An Experimental Study and Flight Testing of Active Aeroelastic Aircraft Wing Structures

by

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Abstract

An experimental investigation on active control of aeroelastic aircraft wing structures using piezoelectric actuators and sensors is presented. To this end, wind tunnel and remotely piloted vehicle wing models were designed, fabricated, installed, and tested. Computational structural and aerodynamic wing models were created, in order to determine the wing natural frequencies and modal shapes, and to predict the flutter speed. A digital controller was designed and implemented. Open and closed-loop vibration and flutter tests were conducted in the wind tunnel and in flight, with excellent correlation achieved with computational predictions. Two different active wing concepts were analyzed: the first model consists of a wing with piezoelectric actuators attached to the wing skin, and the second wing model has piezoelectric actuators mounted in the main spar. The experimental results obtained have shown that the adaptive wing response had improvements in almost all the RPV flying conditions compared to the corresponding passive wing vibration, for both the active skin and the active spar wing concepts. Also, it was demonstrated that the flutter speed of the active wings increased compared to the corresponding passive wings.
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To Helena and Bruno
Chapter 1

Introduction

When an aircraft is flying, the aerodynamic forces cause deformations in the structure (especially in the wings) during the entire flight envelope. These deformations are known as vibrations defined as a motion that repeats itself in time. Although these vibrations are necessary and inevitable, they are also responsible for structural damage. This damage can occur in two different ways: abruptly or caused by fatigue. The fatigue damage is caused by the continuous low vibration of the structure. Abrupt damage happens when a catastrophic aeroelastic event takes place, for example when the aircraft experiences wing divergence or flutter. In both cases, the damage can be catastrophic and cause the loss of people and aircraft. As a result, an important issue in aircraft design is the study of the aeroelastic response of the flight vehicle.

The work presented in this thesis concerns the active aeroelastic response of aircraft structures. In particular, the study of a Remote Piloted Vehicle (RPV) wing deformations and the reduction of these vibrations using piezoelectric actuators and sensors was performed, in order to increase the flight envelope in terms of flutter. It is shown that the piezoelectric sensors and actuators are effective when used in small scale flight vehicles and a considerable increase in flutter speed was observed.
CHAPTER 1. INTRODUCTION

Two adaptive wing concepts are proposed in this thesis: a wing with piezoelectric materials mounted in the wing surface (adaptive skin concept), and a wing with piezoelectric materials mounted in the main spar (adaptive spar concept). The research was carried out in three logical stages: first, computational analyses were performed to predict the response of the adaptive wings in passive mode; next, wind tunnel tests were carried out to validate the computational models; and finally a flight test was performed to verify the performance in real flight conditions. The computational study was performed using commercial finite element (ANSYS) and aeroelastic analysis (ZAERO) programs. The ANSYS program was used to determine the wing natural frequencies and modal shapes of the adaptive wings in passive mode. ZAERO program calculates the wing flutter speed. In wind tunnel and flight tests, the wing was tested in two different configurations: with and without the actuation of piezoelectric materials, i.e., in the active and passive modes. In other words, the wing with vibration control and the wing in free vibration (i.e., without vibration control). After obtaining the vibration results of both wing configurations, it was possible to analyze the differences between them, and measure the wing vibration improvements, i.e., the reduction of vibration in terms of average cycles and magnitude. For control and data acquisition, the MATLAB program and DSPACE tools were used.

In the next Section, the motivation of this thesis is described in detail. The background is presented next in Section 1.2. An overview of the past studies and developments in the area of active aeroelastic structures is presented in 1.2.1. The state of the art in multifunctional materials is presented in 1.2.2. Finally Section 1.3 describes the content of the various Chapters in the thesis.
CHAPTER 1. INTRODUCTION

1.1 Motivation

A RPV is the predecessor of an Unmanned Aerial Vehicle (UAV). The main difference between an RPV and UAV is that the RPV is not a self-piloted aircraft. The RPV needs to have someone flying it, using remote control. Because of that, RPVs still have a range problem, which is limited by the radio transmitter range. The UAV is self-piloted, i.e., autonomous, and carries a computer with the entire flight envelope previously programmed. They can carry cameras, sensors, communication equipment or other payloads. Therefore, they can be used in reconnaissance, intelligence-gathering role and combat missions. Nowadays, UAVs can be divided in two categories: Tactical and Endurance (long range) [1]. Most importantly, UAVs are today widely used in military reconnaissance and forest fire observation missions.

In the last two decades, the technological developments in the areas of materials and computer sciences have been very promising. The combination of multifunctional materials with faster computers and data acquisition systems has resulted in adaptive systems. The development of materials science made the materials multifunctionality possible, such as piezoelectricity. On the other hand, the development of computational sciences made advances in areas such as design, manufacture and control possible. An adaptive system is a structure with embedded sensors, that provide information about its environment, for instance, forces, tension field, displacements, etc. Then, this data is used by a processor and a control module in order to generate a response to the actuators, attached to the structure, in order to change the structure properties. The multifunctional materials applied to structures can mitigate structural problems involving vibration suppression, noise reduction and shape control.
Adaptive systems are also called “smart structures”. These structures are known as “smart” because they sense changes in their environment and respond accordingly to these changes [2]. In the past, some passive solutions were used to solve aeroelastic dynamic problems, such as increasing the structural rigidity or balancing the mass. Increasing the structural rigidity makes the structure heavier. Here, the use of active control systems using distributed actuation is proposed and this approach can result in an improved structural response without the added weight penalty.

The most popular multifunctional materials are the piezoelectrics, electrostrictives, magnetostrictives, shape memory alloys, electrorheological and magnetorheological fluids. Multifunctional materials are also known as “smart materials”. The “smart structures”, mentioned in the last paragraph, integrate “smart materials” and controllers. These materials respond to external stimuli like electric, magnetic or thermal fields. In particular, piezoelectric materials can operate as both sensors and actuators. In sensor mode, they produce voltage when a mechanical strain is applied. In actuator mode, they undergo elongation when an electric field is applied [3]. In general, piezoelectric materials are more suitable for operation at high frequencies compared with the other multifunctional materials. However, since they are easily breakable, the manufacture and handling of piezoelectric crystals are difficult. Although the ceramic properties of the piezoelectrics are enough for several applications, when large displacements and forces are intended or certain frequency ranges are expected, the use of other type of materials is necessary.
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1.2 Background

1.2.1 Overview of the Active Aeroelastic Structures Developments

Aeroelasticity is the interaction between elastic, inertial and aerodynamic loads, acting on the aircraft in operating conditions. In normal flight conditions, these loads may cause the aircraft to become unstable. In real life, aeroelastic events can be static or dynamic phenomena. A classical example of a static problem is the divergence phenomenon, and flutter is possibly the most important dynamic event in aeroelasticity. As an illustration of this event importance, the flutter envelope prediction is crucial to the certification of civil and military aircrafts. Also, the active suppression of aeroelastic instabilities such as flutter or divergence leads to improved performance. Therefore, many control strategies have been applied to suppress flutter or control unacceptable wing motion.

Concerns and considerations about aeroelasticity were considered very early in the history of aviation. The failure of the Langley's monoplane, in October 1903, was considered to be caused by aeroelastic problems, possibly by the wing torsional divergence \[4\]. The Wright brothers took advantage of the wing flexibility to control what is known as the first successful flight, in December 1903. Instead of ailerons or flaps to control their airplane, they twisted the craft wings as a mean to control its rolling motion. This system avoided the extra weight of the aileron control surfaces \[5\].

Aeroelastic solutions generally involve increasing of the structure stiffness or mass balance (passive solutions), which typically involve increase of weight and cost while decreasing performance \[4\].
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The concept of active control to improve the aeroelastic performance of wings emerged in the fifties [6]. Probably, one of the primary efforts in the direction of active control was the US Active Aeroelastic Wing (AAW) program [7]. The AAW concept is a technology that integrates air vehicle aerodynamics, active controls, and structures together to maximize air vehicle performance. They have played with the wing aeroelastic flexibility by using multiple leading and trailing edge control surfaces, activated by a digital flight control system. The energy of the air was used to achieve the desirable wing twist with very little control surface motion. The AAW concept was successfully tested in the NASA Langley transonic dynamics wind tunnel. Based on these tests, a joint Air Force, NASA and Boeing flight test program was launched [8]. In this program, an F/A-18 fighter was modified to demonstrate the AAW concept. At the end of January 2003, the AAW aircraft had successfully flown eleven research missions [9]. The Russian Aerospace Research Institute tested active aeroelastic concepts using a small additional control surface ahead of the wing leading edge, improving the roll control. They also have developed new structural elements that enable large structural deformations of aerodynamic surfaces, in order to obtain control surface deflections with smooth curvature, thus improving the aerodynamic effectiveness [10].

In the last two decades, a new actuation concept for structural control has emerged. This concept uses the multifunctional materials properties to control the structural stiffness and shape of the composite materials. Several studies are being performed to demonstrate applications of adaptive structures in aircraft, helicopters and submarines. The adaptive structures technology is expected to significantly reduce dynamic instabilities and vibrations [11, 12, 13].

In 1990, at the Massachusetts Institute of Technology (MIT), [14], investigations were performed using embedded piezoelectric actuators in laminated materials. In
Japan, [15], projects have focused in the design of adaptive truss structures. In Europe, researches were performed using shape memory alloys at the University of Twente, Netherlands, and using piezoceramics at ONERA, France. The European Space Agency (ESA) has been investigating the application of smart materials in aerospace structures [16]. In 1991, the Smart Structures Research Institute was created, at the University of Strathclyde, in Scotland [17].

In 1998, Forster and Yang [18] examined the use of piezoelectric actuators to control supersonic flutter of wing boxes. The wing box contained piezoelectric actuators that control the twist of the wing, in order to change the free-vibration frequencies and modes, thus, controlling flutter speed. This study has shown that the weight of the wing box can be decreased by adding piezoelectric actuators to meet the flutter requirement at smaller thickness of skins, webs and ribs.

In 2003, several studies were developed using the smart structure concepts. For example, the Italian Aerospatial Research Centre (CIRA) designed torsion tubes to produce geometry variations and transmit deformations to mechanic devices. This tube is a cylindrical anisotropic laminated shell. The numerical and experimental results aimed to maximize the tangential rotations and the transmitted energy, in order to obtain suitable deflections of the control surfaces. The main benefits that they observed include the reduction of negative aeroelastic impacts on aircraft performance and stability; cost reduction, by decreasing the size of stabilizer surfaces and total structure weight; reduction of the emissions, by reducing the engine power demand [19]. At the University of Manchester, United Kingdom, in 2003, a research program investigated the development of “active internal structures” concepts, in order to enable the active aeroelastic control of aerospace structures. Using wing internal structures, in particular through changes in the position and stiffness of wing spars, they aimed to control the wing bending and torsional stiffness. Their analytical and
experimental results showed that it is possible to control the wing twist and bending using this type of internal structures [20]. Also in 2003, at the University of Michigan, USA, a research program has worked to reduce the vibration in a rotorcraft using actively controlled flaps [21].

The active structures concept has also been used in Micro Air Vehicles (MAVs). For instance, at the University of Florida, USA, an investigation has studied the use of morphing as a control effector for a class of MAVs with membrane wings, in the year 2003 [22]. The morphing was restricted to twisting the wing for roll control. Experimental data showed that the morphing can be easily achieved and greatly improves the flight characteristics, when compared with traditional control surfaces.

1.2.2 State of the Art in Multifunctional Materials

Although significant advances in smart materials have taken place in the past decade, the presence of the piezoelectric effect in quartz was experimentally confirmed, over 100 years ago, by Jacques and Pierre Curie [23]. Then, the first application of the piezoelectric crystal effect was force and charge measurement apparatus, patented by the Curies in 1887 [24].

As explained in Section 1.1, piezoelectric materials can be used as sensors and actuators. In sensor mode (called direct mode), the piezoelectric material becomes electrically charged when a mechanical deformation occurs. Piezoelectric sensors can be used in order to detect strain, motion, force, tension and vibrations, since they generate an electric response to these stimuli. In actuator mode (called inverse mode), the piezoelectric material deforms itself when subjected to an electrical field. Piezoelectric actuators can generate motion, force, tension and vibrations. The figure 1.1 illustrates the piezoelectric actuator mode.
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The force and deflection output of the piezoelectric actuators for a given applied voltage can be considered linear, as shown in the figure 1.2. For a given voltage applied to the actuator, its displacement is reduced as the load increases, until the blocking force is reached at zero deflection. On the other hand, the displacement is increased as the load is removed, until the free deflection. The area under the line represents the work done by the piezoelectric actuator. The energy transferred from the actuator to the mechanical system is maximized when the stiffness of the actuator and the mechanical system are matched [25].

It is desirable that piezoelectric sensors have a response that varies linearly with changes in the measured quantity. As a result, piezoelectric elements used in sensors generally operate in the linear region, such that the voltage generated across the element varies linearly with the magnitude of the mechanical stress. For a given piezoelectric material, the amount of voltage produced by the ceramic subjected to a stress can be increased by increasing the thickness of the ceramic [26].

Figure 1.1: Illustration of a piezoelectric actuator deformation.
Piezoelectric materials usually have the form of patches, thin disks, tubes or very complex shapes fabricated using solid free form fabrication or injection molding [27, 28]. Traditional piezoelectric materials are called PZT (lead zirconate titanate), which have small strain levels (on the order of 0.1% to 0.2%). The new relaxor ferroelectric single crystals (PZN-PT and PMN-PT) can develop strains on the order of 1% and have approximately 5 times as much strain energy density as conventional piezoceramics [29]. The amount of strain produced in the material is dependent on the thickness of the element and the magnitude of the voltage applied across the thickness. Piezoelectric materials have been investigated to control vibrations and acoustics in a variety of structures [30, 31].

For the majority of the piezoelectric actuators, the focus of the research has been on an effort to amplify the deflection of the material. Piezoelectric actuators can be classified in three different categories, based on its amplification scheme: internally leveraged, externally leveraged, and frequency leveraged. Internally leveraged actuators generate amplified strokes through their internal structure without using external
CHAPTER 1. INTRODUCTION

mechanical components, including: bender, stack, reduced and internally biased oxide wafers (RAINBOW), composite unimorph ferroelectric driver and sensor (THUNDER), telescoping, C-block, Recurve and Crescent actuators. Externally leveraged actuators are based on external mechanical components to achieve their actuation ability, including: flexure-hinged, Moonie, Cymbal, double-amplifier, bimorph-based, pyramid, X-frame, and flexextensional hydraulic actuators. Frequency leveraged actuators depend on an alternating control signal to generate motion, including inchworm and ultrasonic motors.

Among the internally leveraged actuators, stack actuators are thin piezoceramic patches piled in order to linearly increase their overall deflection, and maintaining a low voltage requirement (see figure 1.3(a)). Displacement and force of a stack actuator are directly proportional to its length and cross-sectional area, respectively. Another internal leveraged actuator is the bender. Bender actuators are composed by two or more layers of piezoelectric material, which are poled and activated such that layers on opposite sides of the neutral axis have opposing strains. These opposing strains of the two piezoelectric layers create a bending moment, causing the entire bender to bend. In order to achieve structural stability, inactive substrates may be added to these active layers. As an illustration, most benders have piezoelectric material extending the full length of the beam, as shown in figure 1.3(b).

Some practical difficulties related with the use of raw piezoceramics as actuators include soldering, cracking because of their fragile nature, and electrical isolation. The QuickPack actuators (manufactured by ACX Inc.) are a step forward in terms of applying piezoelectric technology to commercial products (see figure 1.4). This is the type of actuators used to perform the experiments focused on this thesis. These actuators contain two piezoceramic elements enclosed in a protective polymide insulation material. The QuickPack actuators can be used as patches, to induce in-
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Figure 1.3: (a) Diagram of a piezoceramic stack; (b) Typical bimorph bender actuator.

A unimorph actuator is a special case of a bender actuator. It is a composite beam, plate or disk with one active layer and one substrate (an inactive layer). The mentioned RAINBOW, Crescent and THUNDER actuators are typically referred to as unimorph benders. The RAINBOW actuator is a piezoelectric wafer that is chemically reduced on one side. A partially metallic layer is formed on one side placing the piezoelectric element in compression, forming a hemispherical container. The Crescent actuator is very similar to the RAINBOW actuator. It is a stressed-biased unimorph actuator that is fabricated by cementing (using epoxy or solder) metal and electroded ceramic plates together, at an elevated temperature. When the actuator approaches the room temperature, a prestress is induced in the active material (due to the difference in the coefficients of thermal expansion of the metal
and the ceramic) resulting in a unimorph actuator that is curved in shape. The THUNDER is another prestressed actuator that consists of a layer of a ceramic wafer attached to a metal backing using a polymide adhesive film. The C-block, Recurve and telescoping actuators are called building-block actuators. Building-block actuators have numerous small actuation units, called building blocks, which are combined in series and/or parallel to form larger actuation systems with improved performance.

Externally leveraged actuators use an external mechanism to increase the output deflection, by decreasing the output force. These external mechanisms can be mechanical or hydraulic. A simple way of increasing the displacement of an actuator is the use of a mechanical lever arm. Although this mechanism increases the displacements output, the actuator force is decreased. The flexure-hinged actuator uses this principle. Flextensional actuators use a piezoceramic stack and an external amplification mechanism, in order to convert the motion generated by the stack to a usable output motion in the transverse direction. Moonie, Cymbal and bimorph-based are examples of flextensional actuators.
In the category of the frequency leveraged actuators, the output strain of the actuators is increased by using the frequency performance of the piezoelectric material to rapidly move the actuator in one direction in a series of small steps. The first type of actuator to operate using frequency was the Inchworm. The Inchworm consists of three connected actuators that actuate in sequence to move the actuator down a rod.

Besides piezoelectric materials, there are more multifunctional materials that can be used in order to obtain an adaptive structure. Next, it will be presented some of these multifunctional materials.

Shape memory alloys (SMA) return to their original form when heated above their critical temperature, i.e., they “remember” their original crystalline structure or shape. SMAs are used as actuators to change characteristics of the host structure. They have relatively large actuation force and high strain output damping capabilities. However, they may have large hysteresis, and due to their slow response, they are best suited for low frequency applications. The most common shape memory alloy is nitinol. Some efforts were made in order to use shape memory alloys as actuators, including embedded shape memory alloys in composites or in conventional structures. Some studies were performed using SMA, like the development a model that studies the behaviour of composites with shape memory alloys [32]. SMAs are also quite suitable for slow motion of control surfaces such as flaps in helicopters [33].

Electrostrictive materials are similar to piezoelectric materials. The electrostrictive actuators are characterized by possess the electrostriction property. Electrostriction is a phenomenon observed in all dielectric materials. When an electric field is applied across the dielectric material, the dipoles align themselves with the field. This process induces an internal strain and the material changes its dimensions [34]. When the electric field is removed the dipoles re-orient and the material returns to its original dimensions. The induced strain is proportional to the square of the applied electric
field, thus it is always positive, i.e., the material is under tension. The most popular electrostrictive material is the lead magnesium niobate (PMN), which have high strain capabilities and very low hysteresis properties.

The magnetostrictive materials modify their dimensions when a magnetic field is applied. Their dimension change is a result of a re-orientation of the atomic magnetic moments, or small magnetic domains. As the magnitude of the applied magnetic field increases, more domains become aligned, until magnetic saturation occurs (when all magnetic domains are aligned with the applied magnetic field). When the magnetic field is removed, the material returns to its original dimensions. The produced strain is proportional to the square of the magnetic field. Thus, like the electrostrictive materials, the strain is always positive (tension). Terfenol-D is one of the most popular magnetostrictive materials. Studies [35] have demonstrated that strains produced by magnetostrictive materials are smaller than those produced by electrostrictive materials, and the hysteresis is higher than the electrostrictive material.

The rheological fluids are multiphase materials that consist of field-responsive particles suspended in a carrier non-conducting fluid [36]. They can be electro- or magneto-rheological fluids (ER or MR). The viscosity of ER and MR fluids varies when an electric or magnetic field, respectively, is applied. These active fluids can adapt and respond almost instantly and have been used in clutch, brake, valve-type devices, dampers, and shock absorbers [37, 38].

Optical Fibre sensors respond to strain and temperature when a shift in their optical wavelength takes place. They can be used in the structures skin or directly embedded in the structure. Many optical fibres can be manufactured onto a single optical fibre, and then interrogated independently to provide distributed measurements over large structures, such as civil infrastructures and ships [3].
CHAPTER 1. INTRODUCTION

1.3 Structure of the Thesis

Prior presenting the attained results in the different performed analyses, some fundamentals of flutter are presented in first Section of the Chapter 2 of this thesis. This Section describes the fundamentals of structural vibration, in which is presented the governing equation of a general vibrating system, and its basic elements. The general methods for the determination of the natural frequencies and mode shapes of a system are identified, respectively, as eigenvalue and eigenvector extraction problems. The relation between the wing vibrations and wing flutter speed is explained. The flutter phenomenon is defined and some different types of flutter are presented. The equation of motion of an aeroelastic system, in terms of a discrete system, is explained. Finally, a brief explanation of the ZAERO flutter solution technique is presented.

In Chapter 2, the remaining Sections have the objective to present the results of the computational analyses performed in this project. Two different computational analyses were performed: a finite element analysis, using ANSYS program, and a flutter analysis, using ZAERO program. The finite element analysis is presented in Section 2.2, and was done in order to generate the passive wings natural frequencies and mode shapes. Note that only the passive wings were studied. No ANSYS analysis was performed using the wings in active configuration. In Section 2.3 describes the flutter analysis. The ZAERO program was used, which imports the solution generated by ANSYS program and calculates the wings flutter modes and speeds of the wings in passive configuration. The ZAERO program was used to calculate the flutter speed only when the wing has a no conventional configuration. In the case of a conventional wing, i.e. with a main spar and ribs, a method presented by [39] is used to calculate the flutter speed.
CHAPTER 1. INTRODUCTION

The wind tunnel tests are presented in Chapter 3. First, all the experimental apparatus is described in Section 3.1. In this Section the following hardware components are presented: the wind tunnel, the tests articles, the digital controller, and electronic equipment. In the tests articles Subsection, an extensive description of the respective articles is performed. Several photographs of the articles are shown, and several wing schemes were included in order to explain the active wings control vibration process. In Section 3.2 the wind tunnel tests objectives and procedures are described. In this Section, the entire hardware system used for wind tunnel tests is presented, explaining all the connections in the circuit. Additionally, the approach followed in order to design the control model, and the control model itself, are presented. Yet in Section 3.2, it is performed the description of the tests procedures and measured data. Finally, in Sections 3.3 and 3.4 the wind tunnel results of, respectively, the active skin wing and active spar wing are presented.

The Chapter 4 describes the performed flight tests. The hardware involved in these tests is presented in Section 4.1. This Section includes the description of the RPV flight model and the additional electronic equipment needed exclusively for the flight tests. Like in Section 3.2, in Section 4.2 the flight tests objectives and procedures are described. The control approach used for these tests is also explained. At the end of this Chapter, in Section 4.3, the flight tests results of the RPV with active spar wing are presented.
Chapter 2

Computational Analyses

2.1 Fundamentals of Flutter

As referred in Chapter 1, any motion that repeats itself after an interval of time is called vibration or oscillation [40]. A system that vibrates is generally defined through three properties: elasticity, a mean of storing potential energy; mass or inertia, a mean of storing kinetic energy; and damping, a mean of losing energy. In vibration theory, a vibrating system is generally described with the following elements: mass, spring or stiffness, damper and excitation force. This way, the first three elements describe the physical vibrating system. The mass and spring store energy and the damper dissipates it in the form of heat. On the other and, the energy is given to system by the excitation force. Thus, a single degree of freedom system, with translational mass, is generally defined as shown in figure 2.1. In this figure, $k$ is the spring constant, $c$ is the damper constant, $m$ is the mass, $f(t)$ is the excitation force, and $x(t)$ is the mass displacement.
In terms of the type of excitation force, the vibration is known as *free vibration* when the system, after an initial disturbance, is left to vibrate on its own. On the other hand, the vibration is called *forced vibration* when the system is often subjected to an external force. In terms of damping, an *undamped vibration* happens if no energy is dissipated in friction or other resistance during the vibration; a *damped vibration* occurs if there is energy lost during oscillation. In terms of periodicity, a system has a *deterministic vibration* if the magnitude of the excitation acting on the system is known, or *random vibration* if that magnitude can not be predicted. The vibration of a wing is a *random vibration* since it is excited with the wind.

A degree of freedom of a system is the minimum number of independent coordinates required to determine the positions of all parts of this system, at any time. Systems with a finite number of degrees of freedom are called *discrete systems*. Systems with an infinite number of degrees of freedom are usually called *continuous systems*. Most of the time, *continuous systems* are treated as *discrete systems*, since the methods to analyze *continuous systems* are only applicable to simple problems. In this study, our system is a wing, and because all its components has an infinite number
of mass points, thus has an infinite number of degrees of freedom. Considering the wing system, the governing equation may be represented as follows:

\[ [m]\{\frac{d^2x(t)}{dt^2}\} + [c]\{\frac{dx(t)}{dt}\} + [k]\{x(t)\} = \{f(t)\} \]  \hspace{1cm} (2.1)

The case presented in figure 2.1 is a very simple vibrating case, in which it is possible to analytically determine the exact solution. However, the solution of the systems governing equations is often more complex in real problems, and it is impossible to consider all the details for the mathematical model, or because they have infinite number of degrees of freedom or they have considerable irregularities in the oscillatory motion. A normal procedure is the use of numerical methods involving computers to solve these equations. For instance, the finite element method is a numerical method that can be used for the accurate solution of complex structural vibration problems. This is a numerical method in which the structure is divided and replaced by several pieces or elements. Each one of these elements is assumed to be a continuous structural member called finite element. All the elements defining the structure are assumed to be interconnected at certain points known as nodes. During this method solution process, the equilibrium of nodal forces and the compatibility of displacements between the elements are satisfied, such a way that the entire structure is made to behave as a single entity [40]. Because of the complexity of the structures in study, it was previously decided that a commercial code would be used in order to calculate the solution of the problem. In this thesis, the wings structures were studied using the ANSYS program, which has many finite element analyses capabilities.

Having presented the basic elements of a vibrating system and its general governing equation (2.1), it is now time to describe the type of vibration analysis that was done in this thesis. Since the final objective is to calculate the wings flutter speeds, the vibration analysis that was performed consists in the determination of
the systems natural frequencies and mode shapes, which in ANSYS program is called *modal analysis*. Since the determination of systems natural frequencies is an eigenvalue problem, which solution corresponds to the undamped free vibration of the system, the ANSYS *modal analysis* starts by solving the following equation of motion:

\[ [m]\left\{ \frac{d^2x(t)}{dt^2}\right\} + [k]\{x(t)\} = \{0\} \quad (2.2) \]

Then, for a linear system, free vibrations will be harmonic of the form:

\[ \{x(t)\} = \{\phi\}_i \cos(\omega_i t) \quad (2.3) \]

where \( \{\phi\}_i \) is the eigenvector representing the mode shape of the \( i^{th} \) natural frequency, \( \omega_i \) is the \( i^{th} \) natural circular frequency, and \( t = \) is the time. Thus, equation (2.2) becomes:

\[ (-\omega_i^2[m] + [k])\{\phi\}_i = \{0\} \quad (2.4) \]

This equality is satisfied if either \( \{\phi\}_i = \{0\} \) or if the determinant of \(( [k] - \omega_i^2[m] )\) is zero. The first option is the trivial one and, therefore, is not of interest. Thus, the second one gives the solution:

\[ ||[k] - \omega_i^2[m]|| = 0 \quad (2.5) \]

This is an eigenvalue problem, which may be solved for up to \( n \) values of \( \omega^2 \) and \( n \) eigenvectors \( \{\phi\}_i \), which satisfy the equation (2.4), where \( n \) is the number of degrees of freedom of the system. The ANSYS has several techniques to perform the eigenvalue and eigenvector extraction. In this project the Subspace Method was used to perform this extraction, and its algorithm description is available in the ANSYS manual.

There is a strong relation between the vibration of a wing and its *flutter speed*. 
It is known that a wing in flight is in continuous vibration because of the air flow. Additionally, when the wing is disturbed by the wind (for instance, when a gust strikes the wing), the wing motion may be such that the amplitude of vibration tend to decrease, remain constant or increase. The first case occurs when the airspeed is between zero and the critical flutter speed (stable condition). The wing vibration will remain constant when the airplane is flying at flutter speed (neutral stability). Finally, at speeds higher than the flutter speed, the wing vibration tends to increase, i.e., divergent oscillations take place, which may cause destruction of the wing. It should be stated that the aerodynamic forces which tend to maintain the wing vibrations exists because of the wing vibrations themselves. Thus, the flutter can be defined as an aeroelastic, self-excited vibration, in which the external source of energy is the air stream [41]. The classical type of flutter is called classical flutter and involves the coupling of several degrees of freedom of the structure. A typical example of wing classical flutter is the called wing bending-torsion flutter, which has coupling between bending and torsion mode shapes. It is known that oscillations caused by pure bending or pure torsion modes are rapidly damped. However, when there is a coupling between bending and torsion oscillations, the aerodynamic and inertial forces acquire an unstable effect. The non-classical type of flutter involves only one degree of freedom of the structure, and the stall flutter and aileron buzz are some examples. Also, it is important to remember that the flutter phenomenon, as with all aeroelastic phenomena, is highly sensitive to the structures vibration modes, depending on their natural frequencies and mode shapes.

In this thesis, again because of the complexity of the structures in study, it was decided that a commercial code will be used to calculate the flutter speed and characteristics. The program used to perform these tasks was the ZAERO, which integrates the essential disciplines required by aeroelastic design and analysis. This program im-
CHAPTER 2. COMPUTATIONAL ANALYSES

ports the solution of the free vibration previously generated by the ANSYS solution. Essentially, when this importation is made, ZAERO is importing relevant information about the structural mesh of the structure, the natural frequencies and mode shapes, the mass and stiffness matrices generated by the structural finite element method. The ZAERO program incorporates two different techniques to determine the flutter solution: the \textit{K-method} and the \textit{g-method}. The \textit{K-method} is performed at a given Mach number, $M$, and presented in terms of velocity versus frequency diagram ($V$-$f$ diagram) and velocity versus damping diagram ($V$-$g$ diagram). This method requires only a straightforward complex eigenvalue analysis of each reduced frequency, thus its solution technique is efficient and robust. However, the frequencies and velocities are computed at a given pair of Mach number and air density. This implies that the flutter boundary computed by the \textit{K-method} generally is not a "matched point" solution in that the flutter velocity, $V_f \neq M a_{\infty}$. The matched point solution can be achieved only by performing the flutter analysis at various air densities iteratively until the condition of $V_f = M a_{\infty}$ is satisfied. For $n$ structural modes, the \textit{K-method} normally provides only $n$ roots of the flutter equation. However, the number of roots could exceed the number of the structural modes. Unlike the \textit{K-method}, the \textit{g-method} potentially gives an unlimited number of roots, which could provide important physical insight of the flutter solution. More information about this methods and respective mathematical algorithms can be seen in [42]. Next, the fundamentals of aeroelasticity that ZAERO uses are briefly presented.

The aeroelastic response of an aircraft in flight is the result of the interaction of inertial and elastic structural forces, aerodynamic forces induced by the structural deformations and external disturbance forces. Thus, the equation of motion of an aeroelastic system, in terms of a discrete system, can be derived based on the equi-
librium among these forces, as follows:

\[ [m]\left\{ \frac{d^2 x(t)}{dt^2} \right\} + [k]\{x(t)\} = \{f(t)\} \]  \hspace{1cm} (2.6)

where \([m]\) and \([k]\) are, respectively, the mass and stiffness matrices generated by the structural finite element method, performed by ANSYS, \{x(t)\} is the structural deformation, and \{f(t)\} represents the aerodynamic forces applied on the structure. In general, \{f(t)\} can be divided in two parts: external forces, \{f_e(t)\}, and aerodynamic forces induced by the structural deformation, \{f_a(t)\}, i.e.:

\[ \{f(t)\} = \{f_e(t)\} + \{f_a(t)\} \]  \hspace{1cm} (2.7)

The ZAERO program considers that external forces acting on the system are provided by the user. Typical examples of external forces are atmospheric turbulence and impulsive type gusts. The generation of the aerodynamic forces is based on the theoretical prediction that requires the unsteady aerodynamic computations, and depends on the structural deformation \{x(t)\}. Thus, ZAERO program solves the following equation:

\[ [m]\left\{ \frac{d^2 x(t)}{dt^2} \right\} + [k]\{x(t)\} - \{f_a(t)\} = \{f_e(t)\} \]  \hspace{1cm} (2.8)

that can be represented by the diagram in figure 2.2.

The left hand side of the equation (2.8) is a closed-loop aeroelastic system which can be self-excited in nature. This gives rise to a stability problem of the closed-loop aeroelastic system known as flutter. If \{f_a(t)\} is a nonlinear function with respect to \{x(t)\}, the flutter analysis must be performed by a time-marching procedure solving
the following equation:

\[ [m] \dddot{x}(t) + [k] \ddot{x}(t) - \{f_a(t)\} = \{0\} \]  

However this time-marching procedure is computationally heavy, since it requires a nonlinear time-domain unsteady aerodynamic method. The ZAERO practice of flutter analysis consists in remodel equation (2.9) into a set of linear systems and to determine the flutter boundary by solving the complex eigenvalues of the linear systems. This procedure is based on the assumption of amplitude linearization, which states that the aerodynamic response varies linearly with respect to the amplitude of the structural deformation. This way, the flutter analysis becomes a eigenvalue problem. In this case, the aerodynamic system can be approximated by a linear system for which an aerodynamic transfer function (that relates the aerodynamic feedback \( f_a(t) \) with the structural deformation \( x(t) \)) can be defined. Knowing this transfer function, equation (2.9) can be transformed into the Laplace domain and results in an eigenvalue problem in terms of \( s \), i.e.:

\[
\left(s^2[m] + [k] - q_{\infty}H\left(\frac{sL}{V}\right)\right)\{x(s)\} = 0
\]
where \( q_\infty H \) represents the aerodynamic transfer function, \( q_\infty \) is the dynamic pressure, \( L \) is the reference length (is generally defined as half of the reference chord), and \( V \) is the velocity of the undisturbed flow. More details about the ZAERO approach and methods are available in [42].

### 2.2 Finite Element Analysis

As referred in the previous Section, the analysis of free vibration using the finite element method was performed in order to generate the passive wing natural frequencies and mode shapes. This study was performed using ANSYS program. The intent of the finite element solutions and flutter calculations, presented in the following Sections, is only to determine the wing natural frequencies, mode shapes, and flutter speeds in passive configuration, i.e., without the actuation of the piezoelectrics. It is only in Chapter 3 that both the passive and active wings solutions will be studied and compared with each other.

In this analysis, complete three-dimensional wings analyses were performed, i.e., all the wings components were defined using two-dimensional elements in which the thickness is given. All the wing components (see wings description in Section 3.1.2), like the two carbon fibre plates, ribs, leading and trailing edges, were defined using the SHELL93 8-node structural shell elements. This ANSYS finite element type is proper to model curved skinned components of orthotropic materials, which is the present case (see materials properties in the table 3.1).
2.2.1 Adaptive Skin Wing

The designed final wing finite element model had 10199 nodes, i.e., 10199 degrees of freedom, and in figure 2.3 it can be seen the wing final mesh.

Figure 2.3: Adaptive Skin wing finite element mesh, defined in the ANSYS program.

In terms of constrained nodes, it was considered that the nodes defining the holes of the two balsa wood ribs, one placed on the wing root and another at the distance of 20 cm relatively to the wing root, had zero linear displacements and rotations. Note that, in the real wing, those holes define the connection between the wing and the fuselage (see Section 3.1.2).

After running the ANSYS Modal Analysis task, the first ten natural frequencies and mode shapes were extracted, and the results are displayed in table 2.1 and figure 2.4, respectively. Analyzing these results, it was obvious that this wing does not have a conventional behavior in terms of mode shapes, since several shell local vibrations were found. Note that this result should be expected, since this wing does not have a conventional configuration. Conventional wings have a spar as the main stiff component, and not a stiff skin. Because of this non-conventional wing configuration, a
CHAPTER 2. COMPUTATIONAL ANALYSES

non-classical type of flutter should be expected for this wing. The mode shapes that lead with shell vibrations are: the $2^{nd}$, $4^{th}$, and $8^{th}$ modes, with shell bending, and the $7^{th}$ mode, with shell bending-torsion. This way, the $2^{nd}$, $4^{th}$, $7^{th}$ and $8^{th}$ modes are mainly characterized by the two carbon plates vibration in which the lower and upper carbon plates have deformations in opposite phases. This can be explained due to the fact that the two carbon plates are only connected with each other at some discrete points, along the leading and trailing edges. These modes are not beneficial to this study, since the main objective is to control the vibration of the total wing as a single assembly and not the lower or the upper carbon plates separately. See also Section 3.1.2 in order to read additional information about this point. In Section 3.3.1, the wind tunnel tests proved that the problem which caused these local shell vibrations was related with a few number of connections between the lower and upper carbon plates. On the other hand, the remaining modes have conventional behavior, as follows: $1^{st}$ mode is the first bending, $3^{rd}$ mode is the first bending-torsion mode, $5^{th}$ mode is the first torsion mode, $6^{th}$ mode is the second bending-torsion mode, $9^{th}$ mode is the second torsion mode, and $10^{th}$ mode is the third bending-torsion mode.

Table 2.1: Adaptive Skin wing first ten natural frequencies.

<table>
<thead>
<tr>
<th>Mode</th>
<th>Natural Frequency [Hz]</th>
</tr>
</thead>
<tbody>
<tr>
<td>1</td>
<td>16.435</td>
</tr>
<tr>
<td>2</td>
<td>30.944</td>
</tr>
<tr>
<td>3</td>
<td>38.512</td>
</tr>
<tr>
<td>4</td>
<td>53.153</td>
</tr>
<tr>
<td>5</td>
<td>56.663</td>
</tr>
<tr>
<td>6</td>
<td>63.723</td>
</tr>
<tr>
<td>7</td>
<td>76.256</td>
</tr>
<tr>
<td>8</td>
<td>88.284</td>
</tr>
<tr>
<td>9</td>
<td>90.348</td>
</tr>
<tr>
<td>10</td>
<td>93.023</td>
</tr>
</tbody>
</table>
CHAPTER 2. COMPUTATIONAL ANALYSES

Figure 2.4: Adaptive Skin wing first ten mode shapes.
2.2.2 Adaptive Spar Wing

As in the adaptive skin wing, all the adaptive spar wing components (see Section 3.1.2), i.e., the carbon fibre beam, ribs, leading and trailing edges, were defined using the SHELL93 8-node structural shell elements. The final wing finite element model had 4822 nodes. In figure 2.5 it can be seen the wing final mesh.

![Adaptive Spar Wing finite element mesh, defined in the ANSYS program.](image)

Using the ANSYS Modal Analysis task, the first five natural frequencies and mode shapes were extracted, and the obtained results are displayed in table 2.2 and figure 2.6, respectively. Since this passive wing has a conventional configuration,

<table>
<thead>
<tr>
<th>Mode</th>
<th>Natural Frequency [Hz]</th>
</tr>
</thead>
<tbody>
<tr>
<td>1</td>
<td>19.779</td>
</tr>
<tr>
<td>2</td>
<td>32.576</td>
</tr>
<tr>
<td>3</td>
<td>46.660</td>
</tr>
<tr>
<td>4</td>
<td>84.118</td>
</tr>
<tr>
<td>5</td>
<td>124.303</td>
</tr>
</tbody>
</table>

Table 2.2: Adaptive Spar wing first five natural frequencies.
Figure 2.6: Adaptive Spar wing first five mode shapes.
i.e., with a main beam, internal ribs and a non-stiff skin, conventional flutter was expected. Analyzing the results, no shell local vibrations were found, and the following natural modes were determined: 1\textsuperscript{st} mode is the first bending, 2\textsuperscript{nd} mode is a “swing” mode (bending in the wing plane), 3\textsuperscript{rd} mode is the first torsion, 4\textsuperscript{th} mode is a bending-torsion mode, and 5\textsuperscript{th} mode is the second torsion mode. Since this wing has a conventional configuration, classical flutter (referred in Section 2.1) is expected, i.e., wing bending-torsion flutter.

2.3 Flutter Analysis

2.3.1 Adaptive Skin Wing

After obtaining the ANSYS results, the ZAERO program was used to perform the wing aeroelastic study in terms of flutter. ZAERO program imports the solution of the free vibration generated by the ANSYS program. The flutter analysis was performed using a constant air density of 1.225 Kg/m\textsuperscript{3}, and flutter speeds between 15 and 250 m/s were calculated.

Table 2.3 shows the flutter results obtained using the \textit{g-method}. The first three columns on the left, reading from the left to the right, show the first six flutter modes, their respective speeds and frequencies. The column on the right displays the natural modes contributions for the flutter occurrence, in each flutter mode. The results in table 2.3 state that:

- the 1\textsuperscript{st} flutter mode, at 44.87 m/s, is essentially related with the wing 7\textsuperscript{th} natural mode;
- the 2\textsuperscript{nd} flutter mode, at 60.44 m/s, happens because of the coupling between the wing 3\textsuperscript{rd} and 2\textsuperscript{nd} natural modes;
CHAPTER 2. COMPUTATIONAL ANALYSES

- the 3\textsuperscript{rd} flutter mode, at 114.84 m/s, is related with the coupling of the wing 10\textsuperscript{th} and 9\textsuperscript{th} modes;
- the 4\textsuperscript{th} flutter mode, at 131.92 m/s, occurs because of the strong coupling among the 9\textsuperscript{th}, 8\textsuperscript{th} and 10\textsuperscript{th} natural modes;
- the 5\textsuperscript{th} flutter mode, at 144.39 m/s, is dependent of the 1\textsuperscript{st}, 7\textsuperscript{th}, 3\textsuperscript{rd}, 2\textsuperscript{nd} and 9\textsuperscript{th} wing natural modes;
- the 6\textsuperscript{th} flutter mode, at 194.97 m/s, occurs when the coupling among 7\textsuperscript{th}, 6\textsuperscript{th} and 3\textsuperscript{rd} natural modes takes place.

Table 2.3: Adaptive Skin wing flutter results, using the \textit{g-method}.

<table>
<thead>
<tr>
<th>Flutter Mode</th>
<th>Speed [m/s]</th>
<th>Frequency [Hz]</th>
<th>Natural Modes contribution [%]</th>
</tr>
</thead>
<tbody>
<tr>
<td>1</td>
<td>44.87</td>
<td>71.91</td>
<td>1)0.28, 2)1.49, 3)2.81, 4)0.12, 5)5.48, 6)15.53, 7)100.00, 8)0.47, 9)3.16, 10)0.16</td>
</tr>
<tr>
<td>2</td>
<td>60.44</td>
<td>32.02</td>
<td>1)5.04, 2)61.78, 3)100.00, 4)12.36, 5)2.89, 6)23.99, 7)16.50, 8)0.042, 9)1.51, 10)1.18</td>
</tr>
<tr>
<td>3</td>
<td>114.84</td>
<td>65.05</td>
<td>1)12.38, 2)17.80, 3)13.93, 4)37.13, 5)30.36, 6)26.29, 7)56.06, 8)26.71, 9)92.82, 10)100.00</td>
</tr>
<tr>
<td>4</td>
<td>131.92</td>
<td>86.96</td>
<td>1)12.84, 2)6.81, 3)7.64, 4)0.25, 5)0.36, 6)43.72, 7)16.48, 8)99.85, 9)100.00, 10)80.37</td>
</tr>
<tr>
<td>5</td>
<td>144.39</td>
<td>31.46</td>
<td>1)100.00, 2)92.97, 3)96.61, 4)26.83, 5)56.99, 6)61.73, 7)99.64, 8)18.94, 9)85.99, 10)63.26</td>
</tr>
<tr>
<td>6</td>
<td>194.97</td>
<td>44.71</td>
<td>1)1.84, 2)15.64, 3)88.71, 4)14.33, 5)26.80, 6)91.87, 7)100.00, 8)25.22, 9)4.76, 10)1.13</td>
</tr>
</tbody>
</table>
Additionally, the $V-f$ (velocity versus frequency) and $V-g$ (velocity versus damping) graphics were obtained, as shown in figures 2.7 and 2.8. Finally, the figure 2.9 shows the obtained six flutter modes shapes.

![Figure 2.7: Adaptive Skin wing $V-f$ graphic.](image)

Analyzing the $V-g$ graphics in figure 2.8, one can conclude the following: the wing natural mode that become unstable first is the 7th mode, at 44.87 m/s speed; then is the 3rd mode, at 60.44 m/s speed; after that, it follows the 9th, 10th, 4th and 6th modes at, respectively, 114.84 m/s, 131.92 m/s, 144.39 m/s and 194.97 m/s speeds. Note that these found flutter modes do not have all the same intensity. In fact, the first flutter mode to appear, related with the wing 7th natural mode, as a smoother behaviour and smaller unstable damping values (i.e., positive damping values smaller than 0.025) when compared with other flutter modes. For instance, the second flutter mode, related with the wing 3rd natural mode, has an abrupt behaviour since high unstable damping values (i.e., positive damping values higher than 0.025) are suddenly reached.
Figure 2.8: (a) Adaptive Skin wing V-g graphic; (b) V-g graphic zoom near the zero damping.
Figure 2.9: Adaptive Skin wing flutter modes shapes.
This means that the flutter mode related with the 3rd natural mode is a more abrupt and dangerous phenomenon. Of course, one should not forget that the flutter mode related with the 7th natural mode is the one which first occurs.

In figure 2.9 it is possible to confirm that all the obtained flutter modes are related with shell vibrations. The first and third flutter modes shapes have, essentially, shell vibration. The second flutter mode shape also have bending, the fourth have torsion, and the fifth and sixth flutter modes have bending-torsion.

### 2.3.2 Adaptive Spar Wing

As referred in Section 1.3, the flutter speed of wings with conventional configuration, i.e. with a main spar and ribs, was calculated using a method presented by [39]. This method is based on an empirical investigation and calculates the torsional flutter of gliders and small aircrafts wings. First, it is necessary to estimate the wing first torsion frequency using vibration tests. However, the wing first torsion frequency was already calculated in the ANSYS analysis presented in Section 2.2.2, and its value is $f_T = 46.66\,Hz$. Thus, applying the equation 2.11 it is possible to estimate the wing flutter speed. Later, in Chapter 3, the flutter speed of this wing will be calculated using the wing experimental tests results.

$$V_T = 1.2 c_{0.7} f_T \sqrt{\Lambda}$$

where $V_T$ is the flutter speed in m/s, $c_{0.7}$ is the wing chord length at 0.7b/2 in m, in which b is the wing span, $f_T$ is the wing first torsion frequency in Hz, and $\Lambda$ is the wing aspect ratio.

Since the wing is rectangular, i.e. has a constant chord, thus $c_{0.7} = c = 0.33m$. The wing aspect ratio is $\Lambda = 7.273$ (see table 4.1). The equation 2.11 gives an
empirical estimation of flutter speed for wings with aspect ratios between 6 and 9, which is the present case. Using the referred values of \( c_{0.7} \), \( f_T \) and \( \Lambda \) in equation 2.11, the wing estimated flutter speed is \( V_T = 49.83 \text{ m/s} \).
Chapter 3

Wind Tunnel Tests

3.1 Experimental Apparatus

The hardware involved in these wind tunnel tests is described in four Sections: the wind tunnel, the tests articles, the digital controller, and electronic equipment.

3.1.1 Wind Tunnel

The RPV model with the adaptive wings was tested in the wind tunnel installed at Portuguese Air Force Academy Aeronautical Laboratory, shown in figure 3.1. The wind tunnel is a closed circuit horizontal wind tunnel with a maximum operating velocity of 70m/s, with the air stream temperature control. The test section is 1.3m x 0.8m and 2m long, and can be used in an open or closed configuration. A uniform flow velocity with less than 0.8% in pressure variation can be obtained in a cubic zone 1.1m x 0.6m x 1.4m. The ventilator has a 8 blades fan with 1.6m diameter. The shaft has a 0.63m diameter and rotates at maximum speed of 1600rpm, producing a maximum airflow of 72.8m³/s, with 88% efficiency.
CHAPTER 3. WIND TUNNEL TESTS

The RPV has a total span of 2.40m and the wind tunnel tests were performed with half of the RPV model (1.20m span), in open section configuration. The figure 3.2 shows the test article mounted in the wind tunnel section.

3.1.2 Tests Articles

The RPV with active wings wind tunnel model consists of four main components: the RPV fuselage to mount the half wing, the half wings, the piezoelectric sensors and the piezoelectric actuators. There were three driving factors in the design of the RPV model: half RPV model had to fit inside the wind tunnel section, the wing had to flutter within the wind tunnel envelope and had to have surfaces (in the skin for the active skin wing, and in the main spar for the active spar wing) on which piezoelectric sensors and actuators could be mounted.

The RPV fuselage, shown in figure 3.3, has a 0.4m span, 1.4m in length, and 0.2m in width. It has two fixation points to perform the connection between the fuselage and the half wing. The RPV fuselage is settled in a vertical stationary panel that can
CHAPTER 3. WIND TUNNEL TESTS

Figure 3.2: Photograph of the RPV model with active wing mounted in the wind tunnel section.

rotate in order to allow the variation of the RPV angle of attack. Both wings (active skin and active spar) have 2m of total length, which added with the 0.4m of fuselage span gives the total RPV span of 2.4m, and 0.33m chord. The RPV fuselage and half wing are joined together by means of a main rod tube and an auxiliary leading rod, as shown in figure 3.4.

The active skin wing has a modified NACA0012 airfoil. The wing leading edge skin is made of fibre glass 0.75mm thick, with eight distributed balsa wood leading edge ribs 2mm thick along the wing span, to assure that the leading edge shape is maintained. The wing trailing edge is balsa reinforced and corresponds to the wing flaperon. The active structural elements of the wing are two flat sheets of unidirectional 8-layered carbon fibre, with a total thickness of 1.104mm (the upper and lower surfaces of the wing), used as the skin of the wing. The carbon layers are oriented along the span direction. The leading and trailing edges are both attached to the carbon fibre flat sheets using fibre glass brackets, as can be seen in figure 3.5. The whole wing has only three integral ribs: one in the wing root, one at the distance of 20cm relatively to the wing root (important to make the connection with
CHAPTER 3. WIND TUNNEL TESTS

Figure 3.3: Photograph of the RPV wind tunnel model fuselage.

Figure 3.4: Schematic view of the interconnection between the RPV fuselage and wing (not to scale).
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Table 3.1: Wing material properties.

<table>
<thead>
<tr>
<th>Property [Unity]</th>
<th>Carbon Fibre</th>
<th>Balsa Wood</th>
<th>Fibre Glass</th>
</tr>
</thead>
<tbody>
<tr>
<td>$E_x$ [GPa]</td>
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<td>5.295</td>
<td>38.1</td>
</tr>
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<td>$E_y$ [GPa]</td>
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<td>38.1</td>
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<td>$E_z$ [GPa]</td>
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<td>6.8</td>
</tr>
<tr>
<td>$G_{yz}$ [GPa]</td>
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<td>3.026</td>
<td>4.4</td>
</tr>
<tr>
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<td>-</td>
</tr>
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<td>0.488</td>
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</tr>
<tr>
<td>$\nu_{yz}$</td>
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</tr>
<tr>
<td>$\nu_{xz}$</td>
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<td>0.229</td>
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</tr>
<tr>
<td>$\rho$ [Kg/m$^3$]</td>
<td>1550</td>
<td>176.2</td>
<td>1800</td>
</tr>
</tbody>
</table>

Figure 3.5: Schematic view of the wing airfoil shape and components.

the fuselage), and another in the wing tip. These ribs are made of balsa wood 3mm thick. The materials properties of the active skin wing components are shown in table 3.1.

In the wing lower surface the fibre glass brackets are glued to the carbon flat sheet. In the wing upper surface, those components are connected by bolts, to permit the posterior disassembly of the wing. The figures 3.6 and 3.7 show, respectively, the internal parts of the wing without and with the piezoelectric mounted in the skin. The piezoelectric sensor was mounted near the wing root, in the internal part of the lower carbon fibre plate. The four piezoelectric actuators were mounted in the internal part
CHAPTER 3. WIND TUNNEL TESTS

Figure 3.6: Photograph of active skin wing internal assemblage (without piezoelectrics).

Figure 3.7: Photograph of active skin wing with piezoelectric sensors and actuators mounted in the wing lower surface.

of the two carbon fibre plates. The sensor and actuators location scheme is in figure 3.8. The location of the piezoelectric components was decided from the finite element analysis of the vibration modes, which revealed the points with larger strain values.

In this wing, is important that the two carbon plates have similar displacement. For instance, when the upper plate is deflecting down, the lower plate should also be deflecting down. This way, when the wing is deflecting down, the upper plate extends and the lower plate contracts. In this case, the two carbon plates work like "active skins" when the piezoelectric actuators placed in the upper plate contract and the piezoelectric actuators placed in the lower plate extend, forcing the wing to deflect
Figure 3.8: Schematic view of the piezoelectric sensor (small patch) and actuators (big patches) placement inside the active skin (gray panel) wing.

Figure 3.9: Photograph of the RPV model with active skin wing in the wind tunnel.
in the opposite direction. This process can be seen in figure 3.10(a). On the other hand, when the wing is deflecting up, the upper plate contracts and the lower plate extends. In this case, the two carbon plates work like "active skins" when the upper piezoelectric actuators extend and the lower piezoelectric actuators contract, forcing the wing to deflect down. In figure 3.10(b) it can be seen an schematic view of this process. The figure 3.9 shows the RPV with active skin wing ready to start the wind tunnel tests.

Like the active skin wing, the active spar wing has 1m of span, giving 2.4m of span to the RPV. The wing has a FX63-137 airfoil, which is one of the most desirable airfoils for high-lift low Reynolds models. This wing has a hollow squared beam, which has 1.104\( \text{mm} \) thickness carbon fibre upper and lower horizontal surfaces (caps), and 3\( \text{mm} \) thickness balsa wood vertical surfaces (webs), as shown in figure 3.11. The active elements are the two horizontal carbon fibre plates, which has the same material properties as the carbon fibre used for the active skin wing surfaces. The ribs and the D cell that form the leading edge are made of balsa wood with 2\( \text{mm} \) thickness. The trailing edge is made of high density balsa wood. The materials properties of the active spar wing components are in the table 3.1. The figures 3.12 and 3.13 show, respectively, the internal parts of the wing without and with the piezoelectric sensors and actuators mounted in the main beam. Like in the active skin wing, the piezoelectric sensor was mounted near the wing root, in the internal part of the lower carbon fibre plate, and the four piezoelectric actuators were mounted in the internal part of the two carbon fibre plates. The sensor and actuators location scheme is shown in figure 3.14. Also, the location of the piezoelectric components was decided from the previous finite element analysis of the vibration modes.

In this wing, since the caps are connected with each other by the webs, the hollow squared beam has an overall desired movement, deflecting up and down. When the
Figure 3.10: Scheme of the active skin wing control vibration process (the rectangular patches represent the piezoelectric actuators and the gray panels the carbon fibre plates).
hollow squared beam is deflecting down, the upper cap extends and the lower cap contracts. In this case, the hollow squared beam works like an "active spar" when the piezoelectric actuators placed in the upper cap contract and the piezoelectric actuators placed in the lower cap extend, forcing the spar to deflect in the opposite direction. On the other hand, when the hollow squared beam is deflecting up, the upper cap contracts and the lower cap extends. In this case, the beam works like an "active spar" when the upper piezoelectric actuators extend and the lower piezoelectric actuators contract, forcing the spar to deflect down. The process is similar with the one explained for the active skin wing, but now the spar vibration control is being performed instead of controlling the vibration of the skin. Figure 3.2 shows the RPV with active spar wing ready to start the wind tunnel tests.

The piezoelectric sensors used for this study were the lead-zirconate-titanate piezoelectric patches, shown in the figure 3.15. This sensor type measures strain changes, as extensometers do, and are relatively easy to glue to a surface. The piezoelectric sensors are $22 \times 38\text{mm}$ patches that become electrically charged when subjected to a mechanical strain, producing a variable $\pm 2.5V$ AC electrical signal.

The piezoelectric actuators used were the ACX QuickPack 40W, shown in the figure 1.4, that are rectangular patches of $102 \times 40\text{mm}$. This actuator is built through
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Figure 3.12: Photograph of active spar wing internal assemblage (without piezoelectrics).

Figure 3.13: Photograph of active spar wing with piezoelectric sensors and actuators.

a proprietary manufacturing process that shields the piezoelectric material in a protective polyamide coating with pre-attached electrical leads and quick connectors, improving electrical isolation and adding protection against breakage during assembly and resistance to micro cracks during operation. The manufacturer guarantees a proper frequency response of the actuator for input signals with frequencies between $1\text{Hz}$ and $20kHz$, and the excitation signals can vary from 0 to $\pm 200V$. The figure 3.16 shows the most relevant actuator characteristics.
Figure 3.14: Schematic view of the piezoelectric sensor (small patch) and actuators (big patches) placement in the active spar (gray beam) wing.

Figure 3.15: Lead-zirconate-titanate piezoelectric sensors.
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Figure 3.16: ACX QuickPack 40W actuator characteristics.
CHAPTER 3. WIND TUNNEL TESTS

In figure 3.16, the first plot shows the peak-to-peak strain versus excitation voltage curve, and the second shows the relation between peak-to-peak strain and peak-to-peak force, for each excitation voltage. For instance, an actuator excitation voltage of 140V can induce a 350με of strain at the actuator if it is not constrained, making the resulting strain value conditioned to the maximum force that the actuator can transmit, which is 65lbs in this case.

3.1.3 The Controller

The digital control module used was a dSPACE MicroAutoBox 1401/1501, shown in the figure 3.17. The DS1401 base board is based on the PowerPC 603e processor that forms the main processing unit of the MicroAutoBox and weights 1022g. The DS1401 also includes a high-speed serial interface with a 4-Mbyte-communication memory. The DS1501 I/O board integrates the following components:
- high performance Analog-Digital and Digital-Analog Converters units (A/D and D/A);
- a digital I/O subsystem based on the Motorola 688336 micro controller, not used for this work;
- a Controller Area Network subsystem of the I/O board, based on the Siemens SAB 80C167 micro controller, primarily used for working with expansion boxes in asynchronous working modes, not used for this work.

The DS1401 base board and the DS1501 I/O board are connected via an internal bus. Both boards need a single DC power supply of 12V. Communication with the host PC is done through a dedicated DS815 PCMCIA board. This allows not only the uploading of programs to the DS1401, but also the real-time monitoring and recording variables and their consequent download [43].
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Figure 3.17: Photograph of the dSPACE MicroAutoBox 1401/1501.

The design of real-time applications to the DS1401 was made using MATLAB’s Simulink and a dedicated toolbox provided by dSPACE. This toolbox permits access to internal DS1401/1501 board components, such as A/D, D/A and memory units. After building the Simulink model, the real-time Workshop is used to generate C language source code for the real-time application and start the real-time Interface. The real-time Interface automatically connects MATLAB, Simulink and the real-time Workshop with DS1401/1501 real-time system. The D/A and A/D converters timing and synchronization is selected and guaranteed by the real-time Workshop step-time (limited by the converter settling times already stated above), and thus providing the seamless implementation of the Simulink model even using additional toolboxes and continuous time/discrete-time hybrid systems. The dSPACE’s ControlDesk software enables, through a fully graphical interface, not only to visualize, record and transfer data from the DS1401 board to the host PC (both in binary or .mat format files), but also, real-time parameter tuning, like offset and gain values, and data capturing and recording parameters [44].
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3.1.4 Electronic Equipment

In terms of electronic equipment, there were used the following four components: an amplifier, a signal conditioning circuit, a power supply and voltage regulators.

The used amplifier is a modified Sensor Technology SA-10 High-Voltage Power Amplifier, shown in the figure 3.18, that can be used as two individual ground-referenced amplifiers each one with a 15 times gain. The total input voltage is limited to approximately \(-/+/9V\) setting the output to a maximum voltage swing of \(-/+/140V\). The maximum output current per channel is internally limited to \(50mA\). The amplifier slew rate is \(3.8V/s\). One of the foremost advantages of the modified amplifier is its weight of only \(100g\) and the small dimensions of \(1'' \times 3'' \times 5''\), that allows its usage in the RPV.

The need of a signal conditioning arises from the fact that the output voltage signals from the sensors \((-2.5V \text{ to } +2.5V)\) are incompatible with the input voltage signals from the DS1501 \((0V \text{ to } +5V)\). Also, the voltage signals from DS1501 \((0V \text{ to } +4.5V)\) need to be modified to fit SA-10 amplifier voltage input signals \((-/+/9V)\), in order to achieve the amplifier's maximum output voltage to feed the actuators \((-/+/140V)\). On the other hand, the signal before the SA-10 amplifier must have the DC component removed. The option of trying to use the maximum output voltage
of the actuators is justified because it is required the maximum output energy with the minimum controller gain, in order to achieve high strain rates to control higher frequency vibrations. The build conditioning circuit was based in the one designed by [45] and was mounted before DS1501, as shown in figure 3.19. Since it was decided to limit the operations performed by hardware components (although they are faster and more efficient than software components) because of volume and weight restrictions of the UAV, the main responsibility of performing filtering and control operations was left to software modules running on the PowerPC 603e processor of the dSPACE. The main piece in the signal conditioning circuit is the 741 Operational Amplifier. It needs two power signals (+Vcc and -Vee). Normally +Vcc = Vee and lies between 5 and 15V. The value of +Vcc = Vee = 12V was chosen for this work, to be compatible with dSPACE and SA-10 power supply inputs.

Although a lot of testing in the wind tunnel was conducted with the energy supply provided by the electrical network, the final wind tunnel tests were performed with two batteries in serial connection, shown in figure 3.20. The positive terminal of the first battery and the negative terminal of the second battery provide the reference
signal for all equipments. The first battery is a PowerLite 14.4V, 2000mAh lithium-ion pack. It is used to provide voltage and current to the −12V voltage regulator, in order to feed the −12V amplifier and signal conditioning circuit inputs. Since it feeds the negative part of the circuits, it needs to provide low output current values (0.1A), making the endurance rise to over 2 hours and 30 minutes in continuous working mode. This pack weighs 205g. To feed the +12V input of the amplifier, signal conditioning circuit and DS1401 (the most demanding component in terms of current), a 16.8V NEXcell, 2200mAh and 428g pack is used, connected to a voltage regulator. To feed these components, this battery has to continuously provide 1.2A, and more than 2.05A peak, due to the transient input peak in the dS1401 start-up. Therefore, this is the critical electrical component in terms of endurance.

Since a stable voltage power supply is a fundamental requirement of all components, it was decided to include voltage regulators in the hardware components, between the two batteries and the signal conditioning circuit. The selected components were the MC7912 and LM317K. The MC7912 component provides, on its output gate, a stable voltage of −12V. It allows current loads up to 1A and is manufactured by Samsung GmbH. The LM317K provides, on its output gate, a stable voltage of +12V.
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It works with current loads up to 3A and is manufactured by Thomson GmbH. As this component deals with relatively high values of power, it was necessary to adapt one heat exchanger, as shown in figure 3.19.

3.2 Tests Objectives and Procedures

The general objective of the wind tunnel tests is to demonstrate the advantage of using piezoelectric sensors and actuators to actively control wing structure vibrations. First, the tests were made with the wing in free vibration, i.e., without the piezoelectrics control. Then, similar tests were made but with active piezoelectric control working. It was verified that, with the active control working, the structure damping has increased significantly. In these tests the goal is to reduce the wing vibrations amplitude (and corresponding stress levels) and increase its damping. Second, wing flutter tests were performed. In these tests the wind speed was successively increased, until the interaction between aerodynamic, inertial and structural forces became unstable. Some of these tests were destructive. In both wings, as explained in Section 3.1.2, the piezoelectric actuators were mounted near the wing root. This piezoelectric placement configuration generates, when the piezoelectric control is working, a bending moment that counterbalance the wing vibrations along its span (see figure 3.10). In other words, the active wing concept used consists in controlling flutter vibrations by actuating the wings first bending mode. The goal is to prevent the interaction between the first bending mode and the first torsion mode, preventing the wing flutter and thus increasing the wing resonance frequency by increasing the damping, when compared with the passive wing.

The entire hardware system used for wind tunnel and flight test is shown in figure 3.21. This figure highlights the connections between all hardware blocks, which were
developed and assembled in a modular way. These connections are detailed in the figures legend and include power, signal and digital information flux. All blocks were disposed by functional levels (batteries, voltage regulators, sensors, signal conditioning, control, amplifier and actuators), that were exposed in more detail in Section 3.1. The most important part of this scheme are the modules between sensors and actuators. Thus, monitoring the shape of the wing in real-time, the piezoelectric sensors measurements provide real-time information, required for the feedback wing deformation control, which will determine the operation of the piezoelectric actuators. In other words, the piezoelectric sensors produce a variable ±2.5V electrical signal that is conditioned and sent to the data acquisition system (analog-to-digital converter) of the controller. Then, the controller output passes thought a digital-to-analog converter, and the analog signal is once more conditioned and sent to high-voltage power amplifier. The output signal of the amplifier is used to drive the piezoelectric actuators.

In terms of control model, the classical PID controller would be a possible solution to the problem [46]. Though, as the sensors and actuators do not react to position and do not follow a reference position (they react to strain changes) the tuning of the PID parameters would not be easy, assuming that they would change with speed. Prior studies [47, 48] demonstrated that simple proportional control law, with pre-filter, produced very good results in real-time conditions. Taking all these facts in consideration, the control approach selected was Proportional control, and it was developed with the goal of working in flight. The actuators will react to the provided sensors signal multiplied by a constant. This way, the conceptual approach followed in order to design the control model is presented in the figure 3.22. The plant to be controlled is the active wing (active skin or active spar). The strain rate signal, measured by the sensors, is applied to the controller and the loop is closed by the
Figure 3.21: Scheme of the complete hardware system for the wind tunnel and flight tests.
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Figure 3.22: Block diagram of conceptual active wing control model.

Actuators, which induce a strain, limited by the available force, to the portion of skin/spar under their location, forcing the skin/spar to move.

The control model used for wind tunnel tests was designed in Simulink and is shown in figure 3.23. The first block of the control model is the A/D converter output, which input is the signal of the sensor (after the signal conditioning that adds $2.5V$ to the output sensor signal of $\pm 2.5V$). A constant value of 0.5 ($5V = 1$) is subtracted to this signal to remove the DC component, to obtain again a $\pm 2.5V$ signal. The signal is then filtered and multiplied by two gain values (one for large adjustments and one for small adjustments) and is separated in two: the positive component goes to D/A channel 1 and the negative part, after being inverted, goes to D/A channel 2. The saved signals are controller input without filtering, controller input after filtering and controller output.

As presented in Section 2.3.1, the system identification using finite element analysis revealed that the most important vibration mode for the first flutter (occurring at $44.87m/s$) of the active skin wing was the seventh mode at $71.9Hz$. Also, since
CHAPTER 3. WIND TUNNEL TESTS

Figure 3.23: Diagram representing the control model implemented for wind tunnel tests.

the controller is working in order to control bending vibrations, the first mode at 16.4\,Hz (first bending mode) is very important. For the active spar wing (see Section 2.3.2), the revealed most important mode for the flutter occurrence was the third mode (i.e., first torsion mode) at 46.7\,Hz. Additionally, the first mode at 19.8\,Hz (first bending mode) and the second mode at 32.6\,Hz (second bending mode) are also very important, since the controller is working in order to control bending vibrations. Thus, the system bandwidth is a frequency range between 16.4 and 71.9\,Hz for the active skin wing, and between 19.8 and 46.7\,Hz for the active spar wing. According to Nyquist sampling theorem, the sample rate must be at least higher than twice the highest frequency of the signal to be sampled to avoid the aliasing error. Empirical considerations recommend that one should use a sampling rate five times the system bandwidth upper limit. Thus, in this work a sample rate of 1000\,Hz (0.001s step time) was used to have enough samples and to allow a good frequency discrimination. This value was considered not high enough to generate over-sampling errors, and it did not trouble the A/D and D/A converters. In terms of filtering, its main objective is to cut the high frequency noise. After some tests, it was decided that the filter to be
used would be a second order filter with a cutting frequency of $45\,Hz$ for the active spar wing, and a second order filter with $85\,Hz$ of cutting frequency for the active skin wing. Note that, for the active spar wing, the first bending mode has a frequency of $19.8\,Hz$ and the controller is acting on this mode shape, since the filter is not cutting this frequency. The second order filter with the cutting frequency of $45\,Hz$ also allows that the wing first torsion mode is considered, thus the flutter can be studied. The same points were considered in the active skin wing: the second order filter with the cutting frequency of $85\,Hz$ allows the controller to work in the first bending mode (at $16.4\,Hz$) and in the main frequency related with the first flutter mode (at $71.9\,Hz$).

The wings were tested at the speeds between $10$ and $37.5\,m/s$. These tests were made with the characteristic residual turbulence of the wind tunnel ($2\%$) and a $5\,Hz$ vibration caused by the support model empennages flexibility. With the angle of attack of the airplane nearly calibrated for each speed, the signals of displacement of the wing were recorded for each flying condition. With the displacement values given by the sensor signal, the average and maximum displacements of the passive and active wings were used to calculate the improvements of this technology, shown in Sections 3.3 and 3.4. It is important to notice that the recorded signals correspond to the sensors output voltage, directly proportional to the wing vertical displacement. During tests, the values of proportional gain were tuned watching the controllers output signal and the sensors signal in ControlDesk experiment. The maximum gain that did not saturated the actuators, and managed to reduce the amplitude of controllers input signal was the one chosen to each specific speed.
3.3 Adaptive Skin Wing Tests

3.3.1 Vibration Tests

The RPV with the active skin wing was tested in the wind tunnel at several operating speeds, and with the controller in open and closed loop in order to quantify the benefits of this technology in comparison to a normal passive wing. The signals of displacement of the wing were recorded for each flying condition. After this, the results of these displacement signals (average and maximum values) with control ON (active wing) and control OFF (passive wing) were studied.

For this wing, the results of the displacement signals with control ON and control OFF are shown in the table 3.2. Using the results shown in this table, the improvements of using the active wing were calculated. These improvements are shown in table 3.3. Note that both the average displacement and the maximum displacement are generally lower with the active wing than with the passive one, making the active wing an improved wing version compared with the passive wing, in the majority of flight conditions. The results shown in table 3.2 can also be graphically seen in figures 3.24 and 3.25, where the improvements of the active skin wing can be verified more explicitly.

3.3.2 Damping Analysis

In order to have a deeper understanding of the aeroelastic characteristics of the wing with the controller on, the study of damping at several key frequencies was performed. The damping influence was studied with two objectives: to quantify the amount of damping imposed by the controller in vibrations induced by the tail and to quantify the damping in the vibration modes associated with the wing flutter. All damping
Table 3.2: Average and maximum displacement values for the passive and active skin wing configurations.

<table>
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<tr>
<th>Speed [m/s]</th>
<th>Average displacement</th>
<th>Maximum displacement</th>
<th>Average displacement</th>
<th>Maximum displacement</th>
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</thead>
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<td>25</td>
<td>0.1264</td>
<td>0.279</td>
<td>0.1193</td>
<td>0.2765</td>
</tr>
<tr>
<td>30</td>
<td>0.1235</td>
<td>0.296</td>
<td>0.1123</td>
<td>0.2573</td>
</tr>
<tr>
<td>35</td>
<td>0.0267</td>
<td>0.1358</td>
<td>0.0283</td>
<td>0.1174</td>
</tr>
<tr>
<td>37.5</td>
<td>0.0263</td>
<td>0.1133</td>
<td>0.0249</td>
<td>0.1041</td>
</tr>
</tbody>
</table>

Table 3.3: Displacements improvements of the active skin wing compared with the passive skin wing.

<table>
<thead>
<tr>
<th>Speed [m/s]</th>
<th>Improvement of the average displacement [%]</th>
<th>Improvement of the maximum displacement [%]</th>
</tr>
</thead>
<tbody>
<tr>
<td>10</td>
<td>22.11</td>
<td>11.16</td>
</tr>
<tr>
<td>15</td>
<td>3.41</td>
<td>3.60</td>
</tr>
<tr>
<td>20</td>
<td>13.13</td>
<td>0.00 (no improvement)</td>
</tr>
<tr>
<td>25</td>
<td>5.62</td>
<td>0.90</td>
</tr>
<tr>
<td>30</td>
<td>9.07</td>
<td>13.07</td>
</tr>
<tr>
<td>35</td>
<td>-5.99 (no improvement)</td>
<td>13.55</td>
</tr>
<tr>
<td>37.5</td>
<td>5.32</td>
<td>8.12</td>
</tr>
</tbody>
</table>
CHAPTER 3. WIND TUNNEL TESTS

Figure 3.24: Average displacements of both passive and active skin wing configurations, in the wind tunnel tests.

Figure 3.25: Maximum displacements of both passive and active skin wing configurations, in the wind tunnel tests.
values were calculated using the FFT of the motion of the wing with and without control, throughout the speed range between 10 m/s and 37.5 m/s.

The aerodynamic shape of the tail in conjunction with the vibration modes of the aircraft fuselage generate a steady vibration at around 5 Hz, which shifts to almost 10 Hz as the speed increases. After validating this mode shapes, the signals of the sensors of the wing were transformed to frequency versus amplitude in order to calculate the damping of the wing at several speeds, in the active and passive mode. Like can be seen in figure 3.26, the damping value is the ratio between the difference of the two half-power points amplitude and the natural frequency point amplitude [49]. After analyzing the FFT peak of the vibration imposed by the tail to the wing, the damping values were calculated and the results are shown in table 3.4 and figure 3.27. The tail vibration peak started at 5 Hz at 10 m/s and shifted to 10 Hz at 30 m/s.

The results displayed in figure 3.27 show that the damping values of the wing in the active mode are higher than in the passive mode in almost all the flying
CHAPTER 3. WIND TUNNEL TESTS

Table 3.4: Damping of the active skin wing due to tail vibration, tested at 10, 15, 20, 25, 35 and 37.5 m/s.

<table>
<thead>
<tr>
<th>Speed [m/s]</th>
<th>Passive Wing Damping</th>
<th>Active Wing Damping</th>
</tr>
</thead>
<tbody>
<tr>
<td>10</td>
<td>0.5</td>
<td>0.571</td>
</tr>
<tr>
<td>15</td>
<td>0.625</td>
<td>0.722</td>
</tr>
<tr>
<td>20</td>
<td>0.75</td>
<td>0.767</td>
</tr>
<tr>
<td>25</td>
<td>0.3</td>
<td>0.3</td>
</tr>
<tr>
<td>35</td>
<td>0.857</td>
<td>1.143</td>
</tr>
<tr>
<td>37.5</td>
<td>0.571</td>
<td>0.714</td>
</tr>
</tbody>
</table>

Figure 3.27: Damping curves of both passive and active skin wing configurations, due to the tail vibration.
conditions. These results demonstrate that the controller is improving the aeroelastic characteristics of the wing and decreasing forced vibrations induced to the structure.

### 3.3.3 Flutter Analysis

The flutter tests were conducted in order to verify the predicted flutter speed of the wing working in the passive mode of $44.87 \text{m/s}$ (calculated in Section 2.3.1), and then to investigate the maximum attainable speed increase without fluttering the wing with the controller working. The flutter speeds (with controller on and off) were determined experimentally by calculating the damping of the vibration modes that will induce the first flutter to appear (mainly the seventh mode at $71.9 \text{Hz}$). Using the FFT analysis on the controllers input signal amplitude of the wing, the damping values of these modes were obtained, as shown in table 3.5. Then, a plot of damping versus speed was designed and a second order polynomial was passed through the points in order to find the zero damping speed, which corresponds to the flutter speed, as shown in figure 3.28. This method of finding the flutter speed was validated by application in previous works [48].

<table>
<thead>
<tr>
<th>Speed [m/s]</th>
<th>Passive Wing Damping</th>
<th>Active Wing Damping</th>
</tr>
</thead>
<tbody>
<tr>
<td>10</td>
<td>0.0138</td>
<td>0.0208</td>
</tr>
<tr>
<td>20</td>
<td>0.0256</td>
<td>0.0327</td>
</tr>
<tr>
<td>25</td>
<td>0.0317</td>
<td>0.0394</td>
</tr>
<tr>
<td>30</td>
<td>0.026</td>
<td>0.0367</td>
</tr>
<tr>
<td>37.5</td>
<td>0.0214</td>
<td>0.0324</td>
</tr>
</tbody>
</table>
Looking to the results in figure 3.28, it can be seen that the wing damping for the seventh mode is always higher in the active mode than in passive one. Then, it can be concluded that the active wing has a higher flutter speed when compared with passive wing, as can be verified by the extrapolated curve to the zero damping. The coefficient of determination $R^2$, which represents the proportion of the variance of the damping with respect to the speed, is 0.9791 for the polynomial representing the passive wing, and 0.9925 for the active wing polynomial. Using these results, the flutter speed of the wing in the passive mode is $47.95\, \text{m/s}$ and in the active mode is $53.99\, \text{m/s}$, with variations defined by the previous referred $R^2$ values. This represents an increase in flutter speed of approximately 12.59%, meaning that with a given wing and a control system like this, one can fly 12.59% faster, without structural reinforcements of the structure and without suffering the aeroelastic effects of flutter. Finally, comparing
CHAPTER 3. WIND TUNNEL TESTS

the flutter speed of 44.87 m/s calculated in computational analysis, in Section 2.3, with the experimental value of 47.95 m/s, there is an 6.86% error comparing the experimental value with the computational one.

3.4 Adaptive Spar Wing Tests

3.4.1 Vibration Tests

As performed with the active skin wing, the RPV with the active spar wing was tested in the wind tunnel at several operating speeds, and with the controller in open and closed loop. The results of the displacement signals, with control ON (active wing) and control OFF (passive wing), were recorded for each flying condition. These results are shown in the table 3.6. Using the results shown in this table, the improvements of using the active wing were calculated. These improvements are shown in table 3.7. Like concluded with the active skin wing results, both the average displacement and the maximum displacement are generally lower with the active spar wing than with the passive one, making the active wing an improved wing version compared with the passive wing, in the majority of flight conditions. The results shown in table 3.6 can also be graphically seen in figures 3.29 and 3.30.

These tests were very helpful because the wind tunnel model was in a configuration in which the tail was very flexible, in order to force vibrations on the wing besides the aerodynamic induced vibrations. Then, the tests were repeated but without the flexible tail, in order to simulate the flying version of the RPV. Similar analysis was performed with this RPV configuration, and the obtained results are shown in tables 3.8, 3.9 and figures 3.31, 3.32. Note that the improvements are generally much better in this condition than in the configuration that includes the flexible tail. These
tests proved that the proposed approach for active aeroelastic control significantly decreases both the maximum and average displacements that a wing sustains during flight. This way, the fatigue life will increase leading to a better structure.

Table 3.6: Average and maximum displacement values for the passive and active spar wing configurations.

<table>
<thead>
<tr>
<th>Speed [m/s]</th>
<th>PASSIVE WING</th>
<th>ACTIVE WING</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>Average displacement</td>
<td>Maximum displacement</td>
</tr>
<tr>
<td>15</td>
<td>0.0179</td>
<td>0.0662</td>
</tr>
<tr>
<td>20</td>
<td>0.0282</td>
<td>0.1181</td>
</tr>
<tr>
<td>25</td>
<td>0.0902</td>
<td>0.2042</td>
</tr>
<tr>
<td>27.5</td>
<td>0.1319</td>
<td>0.2574</td>
</tr>
<tr>
<td>30</td>
<td>0.1412</td>
<td>0.2552</td>
</tr>
</tbody>
</table>

Table 3.7: Displacements improvements of the active spar wing compared with the passive spar wing.

<table>
<thead>
<tr>
<th>Speed [m/s]</th>
<th>Improvement of the average displacement [%]</th>
<th>Improvement of the maximum displacement [%]</th>
</tr>
</thead>
<tbody>
<tr>
<td>15</td>
<td>20.11</td>
<td>12.84</td>
</tr>
<tr>
<td>20</td>
<td>1.77</td>
<td>-1.27 (no improvement)</td>
</tr>
<tr>
<td>25</td>
<td>3.33</td>
<td>7.64</td>
</tr>
<tr>
<td>27.5</td>
<td>7.28</td>
<td>4.78</td>
</tr>
<tr>
<td>30</td>
<td>4.11</td>
<td>3.72</td>
</tr>
</tbody>
</table>
CHAPTER 3. WIND TUNNEL TESTS

Figure 3.29: Average displacements of both passive and active spar wing configurations, in the wind tunnel tests.

Table 3.8: Average and maximum displacement values for the passive and active spar wing configurations, using the RPV without the flexible tail.

<table>
<thead>
<tr>
<th>Speed [m/s]</th>
<th>PASSIVE WING</th>
<th>ACTIVE WING</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>Average displacement</td>
<td>Maximum displacement</td>
</tr>
<tr>
<td>15</td>
<td>0.0146</td>
<td>0.0379</td>
</tr>
<tr>
<td>20</td>
<td>0.0236</td>
<td>0.065</td>
</tr>
<tr>
<td>25</td>
<td>0.0477</td>
<td>0.103</td>
</tr>
<tr>
<td>27.5</td>
<td>0.0539</td>
<td>0.1086</td>
</tr>
<tr>
<td>30</td>
<td>0.0664</td>
<td>0.1339</td>
</tr>
</tbody>
</table>
CHAPTER 3. WIND TUNNEL TESTS

Figure 3.30: Maximum displacements of both passive and active spar wing configurations, in the wind tunnel tests.

Table 3.9: Displacements improvements of the active wing compared with the passive wing, using the RPV without the flexible tail.

<table>
<thead>
<tr>
<th>Speed [m/s]</th>
<th>Improvement of the average displacement [%]</th>
<th>Improvement of the maximum displacement [%]</th>
</tr>
</thead>
<tbody>
<tr>
<td>15</td>
<td>17.81</td>
<td>12.40</td>
</tr>
<tr>
<td>20</td>
<td>12.71</td>
<td>23.85</td>
</tr>
<tr>
<td>25</td>
<td>22.43</td>
<td>11.36</td>
</tr>
<tr>
<td>27.5</td>
<td>8.53</td>
<td>10.41</td>
</tr>
<tr>
<td>30</td>
<td>4.22</td>
<td>10.01</td>
</tr>
</tbody>
</table>
CHAPTER 3. WIND TUNNEL TESTS

Figure 3.31: Average displacements of both passive and active spar wing configurations, in the wind tunnel tests (RPV without tail).

Figure 3.32: Maximum displacements of both passive and active spar wing configurations, in the wind tunnel tests (RPV without tail).
3.4.2 Damping Analysis

Similarly as done with the active skin analysis, a study of damping at several key frequencies was performed. The damping influence was studied with the same two objectives: to quantify the amount of damping imposed by the controller in vibrations induced by the tail and to quantify the damping in the vibration modes associated with the wing flutter. All damping values were calculated using the FFT of the motion of the wing with and without control, throughout the speed range between 15 m/s and 30 m/s. After analyzing the FFT peak of the vibration imposed by the tail to the wing, the damping values were calculated and the results are shown in Table 3.10 and Figure 3.33. The tail vibration peak started at 5 Hz at 15 m/s and shifted to 8 Hz at 30 m/s.

Analyzing the figure 3.33, one can conclude that the damping values of the wing in the active mode are always higher than in the passive mode, even though the differences at speeds higher than 25 m/s are not very significant. These results demonstrate, again, that the controller besides improving the aeroelastic characteristics of the wing can also decrease forced vibration induced to the structure, and is more efficient in this task at low speeds, around the cruising speed.

<table>
<thead>
<tr>
<th>Speed [m/s]</th>
<th>Passive Wing Damping</th>
<th>Active Wing Damping</th>
</tr>
</thead>
<tbody>
<tr>
<td>15</td>
<td>0.888</td>
<td>1.25</td>
</tr>
<tr>
<td>20</td>
<td>0.646</td>
<td>0.928</td>
</tr>
<tr>
<td>25</td>
<td>0.571</td>
<td>0.5833</td>
</tr>
<tr>
<td>30</td>
<td>0.571</td>
<td>0.571</td>
</tr>
</tbody>
</table>

Table 3.10: Damping of the active spar wing due to tail vibration, tested at 15, 20, 25 and 30 m/s.
CHAPTER 3. WIND TUNNEL TESTS

3.4.3 Flutter Analysis

As stated in Section 3.3.3, the flutter tests were conducted with two objectives: to verify the predicted flutter speed of the passive wing working (calculated in Section 2.3.2), and to calculate the increase of the flutter speed related with the use of the controller (active wing). The flutter speeds (controller on an off) were determined experimentally by calculating the damping of the vibration modes that will induce flutter (first torsion mode at 46.7Hz). Using the FFT analysis on the controllers input signal amplitude of the wing, the damping values of the first torsion modes were obtained, as shown in table 3.11. Then, a plot of damping versus speed was designed and a second order polynomial was passed through the points in order to find the zero damping speed (corresponding to the flutter speed), as shown in figure 3.34. Note that the damping of the first torsion mode is the most important to
determine the flutter speed, since this wing has a conventional configuration (i.e., with a main beam), thus, it flutters conventionally.

The results in figure 3.34 show that the wing damping for the first torsion mode is always higher in the active mode than in passive one. Then, one can state that the active wing has a higher flutter speed when compared with passive wing, as can be verified by the extrapolated curve to the zero damping. The coefficient of determination $R^2$, which represents the proportion of the variance of the damping with respect to the speed, is 0.9986 for the polynomial representing the passive wing, and 0.9956 for the active wing polynomial. With these results, the flutter speed of the wing in the passive mode is $44.38 \text{ m/s}$ and in the active mode is $53.71 \text{ m/s}$, which represents an increase in flutter speed of 21.02%, with speed variations defined by the previous referred $R^2$ values. This means that with a given wing and a control system like this, you can fly approximately 21.02% faster without structural reinforcements of the structure and without suffering the aeroelastic effects of flutter. Also, in figure 3.34 it can be seen that the difference in damping values on active and passive modes increases with the test speed, suggesting that the controller is working to suppress flutter at high speeds (over $25 \text{ m/s}$). Comparing the passive wing flutter speed calculated in Section 2.3.2, of $49.83 \text{ m/s}$, with the experimental value of $44.38 \text{ m/s}$, there is an error of 12.28%.
Table 3.11: Damping of the wing for the first torsion mode, tested at 15, 20, 25, 27.5 and 30 m/s.

<table>
<thead>
<tr>
<th>Speed [m/s]</th>
<th>Passive Wing Damping</th>
<th>Active Wing Damping</th>
</tr>
</thead>
<tbody>
<tr>
<td>15</td>
<td>-</td>
<td>0.0889</td>
</tr>
<tr>
<td>20</td>
<td>0.0795</td>
<td>-</td>
</tr>
<tr>
<td>25</td>
<td>0.0698</td>
<td>0.0796</td>
</tr>
<tr>
<td>27.5</td>
<td>0.0666</td>
<td>0.0778</td>
</tr>
<tr>
<td>30</td>
<td>0.0568</td>
<td>0.0667</td>
</tr>
</tbody>
</table>

Figure 3.34: Curves of the wing damping for the first torsion mode, and extrapolation polynomial to zero damping.
Chapter 4

Flight Tests

4.1 Experimental Apparatus

The hardware involved in the flight tests is similar as the described in the Chapter 3. However there are some differences, as is next described. The flight tests were performed in the Sintra Air Base takeoff runway, Portugal. In terms of tests articles, the only difference is the RPV platform and the required engine. A new and improved RPV platform was used, which is described in Section 4.1.1. The flight tests were performed using the adaptive spar wing only, since is lighter than the adaptive skin wing. The same piezoelectric sensors and actuators were used. The used digital controller was also the same as described in Chapter 3, although the control law has some differences. In terms of electronic equipment, some additional components were used. Therefore, the next Sections include the description of the following hardware components: the RPV and the additional electronic equipment.
4.1.1 The RPV

The RPV fuselage is lighter than the one used for the wind tunnel tests, in order to have more payload capabilities. This fuselage is less stiff than the other. It was constructed mostly using balsa wood. Also, carbon fibre was used in the places where more stiffness was necessary. The used RPV engine has a power of 1.864 kW. The tables 4.1, 4.2, 4.3 and 4.4 present the RPV main characteristics in terms of geometry, aerodynamics and performance. The figure 4.1 shows the final RPV model ready to start a flight test. Note that the maximum speed at sea level is very high. However, and fortunately, it is smaller than the flutter speed of the passive wing. On the other hand, the stall speed in clean configuration is higher than 15 m/s, which was one wind tunnel testing speed. This speed was tested only to check the performance of the control system.

Table 4.1: RPV external dimensions.

<table>
<thead>
<tr>
<th>Parameter</th>
<th>in S.I. Units</th>
</tr>
</thead>
<tbody>
<tr>
<td>Main Wing Span</td>
<td>2.4 m</td>
</tr>
<tr>
<td>Main Wing Chord (Root)</td>
<td>0.33 m</td>
</tr>
<tr>
<td>Main Wing Chord (Tip)</td>
<td>0.33 m</td>
</tr>
<tr>
<td>Main Wing Aspect Ratio</td>
<td>7.273</td>
</tr>
<tr>
<td>Horizontal Stabilizer Span</td>
<td>0.72 m</td>
</tr>
<tr>
<td>Horizontal Stabilizer Chord (Root)</td>
<td>0.18 m</td>
</tr>
<tr>
<td>Horizontal Stabilizer Chord (Tip)</td>
<td>0.18 m</td>
</tr>
<tr>
<td>Horizontal Stabilizer Aspect Ratio</td>
<td>4</td>
</tr>
<tr>
<td>Vertical Stabilizer Span</td>
<td>0.2 m</td>
</tr>
<tr>
<td>Vertical Stabilizer Chord (Root)</td>
<td>0.184 m</td>
</tr>
<tr>
<td>Vertical Stabilizer Chord (Tip)</td>
<td>0.138 m</td>
</tr>
<tr>
<td>Vertical Stabilizer Aspect Ratio</td>
<td>1.242</td>
</tr>
<tr>
<td>Wheel Track</td>
<td>0.3713 m</td>
</tr>
<tr>
<td>Wheel Base</td>
<td>0.5048 m</td>
</tr>
<tr>
<td>Propeller Diameter</td>
<td>0.38 m</td>
</tr>
</tbody>
</table>
CHAPTER 4. FLIGHT TESTS

Table 4.2: RPV areas.

<table>
<thead>
<tr>
<th>Parameter</th>
<th>$m^2$</th>
</tr>
</thead>
<tbody>
<tr>
<td>Main Wing Area</td>
<td>0.792</td>
</tr>
<tr>
<td>Flaps Area</td>
<td>0.06336</td>
</tr>
<tr>
<td>Ailerons Area</td>
<td>0.06336</td>
</tr>
<tr>
<td>Horizontal Stabilizer Area</td>
<td>0.1296</td>
</tr>
<tr>
<td>Elevator Area</td>
<td>0.03786</td>
</tr>
<tr>
<td>Vertical Stabilizer Area</td>
<td>0.0644</td>
</tr>
<tr>
<td>Rudder Area</td>
<td>0.007491</td>
</tr>
</tbody>
</table>

Table 4.3: RPV weights and loadings.

<table>
<thead>
<tr>
<th>Parameter</th>
<th>in S.I. Units</th>
</tr>
</thead>
<tbody>
<tr>
<td>Maximum Takeoff Weight</td>
<td>10 Kg</td>
</tr>
<tr>
<td>Maximum Wing Loading</td>
<td>12.63 kg/m²</td>
</tr>
<tr>
<td>Maximum Power Loading</td>
<td>0.005365 kg/watts</td>
</tr>
</tbody>
</table>

Table 4.4: RPV performance data.

<table>
<thead>
<tr>
<th>Parameter</th>
<th>$m/s$</th>
</tr>
</thead>
<tbody>
<tr>
<td>Maximum Level Speed (See Level)</td>
<td>42.02</td>
</tr>
<tr>
<td>Cruise Speed (See Level, 75% Power)</td>
<td>38.6</td>
</tr>
<tr>
<td>Stall Speed Clean</td>
<td>15.81</td>
</tr>
<tr>
<td>Stall Speed (45° Flaps)</td>
<td>12.39</td>
</tr>
<tr>
<td>Maximum Rate of Climb</td>
<td>8.44</td>
</tr>
</tbody>
</table>
CHAPTER 4. FLIGHT TESTS

Figure 4.1: Picture of the RPV model ready to the flight tests.

4.1.2 Additional Electronic Equipment

In terms of electronic equipment, the same components as described in Section 3.1.4 were used, i.e., the controller, the amplifier, the signal conditioning circuit, the two power supply batteries and voltage regulators. However, during the flight tests it is important to have the real time information about the RPV airspeed. Thus, a telemetry system with bidirectional data link was installed in the RPV, and the speed information was available during the entire flight envelope. This system has an airborne station and a ground one. The components of the airborne station are the following (see figure 4.2):

- a microcomputer, which performs the signal conditioning, and radio frequency modulation operations; it is connected to the battery pack, pitot’s transducer, air station transceiver and onboard radio control receiver;

- a 7.2V, 1250mAh, 6 cell Ni-Cd battery pack, that supplies the power to the airborne station;

- a pitot sensor (total pressure and static pressure inputs), with a transducer to convert pressure in electronic signal, and a pulse position modulation modulator;
CHAPTER 4. FLIGHT TESTS

Figure 4.2: Photograph of the telemetry airborne station components.

- a transceiver with antenna, which sends radio signals to the ground station;

- a radio-control receiver with 9 channels, which has an independent battery pack.

The ground station components are (see figure 4.3):

- a transceiver that receives the radio signals from the aircraft; it is connected to the PC by the USB slot (power) and RS232 serial adapter (data);

- a radio-control unit, used by the pilot to fly the RPV; the ground system doesn’t work if this unit is turned off.

The software interface between the user and the telemetry system is the Jet-tronic II for Windows. This program has several applications, such as real-time changing parameters, and showing its real-time values in the data display. The figure 4.4
CHAPTER 4. FLIGHT TESTS

Figure 4.3: Photograph of the telemetry ground station components.

illustrates the data display window. It shows throttle and trim controls position, fuel pump voltage, engine RPM and EGT (Exhaust Gas Temperature), airspeed and battery voltage. The last two functions were the only ones used for the flight tests. The figure 4.5 shows the RPV fuselage containing all the electronic components necessary to perform the flight tests.

4.2 Tests Objectives and Procedures

The objective of the flight tests is to prove that the piezoelectrics vibration control technology works not only in a controlled environment (like the wind tunnel), but also in a real environment, and it is portable enough to fly in a small airplane like this one. During the flight, the control was automatically switched on and off every 12 sec, in order to compare the differences between active and passive wing configurations, and
CHAPTER 4. FLIGHT TESTS

Figure 4.4: Picture of the real-time data display.

Figure 4.5: Photograph of the RPV fuselage containing all the flight tests equipment.
the gain was set for a speed of 20m/s (approximately the cruise speed). The RPV was flown by an expert in flying RPVs, who tried to keep constant altitude and speed. The airspeed was monitored in the PC using the Jet-Tronic software, which received the pitot telemetry signal from onboard, and the testing time was being correlated with speed and both values registered.

Similarly to the performed in the wind tunnel tests, the control model used for the flight tests was designed in Simulink, and is shown in figure 4.6. This model is very similar to the wind tunnel control model. A digital clock block provides the simulation time. The fine gain value is replaced by the output of a multiport switch, which has as inputs "0" or the defined gain. The switch command signal is a pulse signal (0 or 1) added to 1, resulting on a square wave with the values 1 or 2. If this wave takes the value 1, the first input value (0) goes to the output. If it takes the value 2, the second input value (the defined gain) goes to the output. The period of the pulse wave is 24sec. With this system, there are 12sec with controller off and 12sec with controller on.

4.3 Adaptive Spar Wing Tests

As referred in the previous Section, the RPV with the adaptive spar wing was tested in flight tests, with the control law automatically switched on and off every 12sec, and the gain was set for the cruise speed of 20m/s. Although the pilot tried to keep the speed around 20m/s, not always was possible to have this exact flight speed, mainly during turns or when some gusts occurred. Like in wind tunnel tests, the results of the displacement signals (average and maximum values), with control on and off were recorded. These data was analyzed and was only validated for the periods in which the airplane flew at 20m/s, which was the target speed for the flight. In this flight,
CHAPTER 4. FLIGHT TESTS

Figure 4.6: Diagram representing the control model implemented for flight tests.
the RPV flew at 20m/s in 4 speed windows (1, 2, 9 and 16) and the results in terms of maximum and average amplitudes for these four windows are shown in Table 4.5. Using the results shown in the table 4.5, the improvements of using the active wing were calculated. These improvements are shown in Table 4.6. Although these are the results from a flight test, in which the weather conditions can not be controlled, the results are stable and positive, since both the average and the maximum displacements are always lower with the active spar wing than with the passive spar wing. Additionally, note that the improvements in maximum displacements are larger than the improvements in average displacements, maybe because the peaks induced by turns are much larger in passive mode than in active mode. The best average displacement improvement occurred at window 9 (i.e., 7%) and the best maximum displacement improvement occurred at window 2 (i.e., 43.84%).

As an illustration, a graph of the wing response in flight with the controller on and off is shown in figure 4.7. In this figure, the signal amplitude versus time is shown, and the controller is off in the first half of the graph (i.e., on the left side of the green line) and is working on the second half (i.e., on the right side of the green line). Looking to this graph, it becomes clear that both the maximum displacements (peaks) are lower when the control law is turned on.

Table 4.5: Average and maximum displacement values for the passive and active spar wing configurations, in the flight tests.

<table>
<thead>
<tr>
<th>Speed Window</th>
<th>PASSIVE WING</th>
<th>ACTIVE WING</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>Average displacement</td>
<td>Maximum displacement</td>
</tr>
<tr>
<td>1</td>
<td>0.0645</td>
<td>0.2684</td>
</tr>
<tr>
<td>2</td>
<td>0.0632</td>
<td>0.2867</td>
</tr>
<tr>
<td>9</td>
<td>0.0543</td>
<td>0.2262</td>
</tr>
<tr>
<td>16</td>
<td>0.0568</td>
<td>0.2545</td>
</tr>
</tbody>
</table>
Table 4.6: Displacements improvements of the active spar wing compared with the passive spar wing, in the flight tests.

<table>
<thead>
<tr>
<th>Speed Window</th>
<th>Improvement of the average displacement [%]</th>
<th>Improvement of the maximum displacement [%]</th>
</tr>
</thead>
<tbody>
<tr>
<td>1</td>
<td>6.5</td>
<td>40.0</td>
</tr>
<tr>
<td>2</td>
<td>4.6</td>
<td>43.8</td>
</tr>
<tr>
<td>9</td>
<td>7.0</td>
<td>39.0</td>
</tr>
<tr>
<td>16</td>
<td>3.9</td>
<td>18.2</td>
</tr>
</tbody>
</table>

Figure 4.7: Amplitude ($V/5$) of the wing versus time, in speed window 7.
Chapter 5

Conclusions and Future Work

This research has resulted in an experimental demonstration of aircraft vibration and flutter suppression using piezoelectric materials. It was shown that for this kind of aircrafts, piezoelectric actuators can be used in order to decrease the wing vibrations. Therefore, it is possible to improve the aeroelastic characteristics of the wing, like increasing the wing flutter speed. A wind tunnel and a flight RPV models were designed, fabricated, installed, and tested. Structural and aerodynamic wing models were created, using ANSYS and ZAERO programs. ANSYS program was used to determine the wing natural frequencies and modal shapes. ZAERO program calculates the wing flutter speed. A digital control law was designed and implemented using MATLAB program. Open and closed loop vibration and flutter tests were conducted in the wind tunnel and in flight, with excellent correlation achieved by computational predictions.

The configuration and dimensions of the RPV models were designed such that it would flutter within the operating envelope of the tunnel, could be safely tested within the available test section, and would have surfaces for mounting the piezoelectric plates. Two wing concepts were analyzed in this thesis: a wing with piezoelectric
materials mounted in the wing surface (active skin concept), and a wing with piezoelectric materials mounted in the main spar (active spar concept). Both wings were constructed using carbon fibre, balsa wood and, in the active skin wing, also using fibre glass. The piezoelectric sensor was mounted near the wing root in both wings. The piezoelectric actuators were mounted in the internal part of the carbon fibre plates. In the active skin wing, they have been glued along the span of the upper and lower skins; in the active spar wing, along the span of upper and lower parts of the main spar. Thus, because of this actuators placement, the implemented control had the objective of controlling bending vibrations.

Since the fabricated wings had complex structure configurations, and an infinite number of degrees of freedom, it was previously decided that a commercial code would be used in order to calculate the solution of the problem. In this thesis, the wings structures were studied using the ANSYS program. The analysis of free vibration was performed in order to generate the passive wings natural frequencies and mode shapes. After obtaining these results, the ZAERO program was used to perform the wings aeroelastic study in terms of flutter. ZAERO program imports the solution of the free vibration generated by the ANSYS solution.

For the adaptive skin wing concept, the first ten natural frequencies of its passive configuration were predicted to be: 16.4Hz, 30.9Hz, 38.5Hz, 53.2Hz, 56.7Hz, 63.7Hz, 76.3Hz, 88.3Hz, 90.4Hz and 93.0Hz. In terms of its mode shapes, some of them lead with shell local vibrations, which are not beneficial for this type of research, as explained in Section 2.2.1. The mode shapes that lead with shell vibrations are: the 2nd, 4th, and 8th modes, with shell bending, and the 7th mode, with shell bending-torsion. This way, the 2nd, 4th, 7th and 8th modes are mainly characterized by the two carbon plates vibration in which the lower and upper carbon plates have deformations in opposite phases. On the other hand, the remaining modes have conventional be-
behavior, as follows: 1\textsuperscript{st} mode is the first bending, 3\textsuperscript{rd} mode is the first bending-torsion mode, 5\textsuperscript{th} mode is the first torsion mode, 6\textsuperscript{th} mode is the second bending-torsion mode, 9\textsuperscript{th} mode is the second torsion mode, and 10\textsuperscript{th} mode is the third bending-torsion mode. Using the ANSYS results, the ZAERO aeroelastic analysis of the open loop system gave a flutter prediction of 44.87\textit{m/s}, associated with the wing 7\textsuperscript{th} mode.

The adaptive spar wing the first five natural frequencies of its passive configuration were predicted to be: 19.8\textit{Hz}, 32.6\textit{Hz}, 46.7\textit{Hz}, 84.1\textit{Hz} and 124.3\textit{Hz}. Since this wing has a conventional configuration, conventional flutter was expected. Analyzing the results, no shell local vibrations were found, and the following natural modes were determined: 1\textsuperscript{st} mode is the first bending, 2\textsuperscript{nd} mode is a "swing" mode (bending in the wing plane), 3\textsuperscript{rd} mode is the first torsion, 4\textsuperscript{th} mode is a bending-torsion mode, and 5\textsuperscript{th} mode is the second torsion mode. The estimated flutter speed for the open loop system was 49.83\textit{m/s}, associated with the wing first torsion mode.

In the wind tunnel tests, experimental vibration analysis was performed. The RPV with the adaptive wing was tested in the wind tunnel at several operating speeds, and with the controller in open and closed loop. The signals of wing displacement were recorded for each flying condition. After this, the results of these displacement signals (average and maximum values) with control ON (active wing) and control OFF (passive wing) were then studied. Also, the structural dampings associated with several key frequencies were determined. The damping influence was studied with two objectives: to quantify the amount of damping imposed by the controller in vibrations induced by the tail and to quantify the damping in the vibration modes associated with the wing flutter.

For the adaptive skin wing, it was concluded that both the average displacement and maximum displacement are generally lower with the active wing than with the passive one, making the active wing an improved wing version compared with the
passive wing, in the majority of flight conditions. The best improvement, in terms of average displacement, is 22.1% at 10m/s, and in terms of maximum displacement, is 13.6% at 35m/s. In terms of structural damping associated with the tail vibration, the damping values of the wing in the active mode are higher than in the passive mode in almost all the flying conditions. These results demonstrate that the controller is improving the aeroelastic characteristics of the wing and decreasing forced vibrations induced to the structure by the tail. For the flutter analysis, the damping associated with 7th wing mode was studied. It was concluded that the wing damping for the 7th mode is always higher in the active mode than in passive one. Then, it can be concluded that the active wing has a higher flutter speed when compared with passive wing. Using these results, the experimental flutter speed of the wing in the passive mode is 47.95m/s and in the active mode is 53.99m/s. This represents an increase in flutter speed of 12.59%, meaning that with a given wing and a control system like this, one can fly 12.59% faster without structural reinforcements of the structure and without suffering the aeroelastic effects of flutter. Finally, comparing the flutter speed of 44.87m/s calculated in computational analysis, with the experimental value of 47.95m/s, there is an 6.86% error comparing the experimental value with the computational one.

Similar conclusions were obtained with the active spar wing. In terms of vibration analysis, both the average displacement and the maximum displacement are generally lower with the active spar wing than with the passive one, making the active wing an improved wing version compared with the passive wing, in the majority of flight conditions. The best improvement, in terms of average displacement, is 20.1% at 15m/s, and in terms of maximum displacement, is 12.8% also at 15m/s. The vibration tests were repeated but without the flexible tail, in order to simulate the flying version of the RPV, and the improvements in terms of vibration were even better:
22.4% at 25m/s for the average displacement, and 23.9% at 20m/s for the maximum displacement. In terms of structural damping associated with the tail vibration, the damping values of the wing in the active mode are always higher than in the passive mode in all the flying conditions. These results demonstrate, again, that the controller is improving the aeroelastic characteristics of the wing and decreasing forced vibrations induced to the structure by the tail. For the flutter analysis, the damping associated with 1st torsion mode was studied. It was concluded that the wing damping for this mode is always higher in the active mode than in passive one. Then, one can state that the active wing has a higher flutter speed when compared with passive wing. With these results, the experimental flutter speed of the wing in the passive mode is 44.38m/s and in the active mode is 53.71m/s, which represents an increase in flutter speed of 21.02%. This means that with a given wing and a control system like this, one can fly 21.02% faster without structural reinforcements of the structure and without suffering the aeroelastic effects of flutter. Comparing the flutter speed of 49.83 m/s calculated in Section 2.3.2, with the experimental value of 44.38m/s, there is an error of 12.28%.

Flight tests were also performed in order to prove that the piezoelectrics vibration control technology works not only in a controlled environment (like the wind tunnel), but can also work in a real environment, and it is portable enough to fly in a small airplane like this one. During the flight, the control was automatically switched on and off every 12sec, in order to compare the differences between active and passive wing configurations, and the gain was set for a speed of 20m/s (approximately the cruise speed). The RPV was flown by an expert in flying RPVs, who tried to keep constant altitude and speed. The airspeed was monitored in the PC using the Jet-Tronic software, which received the pitot telemetry signal from onboard, and the testing time was being correlated with speed and both values registered.
In the flight tests, only the RPV with the active spar wing was tested. Although the pilot tried to keep the speed around 20\textit{m/s}, not always was possible to have this exact flight speed, mainly during turns or when some gusts occurred. Like in wind tunnel tests, the results of the displacement signals (average and maximum values), in open and closed loops were recorded. These data was analyzed and was only validated for the periods in which the airplane flew at 20\textit{m/s}. Although these are the results from a flight test, in which the weather conditions can not be controlled, the results are positive, since both the average and the maximum displacements are always lower with the active spar wing than with the passive spar wing. The best average displacement improvement was 7\%, and the best maximum displacement improvement 43.84\%.

It is recommended that further research be performed in the area of controlling the aeroelastic responses of a vehicle by utilizing piezoelectric actuators. Since this is a realistic study, which incorporates piezoelectric actuators in terms of a real aircraft, more complex work can be performed in the future using this flying platform. One of the studies that can be done in the future is the use of piezoelectric actuators in order to control also wing torsional vibrations. This can be done mounting the piezoelectric actuators along the wing chord, instead of along the wing span as done in the present research. Additionally, a wing with both active skin and active spar control can be implemented. Another interesting study is the use of a different adaptive material, for instance shape memory alloys, to perform the same tasks as done is this thesis.

The development of the adaptive materials application in real aircrafts highly depends on the research in the functional materials area. One can not forget that many of these materials have some limitations. For instance, the sensors used in this study are ceramics, which are very fragile. Also, current adaptive actuators have some limitations in terms of force, displacement and operating frequency.
Flutter suppression is not the only application that may call for actuation made of adaptive materials. Using adaptive materials actuation, local strains could be produced to counter the loads induced within the structure during manoeuvring. This has the potential to extend the service life of an aircraft, which traditionally undergo high g-loading, and to expand its operational limits. The concept of an adaptive wing is worthy of investigation. In the future, it would probably eliminate the hydraulic problems encountered on previous attempts, in order to create a wing which shape can be optimized for several fight conditions.
References


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