining momentum theory and blade element theory is that momentum considerations are applied to an annulus of the rotor disk. This is the formulation used in modeling the aerodynamic forces and moments acting on the samara.
Figure 3.1.2: Flow model for momentum theory analysis of a descending rotor

Figure 3.1.3: Blade element analysis illustrating airflow and resultant elemental lift and drag
Before developing the equations used to describe the aerodynamics, the following assumptions were made. The first defined the flight regime of the samara. It was assumed that the model would describe a samara once it entered autorotation; due to the limitations of the model, the initial conditions include a non-zero descent rate, a non-zero yaw rate, and an orientation that is representative of steady autorotation. All other rates were generally assumed to be zero. Next, it was assumed that during the course of its entire flight, the samara “rotor” operates within the windmill brake state. This is the aerodynamic state in which a rotor extracts energy from the flow, and brakes the flow velocity like a windmill [11]. This is significant to the modeling effort because within this state, a standard formulation for induced flow can be applied. It was assumed that radial airflow would have a negligible effect, and was therefore ignored in the modeling. It was also assumed that the 2-D assumptions made when applying BEMT would provide enough fidelity that correction for 3-D effects such as tip-losses were not necessary. With these assumptions, the model has validity for its intended purpose.

Before discussion of the aerodynamic model begins, the coordinate system used is presented in figure 3.1.4. This system is equivalent to the body frame system used in the dynamic model developed previously, however the \((\hat{b}_1, \hat{b}_2, \hat{b}_3)\) axis labels have been replaced with the more convenient \((X,Y,Z)\) notation. Note that the \(Z\) direction is defined as positive downwards. This convention is applied to the lift, drag, and all other forces and moments.

![Coordinate system used for aerodynamic analysis](image)

The aerodynamic modeling begins by focusing on a single 2-D blade element. For a particular blade element, the vertical component of velocity due to the vehicle’s motion is computed. This component is computed by subtracting the velocity at the element due to the vehicle’s roll from the velocity due to the vehicle’s descent rate:

\[
V_y = -v_3 - \omega_t (r - CM_y)
\]

(3.29)

where:

\[
r = yR_t
\]

(3.30)

To obtain an expression for induced flow, blade element theory is used to generate an expression for differential thrust produced by a blade element. This derivation is presented by Gessow and Meyers [7].
\[ dT = b \frac{1}{2} \rho (\omega_3 r)^2 d(\theta - \Phi) c d r \] (3.31)

Next, momentum theory is used to generate an expression for differential thrust of an annulus:

\[ dT = 4 \pi \rho u v_r c d r \] (3.32)

Equations (3.31) and (3.32) are equated and the resulting quadratic is solved for induced velocity to yield:

\[ v_i = \left( \frac{V_x}{2} + \frac{a \omega_3 r}{16} \right) \left( -1 + \sqrt{1 + \frac{2 \theta \omega_3 R - V_i}{4 V_e^2 + V_e + \frac{a \omega_3 r}{16}}} \right) \] (3.33)

Equation (3.33) is used to compute the induced flow at the blade element in consideration. Knowing the vertical velocity due to vehicle motion and the induced flow velocity, the resultant vertical velocity at the element can be computed:

\[ U_p = v_i + V_v \] (3.34)

Next the resultant tangential velocity of the blade element is computed. This consists of the velocity due to the spinning (yawing) of the wing, in addition to the lateral translation of the vehicle in the x-direction:

\[ U_t = -\omega_3 (r - CM_3) - v_i \] (3.35)

The horizontal and vertical velocity components are combined to yield the overall resultant velocity:

\[ U = \sqrt{U_p^2 + U_t^2} \] (3.36)

Next, the resultant velocity vector’s angle with respect to the horizontal is computed:

\[ \Phi = \tan^{-1} \left( \frac{U_p}{U_t} \right) \] (3.37)

The blade element’s angle of attack is computed by adding resultant velocity’s angle to the pitch angle:

\[ \alpha = \Phi + \theta \] (3.38)

Next, the lift and drag coefficients of the blade section are determined as a function of the angle of attack. These data can take different forms depending on the airfoil being represented in the model. During the course of this analysis, several different sets of data were used, however, the most basic airfoil represented was a flat plate. Flat plate wind tunnel data for high angles of attack (up to 90 degrees) shown in figure 3.1.5, were obtained from NACA technical report 3221 [22].
Using the lift and drag coefficients, the incremental lift and drag produced by the blade element are computed:

\[ dL = -\frac{1}{2} \rho U^2 c C_L dr \]  

(3.39)

\[ dD = -\frac{1}{2} \rho U^2 c C_D dr \]  

(3.40)

Lift is generated normal to the resultant air velocity experienced by an airfoil, while drag is parallel to this velocity. The dynamics portion of the samara model requires external forces and moments to be specified in the body frame. To facilitate this, lift and drag must be rotated into this frame. Note that by the sign convention defined in figure 3.1.4, lift is positive downwards, and drag is positive forwards. The 2-dimensional coordinate rotation to transform lift and drag to normal and axial forces is illustrated in figure 3.1.6. The results of this transformation are the following expressions for incremental normal and axial forces:

\[ dN = dL \cos(\alpha) + dD \sin(\alpha) \]  

(3.41)

\[ dA = dD \cos(\alpha) - dL \sin(\alpha) \]  

(3.42)
The sign convention for angle of attack is positive pitch up. All other vectors follow the sign convention set forth by the coordinate system illustrated in figure 3.1.4. Once the normal and axial forces are obtained, the pitching, rolling and yawing moments generated by the individual blade element can be computed. The moment arms used for these computations depend of the radial distance of the particular blade element from the vehicle’s CM, as well as the chordwise distance from the center of pressure to the CM. The expressions used to compute the aerodynamic moments generated by each blade element about the vehicle’s center of mass are:

\[ M_x = N(r - CM_x) \]  \hspace{1cm} (3.43)

\[ M_y = N(CM_y) \]  \hspace{1cm} (3.44)

\[ M_z = -A(r - CM_z) \]  \hspace{1cm} (3.45)

Figure 3.1.7 shows the conventions used to define the CM locations. The signs follow the coordinate system convention.
In the analysis, several methods were used to estimate the center of pressure location. One method assumed that the center of pressure remained fixed at the quarter-chord position, then moved to half-chord during stall, while another method used data from NACA TN-3221 to relate center of pressure location to angle of attack.

Once all forces and moments have been computed for a blade element, the process is repeated for the remaining elements. The forces and moments are then integrated, and passed back to the dynamic part of the analysis as the resultant external forces and moments imparted on the samara.

### 3.2 Free-flight Simulation Development

With the dynamic and aerodynamic analytical model complete, a time domain computer simulation was created to numerically solve the model’s equations. The simulation was implemented in Matlab, and was programmed as several individual subroutines. This simulation contains the necessary equations of motion and aerodynamic interactions to model a samara in free-flight descent. Figure 3.2.1 illustrates the individual subroutines and the sequence of data-flow between them.

![Flowchart of the samara simulation implementation](image)

Figure 3.2.1: Flowchart of the samara simulation implementation
Once the simulation was assembled, the rigid body dynamics portion was validated. A study conducted by H.S Morton included the development of a rigid body simulation, including numerical results [13]. The Samara simulation was executed using the same mass properties and initial conditions as the simulation in Morton’s study, while ensuring that the aerodynamic subroutine was deactivated. The simulation results matched exactly, providing verification that the rigid body dynamics portion of the simulation functioned correctly. The complete simulation was validated by comparing results to flight tests of a model samara. This is presented in Chapter 4.

3.3 Controlled Simulation Development

A second simulation was developed to include the modeling of a flap actuation scheme. Two schemes were modeled, a flap on the trailing edge of the wing, and a wingtip with a flap.

3.3.1 Wing Trailing Edge Flap

To model the addition of a flap, a standard formulation for the effect of a plain flap on a wing’s lift and pitching moment was used [1]. The formulation includes the use of empirical data to determine the following parameters as a function of the ratio of the flap chord to the wing chord:

\[
\frac{dC_{lf}}{dC_{f}}, \quad \frac{dC_{l}^{2}}{d\delta}, \quad \frac{dC_{m_{lf}}}{d\delta}
\]

The subscript \(f\) indicates that the coefficients refer to the flap rather than the entire wing, while \(\delta\) is the flap deflection angle. These parameters were determined from plots in the text, and were then used to determine the lift and moment coefficients of the flap at a particular deflection. The following equation was used to compute the flap’s lift coefficient as a function of the wing’s lift coefficient, and the flap’s deflection angle:

\[
C_{lf} = \frac{dC_{lf}}{dC_{f}} C_{l} + \frac{dC_{l}^{2}}{d\delta} \delta
\]  

(3.46)

This coefficient was then used to compute the lift the flap generates in addition to the undeflected wing’s lift:

\[
L_{f} = \frac{1}{2} C_{lf} \rho U^2 c_f dr
\]

(3.47)

The \(1/4\) chord moment coefficient due to the flap’s deflection was then calculated:
\[ C_{mf_{1/4}} = \frac{dC_{m_{1/4}}}{d\delta} \]  

(3.48)

This coefficient was then used to compute the pitching moment added by the flap’s deflection:

\[ M_{f_{1/4}} = -\frac{1}{2} C_{mf_{1/4}} \rho U^2 c dr \]  

(3.49)

The increase in drag due to the flap deflection was modeled as an increase in profile drag, while the increase in induced drag was ignored. A new overall lift coefficient of the wing section was determined based on the increase in lift due to the flap. This new lift coefficient was then used to determine a corresponding increased drag value based on the airfoil section’s lift/drag relationship.

Once the aerodynamic model used to represent a flap actuation was developed, the original free-flight simulation was modified to reflect the addition of this flap. The “body initial condition” subroutine was modified to include the following user specified parameters: flap span, spanwise flap location, flap chord, flap deflection angle, the desired vehicle heading, the azimuthal arc over which the actuation should occur, and a flap actuation start and stop time. These times are relative to the overall run-time of the simulation. Finally, the “aerodynamics” subroutine was modified to allow the addition of flap lift force, drag force, and pitching moment to the calculations performed for each blade element. A logical check was added to the simulation to verify if each particular blade element computation should include the computation of the flap’s affects. The logical check included the following if statements:

- If time is greater than actuator start time and less than actuator stop time
- If the blade element is in the span of the flap
- If the wing’s yaw (azimuthal) angle is within the actuation arc

When all three conditions were true, the lift, drag and moment were calculated and added to the baseline values generated by the blade element. The simulation was executed using the same parameters as the free-flight simulation. The results are presented in Chapter 4.

### 3.3.2 Wingtip and Rudder Actuation

In addition to developing a simulation to predict response created by a flap at the wing’s trailing edge, a simulation was developed to model control using a wingtip with a rudder. This rudder was periodically actuated to produce a lateral force and translate the vehicle in the desired direction without considerably tilting the tip path plane. This model used the same formulation presented in the section on the wing trailing edge flap. At the end of the wing, rather than completing the spanwise integration of the blade elements, the integration continues vertically. This vertical integration represents the wingtip. The same method for modeling a flap was used to add a rudder to this wingtip. Again, this simulation was executed with the same parameters used for the previous two simulations. The results are presented in Chapter 4.
Chapter 4

Developmental Testing & Simulation Validation

With the simulation complete, the testing of free flight samara models and various control system hardware was conducted. The goal of this experimentation was twofold; to validate and gain faith in the simulation results, and to verify that the hardware to be used to control the guided samara fulfilled the required performance criteria.

4.1 Free-flight Samara Prototype

To validate the simulation, two free-flight vehicles were designed, constructed, and test flown. The resulting flights were captured on video from multiple perspectives, and by analyzing this video, the vehicle’s rotation rate, descent rate, and coning angles could be determined.

4.1.1 First Iteration

The first prototype samara was designed to resemble a natural maple seed. This included a thin “membrane” for a wing, a stiff, tubular leading edge, thin chordwise ribs to support the wing, and a dense payload connected to the leading edge spar. The planform included the taper at the root that is commonly seen on maple seeds. The design for the first prototype is shown in figure 4.1.1. The “membrane” used for the wing surface was 3mm Depron, a low-density foam manufactured in thin sheets. The manufacturing process results in a thin skin on each side of the foam, which gives it considerable stiffness for its weight. The main spar at the leading edge was comprised of a carbon tube with an outer diameter of 5/16 inch. The chordwise ribs were constructed of 1/16 inch steel music wire, and were mated with precisely drilled holes in the carbon spar using epoxy. The Depron was then cut to the correct planform and attached to the leading edge spar and ribs using light tape. To allow easy adjustability of the vehicle’s center of mass (CM) location, a payload mount was machined. This mount allowed the payload to be moved in the chordwise direction, as well as for the entire mount to pivot about the leading edge spar. An identical mount was
created for the wingtip. This allowed for optional adjustment of the chordwise CM location by adding weight to the tip. The fabricated vehicle is shown in figure 4.1.2.

![Figure 4.1.1 First free-flight samara design](image1)

The fabricated vehicle had a total span of 0.74 m (29 in), a chord of 0.15 m (6 in), and a total mass of 0.31 kg (0.69 lb). Using the solid model presented in figure 4.1.1, the CM location and inertia matrix of the vehicle during its initial flight tests were determined. As presented in figure 4.1.3, the CM locations in the X, Y and Z axes are respectively measured from the wing’s leading edge, root, and the vertical center of the airfoil.
The vehicle’s CM location (in meters):

\[
CM = \begin{bmatrix}
CM_x \\
CM_y \\
CM_z
\end{bmatrix} = \begin{bmatrix}
0.0553 \\
0.0203 \\
0.0039
\end{bmatrix}
\]

The vehicle’s inertia matrix (in kg·m²)

\[
I = \begin{bmatrix}
9.93e^{-3} & -5.86e^{-5} & 5.69e^{-7} \\
-5.86e^{-5} & 2.229e^{-4} & -1.17e^{-5} \\
5.69e^{-7} & -1.17e^{-5} & 1.01e^{-2}
\end{bmatrix}
\]

This vehicle was test flown from the 5th floor of a parking garage and the video of the flights was analyzed to determine the flight performance. Sample screen shots from these videos are presented in figure 4.1.4.
After flying the vehicle, its dimensions and mass properties were entered into the steady state simulation presented in Chapter 3. The initial conditions entered included a yaw rate of 5 rad/sec (47 rpm), no initial descent rate, no initial coning angle (X rotation), and an initial pitch (Y rotation) of -0.1 rad (5.7 deg). The simulation was executed for 8 seconds. The resulting plots are presented in figure 4.1.5.

Four individual plots present each of the major components of the state vector. The vehicle’s position and velocity are presented in all three axes of the inertial frame. Examining these two plots, it can be seen that the vehicle begins to drop, accelerates to a maximum descent rate of 5.8 m/sec, then stabilizes to its autorotative descent rate of 3.1 m/sec. The CG of the vehicle oscillates laterally somewhat, but the vehicle does not translate significantly due to this oscillation. The next two plots present the vehicle’s angular position and angular velocity in the body frame. The angular position plot indicates a steady state coning angle (X angle) of -0.395 rad (-22.6 deg). Due to the sign conventions used, this negative angle corresponds to a cone-up orientation.

Figure 4.1.5: Free-flight simulation results with vehicle specifications and initial conditions presented in section 4.1.1
The results indicate a steady state pitch angle of -0.099 rad (-5.7 deg), and a steady state yaw rate (Z spin) of 18.8 rad/sec (180 rpm). The plot also indicates constant roll and pitch rates, which are consistent with the fact that the vehicle is spinning about the Z axis with constant non-zero pitch and roll. Table 4.1.1 presents a comparison between the experimental data and the simulation predictions for the first free-flight prototype.

<table>
<thead>
<tr>
<th></th>
<th>Experimental</th>
<th>Theoretical</th>
<th>Simulation Error (%)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Descent Rate (m/sec)</td>
<td>2.8</td>
<td>3.1</td>
<td>10.7</td>
</tr>
<tr>
<td>Rotation Rate (rpm)</td>
<td>163</td>
<td>180</td>
<td>10.4</td>
</tr>
<tr>
<td>Coning Angle (deg)</td>
<td>30</td>
<td>22.6</td>
<td>24.7</td>
</tr>
</tbody>
</table>

Table 4.1.1: Comparison of simulation results to flight data

The errors observed when comparing the simulation to flight tests were considered reasonable, and are most likely attributed to the lift and drag data used in the simulation. The baseline test results employ the mass distribution presented earlier. To verify the importance of the pitch moment modeling in the simulation, flight tests were performed while varying the chordwise CM location. The chordwise CM was varied from 0.036m (1.4 in) to 0.08 m (3.15 in) rearward of the leading edge. These two most extreme CM locations correspond to 24% and 53% of the chord respectively, and were at the limits of adjustability of the vehicle’s payload mounting bracket. During this CM variation, the observed descent rate varied from 2.77 m/sec to 3.66 m/sec, and the rotation rate varied from 225 rpm to 160 rpm, however, for all CM settings, stable autorotation was observed quickly and reliably. The simulation results did not yield the observed wide stability range when moving the CP. The simulation yielded a similar trend as the flight tests, but the stable range of CM locations was limited to 28–38% of the blade’s chord. In the simulation, the airfoil’s lift, drag and CP properties were represented as those of a flat plate, and it was hypothesized that the offset leading edge spar present in the first prototype created a considerable error between the observed and predicted pitch stability. To attempt to eliminate the discrepancy between the airfoil used and the airfoil represented in the simulation, a second prototype was built and employed a flat plate airfoil without an offset leading or trailing edge.

4.1.2 Second Iteration

The second free-flight prototype was designed with the intention of obtaining a better correlation with the simulation results. The main change from the first iteration was to implement a true flat plate airfoil, and to ensure chordwise symmetry. Rather than using a single offset tube at the leading edge and a “membrane” wing surface, the second vehicle employed a 3/16 inch Bakelite dowel for both the leading and trailing edges, and a 3/16 inch foam-board surface for the wing. This resulted in a true flat airfoil, with symmetrically rounded leading and trailing edges. Another feature intended to increase correlation with the simulation was the use of a rectangular planform. Similarly to the first vehicle, a payload mounting bracket
was employed to provide chordwise CM adjustment. The second prototype design is shown in figure 4.1.6. The resulting vehicle had a wingspan of 0.762 m (30 in) and a chord of 0.183 m (7.2 in) and a total mass of 0.32 kg (0.71 lb). Upon attempting to fly this vehicle, it was found that it would not enter steady autorotation. Rather, it would spin at a low rate with an extremely large coning angle and descent rate, exhibiting significant pitch instability. The CM was varied considerably, but no location yielded steady autorotation.

Upon investigating the remaining differences between the first and second prototype, it was found that although the CM of the two vehicle’s was the same, the chordwise mass distribution was not. The second prototype’s wing was structurally symmetric in the chordwise axis, hence the CM of the wing was located at 50% chord. Although the payload was positioned to give the vehicle an overall CM location of 33% chord, the spanwise mass moment of inertia vector was angled back towards the wing CM, as shown in figure 4.1.8a. As presented in Chapter 2, this vector can be thought of as the spanwise mass axis; the weighted average chordwise CM location along the span of the wing. On the first prototype, the heavy leading edge and light membrane yielded a wing CM of roughly 25% chord. This, coupled with a payload positioned slightly behind the vehicle’s overall CM, yielded a mass axis that was tilted forward towards the leading edge. It was hypothesized that the orientation of this mass axis may have been the cause for the differing stability observed between the first and second vehicle. To change the mass axis tilt from backwards to forward on the second vehicle, the trailing edge dowel was removed, and a strip of steel was added to the leading edge. The payload was then moved rearward so that the overall chordwise CM remained at 33% chord. The modifications were expressed in the solid model, and this model was then used to verify that the spanwise inertia vector was correctly oriented, as illustrated in figure 4.1.8b. The modified vehicle is shown in figure 4.1.7.

After modifications were complete, the vehicle was again flown. In all subsequent flights, it entered steady autorotation, and exhibited stability in both the pitch and roll axes, similarly to the first prototype. Progressing with the testing of the second prototype, it was found that the correlation with the simulation did not improve. Although the simulation predicted stability after the change in the mass axis orientation, it still did not provide as large of a range of stable chordwise CM locations. It was decided that the simulation provided enough fidelity to perform its intended task, and that work would progress towards the development of the controlled samara. To provide an explanation for the effect of the mass axis orientation, some time was devoted to analyzing this observed behavior. This analysis along with a proposed explanation is presented in Chapter 6.
Figure 4.1.6: Second free-flight samara model

Figure 4.1.7: Samara model with modified mass distribution

Figure 4.1.8: The incorrectly oriented mass axis (a), and the correctly oriented mass axis leaning towards the leading edge (b)
4.2 Controlled Simulation Results

After conducting the two iterations of flight testing and comparing the results to the free-flight simulation, the two controlled simulations were executed to see if reasonable results were produced. For the wing flap simulation, the same vehicle parameters were entered as in the free-flight simulation. In addition, the flap parameters used were as follows: flap chord was 33% of the wing chord, the flap position began at 70% of the span and ended at the wingtip, the flap deflection angle was 20 degrees, the actuation arc was 45 degrees, and the start time was three seconds, while the end time was 4 seconds. The simulation runtime was set for 8 seconds. The results of this simulation are presented in figure 4.2.1.

![Diagram showing controlled simulation results with graphs for CG Position, CG Velocity, Angular Position, and Angular Rates.](image)

Figure 4.2.1: Controlled simulation results utilizing the trailing edge wing flap. Vehicle specifications and initial conditions are the same as those used for the free-flight simulation results presented in section 4.1.1.
The simulation results show that prior to actuation at 3 seconds, the same steady state values are attained as in the free-flight simulation results. The actuation occurs for 1 second, resulting in the vehicle translating roughly 3.5 m (11.5 ft). During actuation, the descent rate increases to a maximum of 4.2 m/sec and the yaw rate of the vehicle decreases to a minimum of 12.75 rad/sec (122 RPM). Both of these effects were anticipated, and appear to be reasonable in magnitude. Additionally, during the actuation, the coning angle ranges from -1 to -0.142 rad (-57- 8 degrees). This indicates that the vehicle tilts to a considerable angle when translating under actuation. It can be observed that the system translates a great deal even after actuation has completed, and there is considerable lag in the overall system’s dynamics. Roughly 1.5 seconds after actuation is complete, the system stabilized back to its prior steady state autorotation conditions. The vehicle was not unstable during actuation, nor was it too stable to be effectively controlled. These results are highly favorable, and indicate a great potential to control the vehicle using a trailing edge wing flap.

Next, the wingtip rudder simulation was executed. Again, the same basic vehicle parameters as in the free-flight simulation were used. The wingtip was specified to protrude vertically from the end of the wing to a height of 0.2 m (7.87 in). The wingtip was the same chord as the wing, and used the same airfoil section. The rudder of the wingtip was specified to span the entire length of the wingtip and its chord was 33% of the wingtip’s chord. Preliminary simulation results showed that the wingtip and rudder actuation scheme provided less control authority than the wing flap method, so to counter this, a greater deflection angle was used, and the actuation occurred for 2 seconds rather than 1 second. The deflection angle used was 30 degrees, with a flap actuation arc of 90 degrees. Again, the vehicle was allowed to reach steady state autorotation, and 3 seconds into the simulation, the actuation was started. At 5 seconds the actuation was stopped, and the simulation was allowed to run for another 3 seconds. The results of this simulation run are presented in figure 4.2.2.

These results indicate that after two seconds of actuation, the vehicle translated 2.3 meters (7.5 feet). The results are similar to the wing flap actuation simulation, although clearly the rudder actuation provides less control authority. One important difference is that the coning angle during actuation is considerably less for the wingtip rudder method. During actuation, the coning angle ranges from -0.48 to -0.32 rad (27.5 to 18.3 degrees). This indicates that although the coning angle fluctuates during oscillation, it always stays above the horizontal, and the maximum change in coning angle was observed to be 9.2 degrees.

Both of the actuation methods were deemed to be feasible and favorable when judged solely by the simulation results. Because of the confidence inspired by the free-flight simulation results, it was decided that the controlled results would serve as a guide for developing the final controllable vehicle. The flight-testing of this final vehicle would serve as an opportunity to validate the controllable simulation variants. These results are presented in chapter 7.
Figure 4.2.2: Controlled simulation results utilizing the wingtip rudder. Vehicle specifications and initial conditions are the same as those used for the free-flight simulation results presented in section 4.1.1

### 4.3 Servo testing

In Chapter 2, the actuators considered for the guided samara design were presented. To determine whether actuation using off-the-shelf servos was feasible, it was necessary to conduct the following experiment.

Four basic requirements were developed and needed to be satisfied for control of the samara to be deemed feasible. First and foremost was actuation frequency. In a control scheme in which a flap is deflected during each revolution of the Samara, the actuator must be capable of operating at the vehicle’s frequency of rotation. The next constraint was for the servo to maintain this actuation frequency as its commanded deflection amplitude increased. Third, the servo had to be capable of actuating at the specified frequency and amplitude given the aerodynamic and inertial loads that would be present on the flap at a
particular RPM and flap deflection angle. The fourth constraint united the previous three, and was deemed fundamental to the successful controlled flight of the Samara. The servo had to be capable of actuating a flap at a particular frequency, to a given deflection, under the load placed on the flap, and perform this with a constant phase shift. This means that at any vehicle rpm, the time-delay from command to full deflection must remain constant regardless of the commanded angle.

An experiment was developed to test various off-the-shelf hobby servos and determine if they met these four criteria. The experiment consisted of placing a known torsional spring load on the servo output, then commanding the servo to actuate at a specified amplitude and frequency. In order to command the servos and monitor their output position, it was necessary to understand their operation on a fundamental level. Hobby servos are controlled using a Pulse Width Modulated (PWM) signal. This is a square wave with amplitude of 5 volts, and a variable duty cycle (pulse width). The standard that has been adopted by hobby servo manufacturers is this 5-volt PWM signal at a frequency ranging from 25 - 50 Hz. The pulse width corresponds to the commanded output position of the servo. A pulse width of 1.5 ms commands the servo to center its position, 0.5 ms commands full deflection in one direction, and 2.5 ms commands full deflection in the other. The servo unit has a controller that executes its actuation based on this command signal and feedback from a potentiometer that measures output position.

Each of the servos being tested was disassembled, and a lead was soldered to the potentiometer wiper. This provided an accurate way of measuring servo output. Two LabVIEW programs were created for this experimentation. One program generated a PWM signal and commanded the servo to oscillate at a specified amplitude and frequency. The second program acquired the analog output position signal from the potentiometer, thus allowing a comparison to be made between the commanded and actual servo positions. The servos where calibrated to relate exact output position to the command signal pulse width, and also to relate the potentiometer voltage to output position. Ideally, the PWM generation and the potentiometer signal acquisition would be performed by a single program. This would ensure that the timing between the commanded output and the measured output would remain in sync. The appropriate data acquisition card to perform both of these tasks simultaneously could not be obtained, so instead, two computers and two cards were used. One system consisted of a card that could support digital output, and this system executed the PWM program. The second system consisted of a card that could support analog input, and this system executed the potentiometer signal acquisition program. Both systems were kept in sync by using a digital trigger. The PWM program is shown in figure 4.3.3, and the potentiometer program is shown in figure 4.3.4. The PWM signal used for this test commanded the servos to move in a square wave. For the purpose of characterizing the servo’s response, it would be more appropriate to use a sine wave input, however for the purpose of achieving the highest actuation frequency with a slew-rate (maximum angular rate of the actuator’s output) limited actuator, a square wave is optimal. As an actuator attempts to follow a commanded square wave, its slew-rate limited output becomes a trapezoidal wave, and at its slew-rate dictated limits, becomes a triangle wave. It can be shown that for a particular maximum slope (slew-rate)
the frequency obtained by a triangle wave is 1.57 times the frequency of the sine wave with the same maximum slope.

For the testing of the actuator system, the design points for the actuator performance were extracted from the free-flight testing results as well as the controlled simulation results. The aerodynamic load expected to be experienced by the actuator was calculated. This calculation assumed a conservative rotational rate of 200 rpm, and a flap whose chord was 33% of the wing chord, beginning at 70% of the span and ending at the wingtip. The method for calculating the moment due to flap deflection was the same method described in chapter 3, used for modeling the effect of a flap for the controlled samara simulation. A Matlab script was created to perform a blade-element integration and compute the total moment generated by a flap at various deflection angles. Table 4.3.1 shows the results of this script.

<table>
<thead>
<tr>
<th>RPM</th>
<th>Deflection Angle (deg)</th>
<th>Moment (oz-in)</th>
</tr>
</thead>
<tbody>
<tr>
<td>100</td>
<td>15</td>
<td>0.0861</td>
</tr>
<tr>
<td>100</td>
<td>30</td>
<td>0.2596</td>
</tr>
<tr>
<td>100</td>
<td>45</td>
<td>0.5264</td>
</tr>
<tr>
<td>200</td>
<td>15</td>
<td>0.3444</td>
</tr>
<tr>
<td>200</td>
<td>30</td>
<td>1.0384</td>
</tr>
<tr>
<td>200</td>
<td>45</td>
<td>2.1058</td>
</tr>
</tbody>
</table>

Table 4.3.1: Calculated aerodynamic moments imparted on a flap as a function of rpm and deflection

Using these measurements as a guideline, three torque watches of varying torsional stiffness were affixed to the servos to provide displacement-based resistance. In addition to the resistance provided by the torsional spring, the watches imparted considerable inertial load due to their steel construction and large diameter. This inertial load included the chuck used to clamp the servo output to the watch, as well as the rotating mass within the instrument. The torque watch and its interface to a servo is shown in figure 4.3.1. Each of the four servos was tested for deflections ranging from 5 to 45 degrees, at frequencies ranging from 1-9 Hz, with 2.4, 20, and 40 oz-in torque watches attached. Admittedly, this test was more comprehensive than it needed to be. The main goal was to determine if the servo could operate with no time shift at a frequency of 3 Hz with the aerodynamic load corresponding to its amplitude. It was found that all of the servos tested could meet this performance requirement. The Hitec HS-50 performed better than the other servos at higher frequencies, and interestingly was also that lightest and smallest of the group.

Figure 4.3.2 presents test data for the Hitec servo. The plots shown represent two test points, taken at an oscillation rate of 3 Hz, at an amplitude of 30 degrees, both with load and no load. A 20 oz-in torque watch was used to apply load during the loaded test. The watch linearly applies torque, with roughly a 300 degree revolution yielding 20 oz-in. At 30 degrees of rotation, the torque watch applied 2 oz-in of torque to the servo. Figure 4.3.2 shows the time delay between the commanded deflection and the servo arriving at that deflection. The unloaded servo yielded a time delay of 0.139 seconds, while the loaded servo’s delay was 0.145 seconds. The six-millisecond difference in time delays is negligible. For a samara rotating at 180 rpm (3 Hz), this difference would result in a phase shift of 6.5 degrees, which is a tolerable heading error. This calculation is a worst-case time shift, in which the actuator is operating at two extremes; at zero load
(zero rpm) and at 2 oz-in of load (180 rpm). During actual flight, simulation results indicate that before, during, and after actuation, the largest variation in rpm will be roughly 60 rpm. This relatively small variation will produce a smaller change in aerodynamic loading, and hence a smaller heading phase shift than the one computed from the test results.

These servo test results indicated that an off-the-shelf servo, namely the Hitec HS-50, delivers adequate performance to be used as the actuator for an appropriately sized guided samara. These positive results, along with the servo’s light weight, cost, and ease of interfacing, dictated that it be chosen over the others.

![Torque watch mounted to the output of a servo](image)

Figure 4.3.1: Torque watch mounted to the output of a servo
Figure 4.3.2: Sample results of Hitec HS-50 servo tests. The upper plot illustrates the time delay for a 3 Hz oscillation at 30 degrees of deflection with no load. The lower plot illustrates the time delay for a 3 Hz oscillation at 30 degrees of deflection with 2 oz-in of load on the servo. The red line represents the commanded position, while the black line is the servo’s actual output position as measured from the potentiometer voltage.
Figure 4.3.3: User interface and the code of the LabVIEW PWM signal generation program written to control the servos during testing
Figure 4.3.4: User interface and the code of the LabVIEW potentiometer signal acquisition program written to record the servo’s output position during testing
### 4.4 Compass Testing

In Chapter 2 the various sensor choices were presented, as was the rationale for choosing to use a 2-axis electronic compass. The specific compass used was the Dinsmore model 1655. This unit consists of two orthogonal Hall effect sensors fixed to the compass housing. At its center, it contains a mechanical movement onto which a permanent magnet is affixed. This allows the magnet to rotate freely in the horizontal plane and always remain aligned with Earth’s magnetic field. As the compass housing is rotated in the horizontal plane, the magnet remains aligned with Earth’s magnetic field, and the Hall effect sensors rotate around the magnet. In this configuration, the Hall effect sensors measure in two axes their angle relative to the magnet, and hence, indirectly, their orientation relative to Earth’s magnetic field. An analog sensor was selected due to the increased flexibility over its digital counterparts. Digital sensors operate with a pre-determined angular resolution and sampling rate, however an analog sensor requires the user to perform the analog/digital conversion. This allows for both the angular resolution and the sampling rate of the compass to be varied within the noise limits of the analog signal. The figure below illustrates the output voltage of each of the two Hall effect sensors during one complete revolution, sampled every 20 degrees.

![Graph](image_url)

**Figure 4.4.1: Representative output of the two Hall effect sensors comprising the Dinsmore 1655 compass**

The output resembles a sine-cosine set of curves that center at approximately 2.5 volts, peak at 3.1 volts, and floor at roughly 1.9 volts. A single curve could be used alone to determine heading, although the relationship between voltage and heading is clearly non-linear. It can be noted from the figure that the curves cross at two points, approximately 2.9 and 2.1 volts. Between these crossing points, the two curves are steep and linear. To compute heading using only the linear portions of the curves, each revolution is segmented into 4 quadrants. In section 17 – 3 on figure 4.4.1, the B lead is linear, while the A lead is non-linear and greater than the upper crossing point voltage of 2.9 volts. In section 3 – 7, the A lead is linear,
while the B lead is non-linear and greater than the upper crossing point voltage of 2.9 volts. In section 7 – 12, the B lead is linear, while the A lead is non-linear and less than the lower crossing point voltage of 2.1 volts. In section 12 – 17, the A lead is linear, while the B lead is non-linear and less than the lower crossing point voltage of 2.1 volts. To obtain uniform resolution during heading measurement, the portion of the curves above and below the crossing voltages are used as indicators to determine which linear segment of the curve is used to compute heading in each of the four quadrants.

4.4.1 Compass Damping

When implementing the Dinsmore compass, a major concern was that the mechanical damping of the internal movement would cause a lag in its rotation when subjected to the rotation rates of the Samara’s flight. To verify that this would not be a problem, a variable rpm turntable was constructed. The device was constructed so that it could accommodate the compass and its supporting circuitry. The turntable was driven using an electric motor with a built in encoder and speed controller, and a computer based user interface allowed the motor’s position and velocity to be controlled. This turntable is shown in figure 4.4.2.

If the compass signal indicated it was within a pre-specified heading range, the microprocessor was programmed to either illuminate an LED on the microprocessor development board or to actuate a servo. Using this arrangement, the compass and electronics were rotated at rates of up to 200 rpm, and a video camera was used to record the turntable and observe if either the LED illumination or the servo actuation shifted in heading as the rotation rate increased. It was found that up to 200 rpm, there was no observable shift in the azimuthal angle of either indicator.
4.4.2 Tilt Error

Chapter 2 provided a comparison between 3-axis and 2-axis electronic compasses. Although a 3-axis compass provides compensation for tilt out of the horizontal plane, it cannot be used for the guided samara due to the constant rotation it would be subjected to. A 2-axis compass is not adversely affected by constant rotation, although it will exhibit a heading error associated with any tilt. The manufacturer of the Dinsmore compass states that “generally, tilt up to 12 degrees is acceptable with little error.” To examine if there were any potential methods to compensate for this error, a tilt error experiment was conducted.

The same setup was used as in the damping tests, although for the tilt error tests, the position control capability of the turntable was used. During a full rotation, the voltage of the two compass outputs was sampled 21 times, once every 18 degrees of revolution, with the first and last point being the same heading. A fixture holding the compass and providing tilt angle adjustability was machined and mounted to the turntable. Figure 4.4.3 shows the resulting plots with the compass at tilt angles of 0, 18, and 30 degrees. This plot is in the same form as that presented in figure 4.4.1.

The plots show that during tilting, there was a deformation of the signals from both leads of the compass. This deformation included a change in slope of the “linear” portions of the output curves, which was the direct cause of the increasing heading error. The “combined tilt results” plot was examined to look for a trend that could be used to compensate for tilt, however, no promising possibilities were observed. Although the tilt error was prevalent and undesirable, it was determined that the Dinsmore compass would still be used for the guided samara. Referring to the controlled simulation results presented earlier in this chapter, it was noted that during actuation using the wing flap actuation scheme there was a drastic tilting of the vehicle’s tip-path plane. However, the wingtip rudder actuation scheme produced substantially less tilting of the vehicle. During actuation, the maximum change in coning angle was predicted to be 9.2 degrees.

It was decided that the wingtip rudder scheme would be used on the controlled samara, and this would allow the Dinsmore compass to be retained. To minimize error due to tilt, the steady state coning angle and pitch angle of the vehicle would be determined prior to mounting the compass. The compass would then be mounted so that during steady state flight, it remained horizontal. This setup would result in no heading error at the start of actuation, and during the maximum tilt caused by actuation, the heading error would be limited to that associated with a tilt of only 9.2 degrees.
Figure 4.4.3: Results of compass tilt experiments. Plots represented outputs of the two hall effect sensors comprising the Dinsmore 1655 electronic compass.
Chapter 5

Final Vehicle Development & Flight Testing

5.1 Vehicle Design and Fabrication

All of the work presented up to this point has been focused on the task of acquiring the knowledge base necessary to develop a prototype guided samara that can be actively controlled in flight. The conceptual design process, simulation, servo testing, compass testing, and the free-flight testing all culminate in the design and fabrication of this vehicle.

5.1.1 Control System

As stated in Chapter 2, the control system of the Samara is considered to be the system of components required to actively change the vehicle’s orientation and position. The end goal of this research was to demonstrate the ability to control the omni-directional glide slope of an autorotating samara. To accomplish this goal, a control system with the following components was designed;

![Flowchart of the guided samara control system](image)

In this system, the sensor is used to determine the azimuthal orientation of the body frame with respect to the inertial frame. The transmitter/receiver pair is not required, but when it is desired to remotely control the vehicle, these two components are used to deliver commands to the vehicle’s microprocessor. On board
the vehicle, the software running on the microprocessor is used to process the sensor data and the command signal from the receiver. Based on these two inputs, the microprocessor then commands the actuator to deflect the control surface and create the desired vehicle motion.

**Electronic Compass**

The first component selected for the final control system design was the azimuthal angle sensor. In Chapter 2 the various sensor choices were presented, as was the rationale for choosing to use a 2-axis electronic compass. The specific compass used was the Dinsmore model 1655. To enable a digital microprocessor to interpret the compass signal, an analog to digital converter (A/D) must be used. The primary consideration in selecting an appropriate A/D was the desired resolution in the heading measurement. A 12-bit converter has 4096 counts per 5 volts, yielding a resolution of 0.0012207 volts per bit. The linear portion of the compass signal ranges from 2.1 to 2.9 volts. The upper reading of 2.9 volts translates to an A/D output of 2376, and the lower reading of 2.1 volts translates to 1720. This yields a range of $2376 - 1720 = 656$ counts per 90 degrees, which results in a final angular resolution of 0.137 degrees. This was more than adequate for the application at hand, so the MCP3202-B 12-bit A/D was selected.

To avoid introducing noise, it was desirable to position the A/D as close as possible to the component providing its input. When developing the circuit board containing the components for the control system, it was decided to mount the electronic compass and A/D together on a stand-alone circuit board. This would allow for the A/D to be located close to the compass, while still permitting flexibility in positioning and orienting the compass relative to other components on the vehicle. The online company “RC Instruments” supplies a small circuit board specifically for this purpose. In addition to accommodating the Dinsmore compass and an 8-pin PDIP A/D, it also contains room for a bypass capacitor, used to smooth the input voltage to the components on the board. This circuit board was used to house the compass components. A schematic as well as a picture of the assembled board with all components installed is shown in figure 5.1.2.

![Figure 5.1.2: The compass circuit schematic and the completed circuit board](image-url)
Microprocessor

The computational requirements for the control system were relatively low, the only requirement being a high enough processing speed to ensure that compass heading was sampled at a sufficient rate. Based on the free-flight testing and simulation results, the expected rotational rate of the guided version of the samara was expected to fall in the range of 150 – 180 RPM (2.5 – 3 Hz). To ensure sufficient resolution and response time it was desired that the control system be able to sample and process the compass signal at a minimum of 10-degree intervals. At a revolution rate of 3 Hz, this corresponds to a sampling rate of 108 Hz. Based on the availability of the hardware, documentation, and support resources, it was decided to use a Microchip brand PIC microprocessor. A popular multipurpose chip, the PIC16F628A, was selected. This chip has a variable processor speed of 4 – 20 Mhz. Running at 4 Mhz, the processor is capable of performing 4 million operations/second, so at this speed it could run a loop containing 37,000 operations and still meet the minimum sampling rate requirement of 108 Hz. The program necessary to process the compass signal and command an actuator would contain far less than 37,000 operations, so it was determined that this processor would be adequate. The PIC16F628A was used in its PDIP package configuration, containing 18 pins, 16 of which were available to use as either inputs or outputs.

Servo and Receiver

In chapter 4, the servo actuator testing and the final selection of the servo were presented. It was determined through this testing that an off-the-shelf hobby servo would have sufficient performance to periodically actuate a control surface of the guided Samara. The servo chosen for the task was the Hitec HS-50 Feather Servo. This unit weighs only 6.1g (0.21 oz), and has a maximum slew rate of 0.09 sec/60 degrees.

The guided samara control system was designed to permit the optional remote control of the vehicle. To facilitate this, a micro receiver was integrated into the electronics package, and two channels of output (Pitch and Roll) were routed from the receiver to two input pins of the microprocessor. The primary consideration for the selection of a receiver was light weight, and compatibility with the selected servo. Based on these criteria, the FMA M5v2 sub-micro receiver was selected. This model weighs 9g (0.3 oz) and has five useable channels.

Power Source

Due to the short flight times (less than 30 seconds), high energy-density was not a large concern in the design of the guided Samara. However, it is always desirable to minimize weight, so a battery pack was designed to be only large enough to provide a very conservative 10 minutes of power to all onboard electronics. Table 5.1.1 shows all of the vehicle’s onboard electronic components, and their current draw. It was assumed for the current draw calculation that the components would be operating at their maximum draw, i.e., the servo operating at stall torque, and the microprocessor operating at a clock speed of 20 Mhz.
Using the total current of 523.4 mA, and designing for a 10 minute battery life, the required capacity was computed to be 87 milliamp-hours (mAh). All of the components in the control system require 5 volt power, with a margin of roughly 1 volt. Because the 10 minute battery life calculation did not take into account voltage drop as the battery discharges, the calculated capacity was roughly doubled.

![Image of Hitec HS-50 servo and FMA M5v2 receiver](image)

*Figure 5.1.3: Hitec HS-50 servo and FMA M5v2 receiver*

<table>
<thead>
<tr>
<th>Current Draw [mA]</th>
<th>Voltage</th>
</tr>
</thead>
<tbody>
<tr>
<td>Hitec HS-50 Servo (No load running)</td>
<td>100</td>
</tr>
<tr>
<td>Hitec HS-50 Servo (Stall)</td>
<td>500</td>
</tr>
<tr>
<td>Dinsmore R-1655 Compass</td>
<td>19</td>
</tr>
<tr>
<td>PIC 16F628A Microprocessor</td>
<td>1</td>
</tr>
<tr>
<td>FMA M5v2 Receiver</td>
<td>3</td>
</tr>
<tr>
<td>MCP3202B A/D Converter</td>
<td>0.375</td>
</tr>
<tr>
<td><strong>Total (Assuming Stall Torque)</strong></td>
<td><strong>523.4</strong></td>
</tr>
</tbody>
</table>

*Table 5.1.1: Current draw of control system components*

Once the capacity was determined, the type of battery technology could be selected. The main factors in selecting a battery type were the standard cell voltage and the specific power output. Of the three most popular types of cells available on the hobby market, Lithium Polymer provide the highest specific power output, followed by Nickel Metal Hydride, and finally Nickel Cadmium. The standard Lithium Polymer cell is 3.7 volts, so to provide 5 volts to the control system circuitry, a two cell pack require a voltage regulator. For the sake of simplicity, a custom 4 cell pack was constructed from 160 mAh, 1.2V Nickel Metal Hydride cells. This yielded a pack with 160 mAh capacity at 4.8 volts, and a total weight of 18.1 g (0.64 oz).
Electronics Integration

To complete the control system, the compass, receiver, servo, battery, and microprocessor had to be integrated into one circuit that could be effectively mounted on the guided samara prototype. The first part of this integration was the design and fabrication of a circuit board to house the microprocessor, its resonator, and the header pins that would allow the other components to be connected to the microprocessor’s I/O pins. It was initially intended to power the circuit board, receiver and servo with a single battery, however during the debug period a problem arose when a servo was powered via the circuit. When monitoring the servo command signal generated by the microprocessor, it was observed that as soon as the servo began to move, the command signal became distorted. Switching the power source from a battery to a regulated power supply eliminated this problem. It was determined that the momentary load placed on the battery by the servo’s high gain controller caused a fluctuation in the voltage. As shown in figure 5.1.5, the voltage requirements of the microprocessor are more stringent when it is operated at 20 Mhz, and therefore, the voltage drop due to the servo’s actuation was enough to affect the processor’s operation. This behavior persisted even when a voltage regulator was used. As a quick fix to this problem, two batteries were used, one for the servo and receiver, and one for all other components.

The circuit diagram for the complete control system is shown in figure 5.1.7. The two batteries are not illustrated; only each component’s connection to ground and positive supply voltage is shown. Because two batteries were used, it was necessary for proper operation of the servo to ensure that all grounds were made common. An optional practice when working with microprocessors is to tie all unused pins to ground. This along with the common grounding is not shown in figure #. In addition, two switch harnesses were added between the batteries and their connections to the circuit and receiver. Rather than fabricating a custom printed circuit board, a generic prototype breadboard made to house an 18-pin PIC was modified and used. Header pins were added to the board and wire wrapped to interface the batteries and compass circuit to the microprocessor. The complete circuit board with the microprocessor, resonator, and header pins is shown in figure 5.1.6.
Figure 5.1.5: PIC16F628A voltage requirements as a function of frequency (from user manual)

Figure 5.1.6: Control system circuit board
Figure 5.1.7: Control system circuit schematic
Control Software

MicroEngineering Labs’ PicBasic compiler was used to program the PIC microprocessor. BASIC was chosen as the programming language due to its simplicity and fast development time compared to programming using Assembly. A major drawback of the PicBasic compiler is its inability to handle floating-point math. To work around this limitation, all non-integers had to be multiplied by a scaling factor to convert them to integers, the mathematical operation was performed, and then the result was again scaled if necessary. This process was complicated by the limitations imposed by the microprocessor’s architecture. The PIC16F628A processor used in this project is an 8-bit processing unit, but uses two 8-bit registers to effectively represent values with 16-bits of precision. This 16-bit precision means that the largest number that can be represented is 65,535, so the scaling of non-integers had to be performed creatively to prevent overflow.

Two main programs were created to control the guided samara. The first program requires the user to specify a desired heading in which the vehicle will fly, the angle over which the servo will actuate, and the flap’s deflection angle. The user also specifies a delay time before the vehicle is commanded to fly in the specified heading, and a duration for the controlled portion of the flight. This program will be referred to as the “step response” program. A complete flight would occur in the following sequence: The control system is turned on and the vehicle is released. The vehicle enters steady autorotation descending vertically. Once the delay time has elapsed, the servo begins periodically actuating over the specified angle during every revolution, causing the vehicle to move laterally in the commanded direction. Once the duration of the controlled flight has elapsed, the servo stops actuating, and the vehicle once again settles into steady autorotation descending vertically. Figure 5.1.8 presents a high level flowchart of the step response program.

The second program written allows for a user to pilot the guided Samara using a remote control transmitter. The user commands the samara to fly in a specific heading at a specific rate using a single stick for input. In figure 5.1.9, the shaded arcs represents the range over which the flap on the samara is actuated during one revolution. The angle of the transmitter stick deflection determines the average angle of the actuation arc, and the amplitude of the transmitter stick determines the magnitude of the angle the arc subtends. All heading inputs are relative to a pre-specified reference heading. This second program, referred to as the remote control program, uses the same basic structure as the step response program, with several additions. Rather than the user specifying desired heading and the angle that the actuation arc subtends, these parameters are determined via the two channels (pitch and roll) sent to the microprocessor from the transmitter via the receiver. Because they drive hobby servos, standard hobby receivers output a PWM signal for each channel. This signal’s duty cycle varies from 1 ms to 2 ms. As an example, when the stick of the transmitter is centered, both corresponding channels of the receiver output a signal with a 1.5 ms duty cycle. When the stick is pushed all the way to the right, but remains vertically centered, the channel corresponding to lateral stick movement outputs a 2 ms square wave, while the other channel stays at 1.5 ms. In this manner, by measuring the duty cycles of the two receiver channels, the angle of the stick
and the magnitude of the sticks deflection can be computed and then processed into the desired vehicle heading and the actuation arc angle.

Figure 5.1.8: Step response control program flowchart

Figure 5.1.9: Scheme used to relate transmitter stick deflection to actuation arc size and orientation
5.1.2 Airframe

The primary consideration when designing the guided samara was the development of the control system. The focus of this project was not to develop a vehicle with optimum aerodynamic performance, but rather to demonstrate the ability to control such a vehicle in flight. Therefore, the goal in fabricating the airframe was to provide a platform for the control system that meets the following criteria:

- **Robustness** - The airframe must be resistant to impacts and withstand repeated flight tests.
- **Minimal rotation rate** – While keeping the vehicle manageable in size, it is desirable to minimize the steady state rotation rate. This lowers the rate at which the compass and servo actuator must operate and reduces the potential for heading errors due to phase lag.
- **Simple modeling** – It is desirable to simplify the planform and airfoil in order to simplify the modeling of the vehicle as much as possible. This assures the best correlation between flight test results and the dynamic simulation results.
- **Electronics packaging** – The design of the airframe must cleanly accommodate the control system’s components.

**Design**

In order to minimize weight and to facilitate transport and handling, the wingspan was limited to three feet. This considerable increase in length over the free-flight Samara models was anticipated to decrease rotation rate by 20-30 RPM. A rectangular planform with no taper or wing twist was selected to simplify the modeling correlation. A fully symmetric airfoil section was chosen. In order to provide enough depth to embed the electronics within the wing given the chosen chord, a NACA 0012 section was used. With a 7 inch chord, this airfoil yielded a 0.84 inch thick section. Several factors influenced the type of control scheme used to control the vehicle. Simulation results presented in Chapter 4 show that using a control surface on the trailing edge of the wing to change the lift caused the vehicle to tilt considerably during actuation of the flap. The testing of the electronic compass, also presented in Chapter 4, revealed no evident method of compensating for tilt error. This limitation of the compass dictated that the lateral thrust generation control scheme be used to limit change in coning angle during actuation. Figure 5.1.10 shows the solid model of the final guided samara design. This solid model was primarily used as a tool for verifying the location of the vehicle’s CM and the orientation of the axes of inertia. Both of these mass properties have a great effect of the start-up and autorotation characteristics of the samara. The design allows flexibility in the placement of the wingtip. It can be vertically centered as shown in the figure, or protrude upwards or downwards from the end of the wing. Initially the intent was to design the wingtip so its angle of attack was zero to ensure no lift was generated without a flap deflection. The radius of curvature of the tip-path is approximately equal to the span of the wing, and to ensure zero effective angle of attack the curvature of the wingtip would have to match this. However, with only a 7 inch chord, this
curvature was so negligible that it was assumed it could be ignored. A fully symmetric NACA 0010 section was used for the wingtip. The thinner airfoil was chosen to minimize drag, since the tip sees the highest velocity, and also to minimize weight at the tip, which keeps the CG and thus center of rotation closer to the root.

![Figure 5.1.10: Model of the guided samara design](image)

The size of the wingtip and flap were chosen using the dynamic simulation. A 6 inch wingtip with a full-length flap with a chord of 2.5 inch was selected to provide sufficient control authority. As shown in the model, a pocket was created in the wing to house the electronics, while the compass circuit was mounted outside of this pocket near the center of rotation. In addition to the electronics being mounted as far towards the leading edge as possible, a lead mass was formed to the leading edge and was mounted as needed to adjust the mass properties of the vehicle. An aluminum bracket was designed to offer adjustable positioning of the main payload. A 10 inch long, 0.25 inch diameter carbon rod embedded in the wing provided the mounting point for the payload bracket.

**Fabrication**

The airframe was fabricated using a fiberglass laminated foam core technique. This is a common construction method used for both model and full-scale aircraft, and has the potential to yield an extremely light and stiff wing. The role of the core is to provide compression resistance and to maintain the airfoil’s shape, while the composite skin provides both bending and torsional stiffness. Varying density and strength of foam may be used, and the composite skin can be laid in varying thicknesses and orientations to provide increased stiffness or puncture resistance to any particular area of a structure.
The wing core consisted of 1.8 lb/ft³ pink foam. This foam is generally used in building construction, but is well suited to model aircraft fabrication due to its availability, strength, and ability to be cut by hot wire. A computer controlled hot wire cutter was used to cut the cores to a NACA 0012 cross section. The foam core was laminated using 2 oz/yd² fiberglass cloth and West Systems 105 epoxy resin. Several foam cores were fabricated using both wet lay-up and vacuum-bag techniques. It was found that the best surface finish was obtained by vacuum bagging the wing while incorporating a Mylar sheet over the wetted glass. The steps in vacuum bagging the wing are presented below:

- In order to reinforce the vulnerable leading edge of the wing and protect it against puncture and deformation, a two-inch strip of double layerd glass was wetted with epoxy and placed on the leading edge.
- Two sheets of Mylar were cut to fit the top surface of the wing. They were then taped together lengthwise on one side.
- A single sheet of fiberglass large enough to cover both sides of the wing was then placed on the two sheets of Mylar and thoroughly wetted.
- The trailing edge of the wing was placed on the tape at the seam of the two pieces of Mylar. The Mylar was then folded onto the surface of the wing, starting from the seam at the trailing edge. The two Mylar sheets were cut to fit the surface of the wing, and hence almost met at the leading edge.
- The wing was then placed back into the foam cradle from which it was hot-wire cut. This practice provides additional support during the curing process to ensure the wing cures straight.
- The foam cradle was covered in breather cloth to ensure all air was drawn from the vacuum bag.
- The foam cradle was inserted into a vacuum bag, the bag was sealed, and it was ensured that there were no leaks. The wing was allowed to cure for 24 hours under vacuum.
- The Mylar was removed, the trailing edge was trimmed, and the leading edge was sanded to remove any ridges of epoxy.

The same procedure was used to fabricate the wingtip, although blue foam was used rather than pink. The blue foam, although roughly the same density, has double the compression strength of the pink. It was not available in large amounts, so the entire wing could not be made of it, but it was used for the wingtip where increased durability was anticipated to be required. Two threaded balsa wood blocks were embedded in the tip of the wing, and the wingtip was attached using these blocks and nylon screws. Nylon screws were used so that in case of an impact the screw heads would shear and leave the wingtip intact. A control surface was created by cutting through one side of the wingtip’s fiberglass skin and foam, but leaving the fiberglass on the opposite side intact to act as the hinge. Using 3M 77 adhesive, fiberglass tape was then added over the hinge as extra reinforcement. A servo was embedded in the wing near the wingtip. This facilitated the use of a short control arm to connect to the control surface of the wingtip, and also
granted the extra flexibility of allowing the servo to be used to actuate a control surface on the wing’s trailing edge if desired. The completed wingtip assembly is shown in figure 5.1.11.

![Figure 5.1.11: Wingtip assembly of the guided samara](image)

A hole was drilled into the foam core to accommodate the carbon spar, which was epoxied in place. An aluminum bracket to mount the payload to the carbon spar was machined. This bracket was designed to allow the payload’s position to be shifted both in the vertical and in the chord-wise direction. This flexibility was required in order to allow adjustment of the vehicle’s CM location and mass axis orientation. A pocket was cut into the wing near the leading edge to accommodate the control system electronics. The placement of all electronics close to the leading edge was chosen due to the thickness of the cross section at this point, but also to shift the span-wise mass axis forward. Due to the error associated with tilting the compass from horizontal, the compass was mounted on an angled aluminum block. One face of this block was machined to an angle of 20 degrees, roughly the coning angle anticipated for the guided samara during steady state autorotation. This mounting orientation was an attempt to ensure that the compass remained as close as possible to horizontal during the flight regime when it would be used to actuate the servo. Figure 5.1.12 presents several views of the completed vehicle, including the payload and mounting bracket, the embedded electronics, and the mounted compass circuit.

During flight tests, the electronics bay and the compass circuit were protected by energy absorbing foam. Although not shown in the figure, a sheet of foam was cut to fit over the electronics, and a thin sheet of Kevlar held the foam flush to the wing’s top surface. Both the foam and the Kevlar had two cutouts through which the switches for the two switch harnesses protruded, allowing the vehicle to be turned on or off without their removal. A small foam hump was shaped and mounted over the compass to provide protection in case of impact.
5.2 Vehicle Flight Testing

All of the samara free-flight testing performed during the course of this research was conducted from the tallest structures the author could gain access to around the MIT campus. These short flights, usually conducted from less than 60 feet, were sufficient for the purpose of demonstrating free-flight autorotation or to collect data for simulation validation. To effectively demonstrate omni-directional glide slope control of the guided samara, it was desirable to have a much taller and safer vantage point from which to conduct flight tests. Dropping a samara from a building presents the following problems:

- The vehicle must be thrown away from the building to avoid hitting the side. This generates a lateral velocity away from the building as the vehicle descends.
- There are safety concerns such as hitting bystanders, cars and other buildings.
- It is not possible to video tape the vehicle’s flight from directly above
- After roughly 20 feet of descent needed to enter autorotation, the remaining flight lasts only 2 – 4 seconds, depending on descent rate and initial drop height.
A better test location would allow the vehicle sufficient flight time to use the step response control program. Running this program when dropped with sufficient altitude, the vehicle would enter autorotation, descend vertically in stable flight for several seconds, the control system would then cause a transient into several seconds of lateral translation, after which the vehicle would again enter pure vertical descent. With sufficient altitude this sequence of events would provide clear evidence of the ability to decisively control the omni-directional glide slope of the samara. If video taped from above and from a lateral angle, flight data could be extracted and used to validate the controlled simulation results.

5.2.1 Flight Test Facility and Procedure

A location deemed ideal for the guided Samara flight-testing was NASA Langley’s Impact Dynamics Research Facility in Hampton, Virginia. This facility comprises of a gantry structure from which full scale aircraft are crash tested. The gantry’s top level reaches a height of 240 feet, while an intermediate level resides at 150 feet, and a 400 by 280 foot concrete slab provides an unobstructed area for aircraft to impact. The facility also features a large backdrop and cameras used for motion tracking. This provides an accurate method of acquiring position and velocity data during a test. In addition, the facility’s control tower actively monitors wind speed and direction, allowing the timing of test flights to be coordinated with optimum wind conditions.

![NASA Langley’s Impact Dynamics Research Facility](image)

Figure 5.2.1: NASA Langley’s Impact Dynamics Research Facility

With the help of contacts acquired as a student researcher at the Impact Dynamics Research Facility, the author wrote and submitted a proposal to NASA, requesting the use of the facility in support of the guided samara flight-testing. The proposal outlined the test procedure, presented simulation results to support the expected in-flight behavior of the guided samara, and addressed safety concerns. The proposal was accepted, and the flight-testing was planned for five days in June. Due to the ambiguity regarding the vehicle’s response to control inputs during its maiden flights, it was deemed necessary to conduct the tests
during periods of very low wind. The five day window for demonstrating controlled flight was scheduled to accommodate this need for good weather.

The abbreviated test procedure consisted of a video camera mounted at the top of the gantry pointing downwards. A high speed camera on the ground was pointed horizontally at the backdrop. The parameters entered into the step response program were an initial pause of 8 seconds, and then steady actuation in the longitudinal axis of the gantry (East-West) until the vehicle impacted the ground. The goal was to capture with both cameras the transition from vertical descent to lateral translation, and to have the vehicle translate across the frame until impact. Once control was demonstrated using the step response program, it was planned to use the remote control program to attempt to demonstrate more complex maneuvering.

5.2.2 Flight Test Results

In anticipation of potential hard landings/crashes, two copies of the guided Samara airframe were fabricated and taken to NASA for the week of flight-testing. One vehicle was outfitted with a control system and one was fabricated as a flying airframe lacking any electronics. This “dummy” airframe was constructed for the purpose of verifying that the mass distribution and aerodynamics of the guided Samaras yielded reliable entrance into autorotation and stable steady state descent. In addition to these two airframes, several spare wingtips varying in length from 6 inches to 10 inches were also packed. Prior to leaving for NASA, a free flight samara using the same airfoil section as the guided Samara with a 24 inch wingspan and a 6 inch wingtip was flown to ensure the wingtip did not create unforeseen problems. With the particular configuration used, the vehicle entered autorotation and flew reliably.

A sequence of test flights leading up to a controlled test was planned. Initially, several tests using the dummy airframe would be conducted to ensure that the mass properties were correct and that the wingtip configuration allowed for stable flight. Then the controlled samara with electronics would be flown in a passive, free-flight state (no actuation). Finally, once it was ensured that this vehicle achieved free-flight without any problems, the controlled flights would be attempted. The events and results of the five days at NASA are presented in the following section.

Day 1

The facility was not available for use. Instead, the 12 meter (39.4 ft) tall photography backdrop was used to conduct preliminary free-flight tests of the dummy Samara. During this set of flights, a 6 inch wingtip was mounted protruding upwards from the wing. Prior to the initial flight, the mass properties were not verified using the computer solid model. The payload and lead mass were positioned so that the chord-wise CM was at roughly 33% chord, the vertical CM was just below the wing centerline, and it was estimated that the mass axis was tilted forward slightly. In this configuration, the vehicle did not autorotate reliably. It was
determined that there were two reasons for this. The mass axis was not angled forward, and there was not enou... edge and forward swept trailing edge are shown in figure 5.2.3. Figure 5.2.3: Two wingtip configurations used

Day 2

The dummy samara was flown from the 150 foot level of the gantry. Again the 6 inch wingtip was mounted protruding upwards, the CM was at 33% chord, the vertical CM was just below the wing centerline, and the mass axis was angled forward. In this configuration the vehicle again did not autorotate reliably. The wingtip was then centered vertically rather than protruding upwards. In this configuration the vehicle entered autorotation reliably and flew in a stable manner. Figure 5.2.4 shows the samara in descent from the 150 foot level of the gantry. After it was verified that the samara with a centered wingtip flew well, two other wingtip configurations were tried. A wingtip with backward swept leading edge and forward swept trailing edge are shown in figure 5.2.3.

Figure 5.2.3: Two wingtip configurations used
Flight tests revealed that the vehicle with the backward swept wingtip entered autorotation well, but as the RPM increased, the coning angle did as well. This increase in coning angle caused the samara to fall out of autorotation. The forward swept wingtip did not exhibit this behavior. Brief analysis of the behavior is presented in Chapter 6 along with an explanation of how wingtip shape and location affect flight performance. The result of the second day was that a stable configuration for the dummy vehicle was found, and it was decided that on day 3 the similarly configured guided samara would be flown in free flight and then in a controlled attempt.

![Image](image.png)

**Figure 5.2.4: The dummy samara descending from the gantry’s 150 foot level**

**Day 3**

A northeast wind was blowing with gusts over 20 mph. This persisted throughout the day, so no flights could be conducted. Flights were planned for early the next morning.

**Day 4**

Again, the wind was gusting at over 20 mph. The weather forecast was poor for the rest of the day and through the final day of testing. With the window of opportunity to demonstrate controlled flight closing, it was decided that other options should be considered. With the weather hampering outdoor flights, the facility coordinator of the NASA Langley aircraft hanger was contacted. Because no large aircraft were in the hanger at the time, the coordinator kindly allowed the author to conduct flight tests from the catwalk in the hanger’s roof structure. With a covered area of 99,000 square feet and a maximum height of 80 feet, the hanger seemed like an ideal location to perform short flight tests without interference from the elements.
In the hanger, a camera was affixed to the catwalk pointing down at the drop location. In addition, two high-speed cameras were set up to capture lateral views of the samara as it descended. One camera was set up with a wide angle lens to capture a greater field of view, and the other had a lens with a longer focal length.

![Image of hanger](image.jpg)

Figure 5.2.5: The location in the hanger from which flights were conducted

It was decided to first drop the dummy samara without a wingtip. The purpose of this flight was to determine with the simplest configuration if the drop height was enough to ensure that steady autorotation was achieved, and to make sure there was no chance of striking one of the six NASA aircraft in the hanger at the time. The dummy samara was dropped wingtip first. This drop orientation generates rotation as the vehicle attempts to fall mass-first. In previous flight tests this rotation has been observed to accelerate the entrance into autorotation. The first flight was successful, although it took roughly two-thirds of the drop height for the vehicle to enter autorotation. Next the guided samara with electronics and wingtip was tested in an unguided free-flight. For this flight, the vehicle was configured with an 8 inch centered wingtip, the same mass properties as the successful flights from day 2, and a foam hump was placed over the compass. The vehicle was dropped wingtip first, but did not enter autorotation. The factors that may have adversely affected the flight were the use of the larger wingtip, and the foam hump over the compass. It had been assumed that because the angle off attack near the root was very large, and the tangential velocity component very low, that the addition of this hump would not affect the flight performance a great deal.

To test which of these two factors was causing the poor flight performance, the dummy samara was outfitted with the 8 inch wingtip and the compass foam was left off. In this configuration the vehicle was dropped wingtip first, and again, did not enter autorotation. During this attempt, it was noted that the vehicle did not rotate into a mass first descent and then smoothly continue this motion into steady autorotation. Instead it seemed to rotate into a mass first orientation, fall for some time, then enter autorotation. Based on this observation, a flight attempt with the same vehicle was made, this time dropping it mass first. The result of this flight was that it fell straight down and impacted without rotating at
all. This damaged the main spar and the session was cut short in order to make repairs. It was hypothesized that the extra drag created by the wingtip caused this observed resistance to entering autorotation from the mass-first orientation.

After making repairs, the dummy samara was outfitted with a smaller, 6 inch centered wingtip, and was dropped wingtip first. The vehicle entered autorotation, but again, took nearly two thirds of the drop height to autorotate. This left only 25 feet of autorotation with which to demonstrate controlled flight. To kick-start the autorotation, a new drop technique was attempted. As shown in figure 5.2.6, the samara was dropped wingtip first, but while being released was rotated in a leading-edge-first direction. This resulted in autorotation after only falling roughly one third of the total drop height. This was a great improvement, and the technique was used for all subsequent flights.

![Diagram](image)

Figure 5.2.6: The moment imparted on the samara when “kick-starting” its autorotation (a), and the resulting rotation and descent (b)

After it was determined that reliable autorotation could be achieved in the hanger using a smaller wingtip and the kick-start technique, the compass foam was added to the dummy samara to see if this negatively affected its flight. With the addition of the compass foam, the vehicle entered autorotation well, although it was observed that near the end of its flight, it began to wobble and its rotation became unstable. The compass foam was removed and another flight was attempted with the 6 inch centered. During this flight the vehicle was kick-started and it entered autorotation quickly and flew in a stable manner. Although the observations about the compass foam were far from conclusive, to eliminate possible problems the foam was eliminated for future attempts. To facilitate removal of the padding, compass was embedded deep into the wing. This modification was performed and controlled flights were planned for the next day.
**Day 5**

Due to acceptable wind level in the morning, a controlled flight was attempted from the gantry. The first flight was performed from the 240 foot level with no control actuation. If this flight proceeded without problem a controlled flight would be attempted next. The vehicle was configured with all of the lessons learned from the prior days; a centered 6 inch wingtip, no compass foam, and when dropped, it was kickstarted. The samara entered autorotation quickly, but during its descent it fell out of autorotation three times, each time falling some distance and then re-entering autorotation. It was initially assumed that perhaps wind gusts had caused the departure from steady flight. After further consideration, it was noted that the instability occurred in a periodic manner. The vehicle descended the same amount each time before falling out of autorotation. This behavior indicated a possible instability at a particular rpm. A proposed explanation for this is presented in Chapter 6.

It was noted that due to the shorter drop height in the hanger, the vehicle did not descend far enough to begin the periodic instability observed when dropping from 240 feet. This indicated that hanger flights could be continued without modifications to mitigate the problem. Additionally, it began to rain after the initial outdoor flight, so it was decided to continue attempts indoors.

At the hanger, an attempt was made at a controlled flight. The step response program was set up to actuate over a 90 degree azimuthal angle. The heading in which the vehicle was programmed to translate was perpendicular to the direction in which the high-speed cameras were pointing. In order to allow the samara to enter autorotation and stabilize, control actuation was delayed until 2 seconds after the vehicle was dropped. During the first attempt, the vehicle autorotated well, and descended in a stable manner, however no lateral translation was observed. When the vehicle impacted and stopped rotating, the azimuthal angle of the wing was within the 90 degree actuation arc, and the flap was actuated. The vehicle was picked up and rotated, and the control system still functioned and actuated the flap within the correct arc. It was not clear if the flap had been actuating during flight. The high-speed videos were inspected, although the detail of the wide angle camera was not enough to make a determination, and the vehicle had fallen just out of the frame of the camera with the telephoto lens. The camera providing the top view of the flight did not have high enough resolution discern if the flap had been actuating during the flight. It was decided to repeat this flight with all parameters the same. The high-speed camera was repositioned so that the landing spot from the previous flight was within the frame.

During the second controlled flight attempt, the author incorrectly dropped the vehicle, and as a result, it took longer to autorotate and impacted at a high rate, breaking the carbon spar that held the payload. To fix this, the spar was cut flush to the wing root, and a 1/8” aluminum plate was cut to match the shape of the airfoil and fiberglassed to the wing root. This plate was then drilled and threaded at the appropriate chord-wise location of the payload, and the payload was bolted directly flush to the plate. This quick fix would ensure that the moment generated by a hard impact would not be taken solely by a spar but would rather be transferred via the plate to the wing. During the course of the repairs, the electronics were tested and it was observed that the servo actuation was behaving erratically. A broken connector was found
on the main circuit board. This broken connector was causing intermittent operation, and may have been the cause of the failed first controlled flight attempt.

The author returned to the hanger in the afternoon with the repaired vehicle, and attempted to perform another controlled flight test. It was found that with the repaired vehicle did not autorotate reliably. Due to the necessary repairs to the payload mounting, the mass properties of the vehicle were modified, and there was insufficient time to determine the correct mass properties with the solid model. After several more failed attempts to fly the repaired vehicle, time was up and the testing had to be called off.

### 5.2.3 Flight Test Conclusions

The controlled flight tests were inconclusive. Many factors combined to provide difficulty in demonstrating controlled flight. These included poor weather, instability of the airframe design, and damage to the control system electronics. Several valuable lessons were learned from the week of testing, and high-speed video footage of the samara in flight yielded useful data for analyzing the vehicle’s flight and validating the simulation. Figure 5.2.7 shows screen shots from a high-speed camera with a lateral view and a top view of a samara flight in the hanger. Chapter 6 and 7 present lessons learned regarding the samara’s flight performance, and provide commentary on how the guided samara airframe could have been improved.

![Figure 5.2.7: Screenshots of flights within the hanger](image-url)
Chapter 6

Samara Flight Behavior Discoveries

Over the course of this project, an effort was made to develop an understanding of the variables that affect the samara’s flight characteristics. This effort began with high-level conceptual brainstorming. It then progressed into a development of an analytical description of the samara’s dynamics and aerodynamics. Next a computer simulation was developed, and multiple runs were executed with various configurations and control scheme variations. In parallel with use of the simulation, two iterations of free-flight vehicles were designed, constructed, and test flown while systematically varying vehicle parameters. Finally, a working control system was designed, implemented, and integrated onto a samara airframe. This vehicle was then flight tested in various configurations and conditions.

The combined efforts described above have yielded a number of discoveries about the flight behavior of the samara, several of which have not been documented in prior literature. The following chapter presents these discoveries, along with the observations and analyses that may help to substantiate them.

6.1 Effect of Mass Axis Orientation

To explain how the orientation of the principal axis of inertia affects the stability of a samara, the effects of an external torque on a generic dumbbell will be examined. Figure 6.1.1a shows a dumbbell consisting of two point masses separated by a fixed distance. This rigid body is used as an example because its principal axes are in line with its axes of symmetry, and therefore its inertia matrix contains only diagonal terms. Additionally, this rigid body’s axis of least inertia is its longitudinal axis; the same is true of the samara.

In the figure, a torque (T) is applied to the dumbbell’s center of mass in the X-Y plane. The resultant rotation has an angular acceleration that is related to T and the body’s moment of Inertia. The equation of motion relating a moment and the resulting angular acceleration is:

\[ T = I \cdot \ddot{\omega} \]  

(6.1)
Figure 6.1.1: Dumbell with an applied torque and resultant angular acceleration (a), and a samara model with an applied pitching moment and the resultant angular acceleration (b)

Solving equation (6.1) for the angular acceleration:

\[
\dot{\omega} = I^{-1}T
\]  \hspace{1cm} (6.2)

It was stated earlier that this body’s inertia matrix contains only diagonal terms. Revealing the inertia matrix and the torque matrix in the angular acceleration expression yields:

\[
\dot{\omega} = \begin{bmatrix} I_{xx}^{-1} & 0 & 0 \\ 0 & I_{yy}^{-1} & 0 \\ 0 & 0 & I_{zz}^{-1} \end{bmatrix} \begin{bmatrix} T_x \\ T_y \\ 0 \end{bmatrix}
\]  \hspace{1cm} (6.3)

Performing matrix multiplication yields:

\[
\dot{\omega} = \begin{bmatrix} I_{xx}^{-1}T_x \\ I_{yy}^{-1}T_y \\ 0 \end{bmatrix}
\]  \hspace{1cm} (6.4)

Equation (6.4) indicates that the resultant angular acceleration vector will reside in the X-Y plane, and the magnitude of its X and Y components will be inversely proportional to the magnitude of the rigid body’s principal axes of inertia in the X and Y directions. This ensures that the vector will be positioned between the applied torque and the axis of least inertia.

Figure 6.1.1b illustrates the effect this has when applied to the samara. Due to the vehicle’s geometry, the axis of least inertia will always be in the spanwise (Y) direction. The applied torque shown is the net pitch-up moment generated by the blade during flight. The forward titled spanwise mass axis, which is also the axis of least inertia, results in an pitch-up angular acceleration about the Y axis, but also in a
smaller component of rolling (X) acceleration. This positive X acceleration results in a reduction of the coning angle during the autorotation startup procedure and during steady state autorotation. Given a positive (pitch-up) pitching moment, the results presented above dictate that a forward tilting mass axis is required in order to produce a stabilizing cone-down rolling acceleration. The coupling of the pitch and roll moments can act in a highly stabilizing or destabilizing manner, with great sensitivity to minute changes in the mass axis orientation. This relationship can be harnessed when designing and configuring a samara to result in a highly stable system, or to intentionally de-stabilize the system, which is a desirable when attempting to force rapid entrance into autorotation.

6.2 Effect of Mass Center Location

6.2.1 Chordwise Center of Mass

It is well known that the chordwise center of mass (CM) location has an effect on the samara’s flight characteristics. Norberg first presented a mechanism for pitch stabilization due to chordwise center of pressure (CP) movement [14]. Because CP location is a function of the airfoil’s angle of attack, when a disturbance upsets the equilibrium angle of attack, a restoring moment is generated as a result of the new CP location. The forces contributing to the equilibrium of the wing in the pitch axis are the resultant aerodynamic force, the perpendicular component of the centrifugal force, and the weight. This mechanism is presented in figure 6.2.1. Note that the figure neglects to illustrate the gravity vector acting at the CM.

![Figure 6.2.1: Samara pitch stabilization mechanism](image)

This is the same mechanism that provides pitch stability for a wing in a straight glide, with the exception that in a conventional wing, only the force of gravity opposes the aerodynamic force. The centrifugal force acting on a blade element tends to lay the samara blade parallel to the tip-path plane. This force can be resolved into a spanwise and perpendicular component. Both act at the chordwise CM, the location of
which is fixed at any angle of attack. The chordwise CP varies with angle of attack, depending on the particular airfoil section. Figure 6.2.2 shows the experimentally observed CP location as a function of angle of attack for a flat plate [22].

![Flat Plate Center of Pressure Location – NACA TN 3221](image)

Figure 6.2.2: Chordwise location of the center of pressure as a function of angle of attack for a flat plate

It can be observed that as the angle of attack increases, the CP moves towards the 50% chord point. If the CM is located behind the most forward potential CP location, a reduction in angle of attack below the equilibrium position is countered by a restoring moment. Similarly, if the CM is located ahead of the aftmost potential CP location, an increase in angle of attack (above the equilibrium one) will also be countered by a restoring moment. Because of the variation of tangential velocity in the spanwise direction, each blade element of a samara blade experiences a different angle of attack. Near the root this angle of attack is nearly 90 degrees, while at the tip it can become negative. The CP movement mechanism provides static and dynamic pitch stability to the entire wing as an average of the restoring moments created by each blade element. Norberg achieved best stability in model samaras when the CM was located between the 27–35% chord points [14].

This pitch stability mechanism relies on proper CP movement, and Norberg notes that many airfoils do not exhibit this trait. For example, symmetric airfoils have very little variation in CP location until after stall occurs, and many asymmetric profiles have s CP movement that is reverse of that for a flat plate, yielding pitch instability [14]. During flight-testing conducted with the first free-flight prototype, it was observed that the vehicle’s stability was not affected by large variations in chordwise CM position. The location was varied from the 23–53% chord positions, with no noticeable change in startup or autorotation stability. The airfoil used for this vehicle was essentially a flat plate with a 0.25 in. tubular leading edge offset from the center of the plate. This airfoil is shown in figure 6.2.3a, next to a truly flat airfoil section. Because of the offset tubular leading edge, this airfoil had some effective curvature, an approximation of which is illustrated by the dotted line. Interestingly, this prototype reached its stable equilibrium upside down from the orientation shown in figure 6.2.3a. It is hypothesized that in this configuration the first prototype’s airfoil had a very wide CP movement range, and hence yielded a wide range of stable CM locations. Additionally, the upside-down orientation of this airfoil would increase the tendency for flow to
separate over its top surface, even at relatively low angles of attack. This would result in the majority of the wing residing in a separated flow regime. This separation is shown in figure 6.2.4.

Figure 6.2.3: The airfoil used on the first free-flight samara yielded effective curvature (a), as compared to a flat plate (b)

Figure 6.2.4: Separated flow due to the orientation and shape of the airfoil used on the first free-flight samara

**Separation Induced Pitch Stability**

It is thought that this separated flow is one of the main factors for the extreme pitch stability that was exhibited by the first free-flight prototype, and the instability exhibited by the guided samara. Most symmetric airfoil sections show very little or no CP movement at low angles of attack prior to stall. In the pitch stability mechanism described above, the greater the range of CP movement, the greater the range of angles of attack the wing can recover from. This means that in the case of a symmetric airfoil flying in an un-stalled condition, there is very weak pitch stability. Conversely, the free-flight prototype’s wide range of CP movement granted it extreme pitch stability. This explains very well the periodic pitch instability observed during the NASA flight-testing. The guided samara flown in these tests utilized a fully symmetrical NACA 0012 airfoil, and had a wingspan nearly 50% greater than the initial free-flight prototype. The guided samara had a similar descent rate to the free-flight prototype, however, because of its greater span, its average tangential velocity was higher, yielding a lower average angle of attack. This means a larger portion of the wing was in an un-stalled condition, due both to the increased tangential velocity, and also due to the decreased separation tendency of the NACA 0012 as compared to the free-flight prototype’s airfoil (shown in figure 6.2.3). The end result is that as the guided samara entered autorotation and accelerated its spin, a larger fraction of the wing became un-stalled, and hence entered a
realm of poor pitch stability. When a critical rpm was attained, the pitch instability was such that a small perturbation knocked the vehicle out of autorotation, beginning the process once more.

Although this proposed mechanism of separation induced stability conceptually makes sense, and was verified through flight observations, great difficulty was encountered when attempting to verify it using the simulation. Indeed, there was difficulty modeling any aspect of pitch stability, including the CP movement stabilization mechanism. Conversely to the mechanisms just presented, the simulation results showed greatest stability when the CP position was fixed at the 25% chord location, which is the CP location in agreement with classical thin airfoil theory. Interestingly, a samara simulation developed by Seter and Rosen yielded the same results. In this simulation it was found that adding CP movement resulted in a great deviation between theoretical and experimental results [18]. They concluded that there must be another mechanism involved in the process that was unaccounted for.

### 6.2.2 Vertical Center of Mass

The vertical (Z axis) center of mass also has an effect on pitch stability. A CM below the vehicle’s resultant center of pressure yields a restoring moment that acts in addition to the CP movement restoring moment. It has been observed during flight-testing that the direction of spin during stable flight has generally coincided with a vehicle orientation that yields a vertical CM below the vehicle. However, in the flight testing of the first free-flight prototype, it was found that within the range of adjustability, regardless of the vertical CG position the vehicle always autorotated with the airfoil in the “upside down” orientation shown in figure 6.2.4. This was the case even when the vertical CM was slightly above the center of the wing. This indicates that the restoring moment generated by the CP movement mechanism was much greater in magnitude than the destabilizing moment due to the vertical CG location.

### 6.2.3 Spanwise Center of Mass

Norberg observed through experimentation that the spanwise center of mass of a samara must lie between 0–30% of the span length from the root of the blade [14]. It is ideal to minimize the sink rate, both from a performance standpoint but also from a stability standpoint. A main criterion defining steady autorotation for the samara is minimal coning angle. If this angle becomes too great, the helical tip path becomes very spread out and sideslip velocities increase, leading to instability [18]. Keeping the CM, and hence the center of rotation (CR) near the root increases the swept area of the blade, which minimizes the disk loading, which in turn decreases the descent rate and the coning angle. This trend was observed during all stages of flight testing conducted during this research.

Another reason for keeping the CM close to the root is to prevent the proximal end of the blade from acting as an airfoil in reverse flow. Although the tangential velocities experienced between the root and the CM are very low, the pitching moments created by this reverse flow could destabilize a marginally stable samara.
6.3 Effect of Wingtip Shape and Orientation

During the controlled flight-testing at NASA, wingtips of several sizes, shapes and orientations were used. As presented in Chapter 5, during flight-testing it was found that a centered wingtip performed better than a cantilevered wingtip protruding vertically from the end of the wing. It was also found that a wingtip with a swept leading edge created periodic instability while a wingtip with a swept trailing edge or lacking any sweep enhanced stability. It is hypothesized that this effect of the wingtip is due to the differing angle of attack the wingtip experiences as its mean aerodynamic axis is moved in the chordwise direction. Figure 6.3.1 illustrates that due to the radius of the vehicle’s tip path, a wingtip located forward of the vehicle’s yaw axis experiences a positive angle of attack, while a wingtip located to the rear of this axis experiences a negative angle of attack.

The lift generated by the offset wingtip generates a rolling moment (X moment) due to the vertical offset of the CM from the center of the wingtip. This moment ($R_m$) is shown in figure 6.3.2. If the wingtip is offset forward and a outboard lift vector is created, the moment generated acts to decrease the Samara’s coning angle. If the wingtip is offset backwards and an inboard pointing lift vector is created, the generated moment acts to increase the coning angle. This moment is increased if the wingtip is cantilevered from the end of the wing and protrudes upward.

During flight testing it was observed that when the wingtip was shaped so that its net aerodynamic axis was shifted backwards, when the vehicle reached a particular rpm, the coning angle increased to an angle such that the vehicle exited autorotation. The same principles can be used to control the coning angle using the wingtip’s rudder control surface. Rather than shifting the wingtip forward to decrease the coning angle and increase stability, the control surface can be deflected during autorotation to create a cone-down moment.
Figure 6.3.2: Moment arm due to offset of CM from wingtip lift vector
Chapter 7

Conclusions and Future Work

7.1 Summary and Conclusions

Analytical and experimental studies were performed to investigate the viability of actively controlling a vehicle in the form of a samara. The primary goal was to demonstrate the feasibility of omni-directional glide slope control during steady autorotation. To achieve this goal, the following steps were taken:

- Conceptual vehicle design
- Development of a 6 degree-of-freedom analytical model describing samara autorotation
- Development of computer simulation to numerically solve this model
- Free-flight testing of samara models to validate simulation
- Testing of servos to determine if they met performance criteria
- Testing of electronic compass to determine if it met performance criteria
- Use of simulation to predict samara’s response to control inputs
- Design and fabrication of guided samara airframe
- Design and fabrication of guided samara control system
- Flight testing of guided samara

The conceptual design process resulted in several control concepts, and several choices for actuators and azimuthal angle sensors. Testing and simulation results were required to provide insight in selecting the best control concept and hardware for the final guided samara design. The free-flight simulation results predicted descent and rotation rates within approximately 10% of experimentally observed performance, while the coning angle was predicted within 25% of the observed angle. The controlled simulation employing a trailing edge wing flap indicated a large increase in the vehicle’s tilt during control input, while the simulation employing a wingtip rudder indicated considerable less tilt. Although the wingtip rudder was predicted to provide less control authority, it was chosen as the control scheme due to its decreased induced tilt. Based on compass experimentation, it was determined that if the wingtip rudder scheme was used, a 2-axis electronic compass could be employed and the error due to tilt would be within
tolerable levels. Servo experiments indicated that for the rotation rates, amplitudes and aerodynamic loads anticipated, off the shelf hobby servos provided adequate performance.

An airframe was designed and fabricated using fiberglass and foam core construction. A control system was designed, and employed a PIC microprocessor to process the compass signal and command the servo to periodically actuate in sync with the vehicle’s rotation. Flight-testing of the guided samara was conducted at NASA Langley’s Impact Dynamics test facility. During the course of the five day testing period, poor weather hampered outdoor flights. Flights were attempted indoors within a hanger, however, due the short drop distance and technical difficulties, the tests were inconclusive.

Although the final demonstration of controlled flight was inconclusive, the control system functioned as designed, and based on the favorable controlled simulation results, the prospect of achieving controlled flight remains very close at hand. Through the course of the flight testing, many samara flight behavior observations were made, the most notable of which was the effect of the mass-axis orientation. To the best of author’s knowledge, this is the first time this behavior has been observed and explained.

7.2 Future Work

The most urgent work to be completed is the successful controlled flight demonstration. Although weather played a major role in hampering the tests, it would be advisable to perform a redesign of the airframe to provide a more stable platform. Implementing the observed flight behavior discoveries, a new airframe would possess greater reliability in entering autorotation, as well as enhanced stability during autorotation. Although not necessary for successful demonstration, replacement of the 2-axis electronic compass with a 3-axis rate gyro would eliminate tilt error considerations, and would allow the use of the more effective wing trailing edge flap control scheme.

To describe more accurately the full behavior of the samara, an analysis of the transient from dart-mode to autorotation should be developed. This would also aid the development of the dart-mode control system. Additionally, the pitch stabilization mechanism of the samara is still not fully understood, and further analysis into this realm is warranted.
Bibliography


