Design and Testing of a Flight Control System for Unstable Subscale Aircraft

ALEJANDRO SOBRÓN RUEDA
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Master’s Thesis in Aeronautical Engineering

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Abstract

The primary objective of this thesis was to study, implement, and test low-cost electronic flight control systems (FCS) in remotely piloted subscale research aircraft with relaxed static longitudinal stability. Even though this implementation was carried out in small, simplified test-bed aircraft, it was designed with the aim of being installed later in more complex demonstrator aircraft such as the Generic Future Fighter concept demonstrator project. The recent boom of the unmanned aircraft market has led to the appearance of numerous electronic FCS designed for small-scale vehicles and even hobbyist-type model aircraft. Therefore, the purpose was not to develop a new FCS from scratch, but rather to take advantage of the available technology and to examine the performance of different commercial off-the-shelf (COTS) low-cost systems in statically unstable aircraft models. Two different systems were integrated, calibrated and tested: a simple, gyroscope-based, single-axis controller, and an advanced flight controller with a complete suite of sensors, including a specifically manufactured angle-of-attack transducer. A flight testing methodology and appropriate flight-test data analysis tools were also developed. The satisfactory results are discussed for different flight control laws, and the controller tuning procedure is described. On the other hand, the different test-bed aircraft were analysed from theoretical point of view by using common aircraft-design methods and conventional preliminary-design tools. The theoretical models were integrated into a flight dynamics simulator, which was compared with flight-test data obtaining a reasonable qualitative correlation. Possible FCS modifications are discussed and some future implementations are proposed, such as the integration of the angle-of-attack in the control laws.

Keywords: aircraft design, systems integration, subscale flight testing, avionics, flight control system, remotely piloted aircraft.
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This work is dedicated to every one who has inspired and encouraged me to keep always learning over the years. This includes remarkable professors, talented colleagues, and above all, those who taught me the most important lessons in life: my parents Emma and José María.

Alejandro Sobrón Rueda, Linköping, June 2015
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<th>Description</th>
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<tbody>
<tr>
<td>ADC</td>
<td>Analog-to-Digital Converter</td>
</tr>
<tr>
<td>AOA</td>
<td>Angle-Of-Attack</td>
</tr>
<tr>
<td>AVCS</td>
<td>Angular Vector Control System</td>
</tr>
<tr>
<td>BVLOS</td>
<td>Beyond Visual Line-Of-Sight</td>
</tr>
<tr>
<td>CAD</td>
<td>Computer-Aided Design</td>
</tr>
<tr>
<td>CAS</td>
<td>Control Augmentation System</td>
</tr>
<tr>
<td>CFD</td>
<td>Computational Fluid Dynamics</td>
</tr>
<tr>
<td>CG</td>
<td>Centre of Gravity</td>
</tr>
<tr>
<td>CHR</td>
<td>Cooper-Harper rating scale</td>
</tr>
<tr>
<td>COTS</td>
<td>Commercial Off-The-Shelf</td>
</tr>
<tr>
<td>EKF</td>
<td>Extended Kalman Filter</td>
</tr>
<tr>
<td>FBW</td>
<td>Fly-By-Wire</td>
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<tr>
<td>FCL</td>
<td>Flight Control Law</td>
</tr>
<tr>
<td>FCS</td>
<td>Flight Control System</td>
</tr>
<tr>
<td>GFF</td>
<td>Generic Future Fighter</td>
</tr>
<tr>
<td>HIL</td>
<td>Hardware-In-the-Loop</td>
</tr>
<tr>
<td>I/O</td>
<td>Input/Output</td>
</tr>
<tr>
<td>IMU</td>
<td>Inertial Measurement Unit</td>
</tr>
<tr>
<td>MAC</td>
<td>Mean Aerodynamic Chord</td>
</tr>
<tr>
<td>MEMS</td>
<td>Micro-Electro-Mechanical System</td>
</tr>
<tr>
<td>NASA</td>
<td>National Aeronautics and Space Administration (United States)</td>
</tr>
<tr>
<td>NP</td>
<td>Neutral Point, control fixed</td>
</tr>
<tr>
<td>PCB</td>
<td>Printed Circuit Board</td>
</tr>
<tr>
<td>PIO</td>
<td>Pilot-Induced Oscillation</td>
</tr>
<tr>
<td>PSD</td>
<td>Power Spectral Density</td>
</tr>
<tr>
<td>PWM</td>
<td>Pulse Width Modulation</td>
</tr>
<tr>
<td>R/C</td>
<td>Radio Control</td>
</tr>
<tr>
<td>RPA</td>
<td>Remotely Piloted Aircraft</td>
</tr>
<tr>
<td>RPAS</td>
<td>Remotely Piloted Aircraft System</td>
</tr>
<tr>
<td>RTOS</td>
<td>Real-Time Operating System</td>
</tr>
<tr>
<td>SAS</td>
<td>Stability Augmentation System</td>
</tr>
<tr>
<td>TAS</td>
<td>True Airspeed</td>
</tr>
<tr>
<td>UAS</td>
<td>Unmanned Aircraft System</td>
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<td>UAV</td>
<td>Unmanned Aerial Vehicle</td>
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<tr>
<td>VLM</td>
<td>Vortex Lattice Method</td>
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<td>VLOS</td>
<td>Visual Line-Of-Sight</td>
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### Symbols

<table>
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<th>Symbol</th>
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<tr>
<td>$\alpha$</td>
<td>Angle-of-attack [rad]</td>
</tr>
<tr>
<td>$\bar{C}_f$</td>
<td>Equivalent flat-plate skin-friction coefficient</td>
</tr>
<tr>
<td>$\delta_e$</td>
<td>Elevon deflection [rad]</td>
</tr>
<tr>
<td>$\eta$</td>
<td>Efficiency factor</td>
</tr>
<tr>
<td>$\rho$</td>
<td>Density [kg m$^{-3}$]</td>
</tr>
<tr>
<td>$\theta$</td>
<td>Pitch angle [rad]</td>
</tr>
<tr>
<td>$\upsilon$</td>
<td>Electric potential tension or voltage [V]</td>
</tr>
<tr>
<td>$c$</td>
<td>Chord length [m]</td>
</tr>
<tr>
<td>$C_x$</td>
<td>Coefficient of the magnitude $x$</td>
</tr>
<tr>
<td>$D$</td>
<td>Drag force [N]</td>
</tr>
<tr>
<td>$e$</td>
<td>Electric signal</td>
</tr>
<tr>
<td>$g$</td>
<td>Standard acceleration due to gravity: 9.81 [m s$^{-2}$]</td>
</tr>
<tr>
<td>$I_{xx}$</td>
<td>Moment of inertia about the $x$-axis [kg m$^2$]</td>
</tr>
<tr>
<td>$I_{xy}$</td>
<td>Product of inertia in the $xy$-plane [kg m]</td>
</tr>
<tr>
<td>$L$</td>
<td>Lift force [N]</td>
</tr>
<tr>
<td>$n$</td>
<td>Normal acceleration or load factor</td>
</tr>
<tr>
<td>$P$</td>
<td>Power [W]</td>
</tr>
<tr>
<td>$Q$</td>
<td>Flight dynamic pressure [Pa]</td>
</tr>
<tr>
<td>$q$</td>
<td>Pitch angular rate [rad s$^{-1}$]</td>
</tr>
<tr>
<td>$Re$</td>
<td>Reynolds number</td>
</tr>
<tr>
<td>$S$</td>
<td>Area [m$^2$]</td>
</tr>
<tr>
<td>$T$</td>
<td>Thrust force [N]</td>
</tr>
<tr>
<td>$W$</td>
<td>Weight force [N]</td>
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1

Introduction

In early design stages of aircraft and spacecraft, designers have constantly taken advantage of the development of advanced analysis tools and state-of-the-art verification techniques. The ability to analyse and predict crucial characteristics of the vehicle plays a critical role in the development process, not only by accelerating and optimizing it but also by reducing risk and redesign costs. A good example of this can be found in the massive development of powerful computational tools over the last decades which has brought unprecedented analysis capabilities to engineers, especially in fluid dynamics and structural design. However, as NASA's researcher J. Chambers points out, one of the most effective and valuable tools used in aerospace since its birth has been testing of subscale models \[1\]. According to this author, subscale models can be defined in this context as physical, downsized reproductions of components or vehicles used to examine characteristics of larger full-scale counterparts. As a result of technological progress, model aircraft testing has been firmly established in aerospace research and it plays an important role in the aircraft development process, from static wind tunnel investigations to dynamic free-flight models and advanced concept demonstrators. Especially in the recent years, miniaturization of electronics, sensors, and data acquisition systems as well as their cost decrease, have opened up tremendous possibilities in subscale flight testing. It has also become affordable for small companies and academic institutions, where it additionally serves as an extraordinary practical training for aerospace engineering students as reported by Jouannet, Berry and Krus \[2\]. This work investigates the implementation of low-cost control systems originally developed in the fields of robotics and unmanned aerial vehicles to research-oriented flight testing of aircraft models, focusing on configurations that exceed the manual control capabilities of the pilot: flying statically unstable subscale aircraft.

1.1 Objectives and Limitations

The primary objective of this thesis was to study, implement, and test low-cost electronic flight control systems in remotely piloted subscale research aircraft with relaxed static longitudinal stability. Even though this implementation was carried out in small test-bed aircraft, it was designed with the aim of being installed later in more complex demonstrator aircraft such as the Generic Future Fighter concept demonstrator which will be mentioned later. However, this somewhat broad objective introduced several indirect research questions:
1 Introduction

• What is the maximum level of relaxed stability that can be handled manually by a remote pilot?

• Are low-cost Commercial off-the-shelf (COTS) inertia-based controllers precise enough and are they able to handle the high frequency dynamics of a small and light unstable aircraft model? Can this be a problem for conventional hobbyist-type components?

• Is it possible to build a flight dynamics model of the test-bed aircraft and estimate the response of the system? How will it correlate with the real system?

• How can a COTS controller be modified in order to integrate external sensors (e.g. airspeed, angle-of-attack) in the control laws? Can the control system be programmed to handle non-conventional aircraft configurations such as delta-canard?

• Can the implemented system be evaluated by flight-testing simplified aircraft platforms? Are data suitable for scientific analysis?

These issues involve different technical fields and some could be subjects of thorough dedicated investigation. Therefore, due to time and scope restrictions it was necessary to define some limitations: simplified models and low-mid fidelity tools were used for the theoretical study of the aircraft and the control system. The purpose was to demonstrate its functionality rather than to obtain high-accuracy results. The theoretical models are however open to further development and more sophisticated analysis tools can be implemented in future. Additionally, the theoretical analysis was limited to longitudinal dynamics.

Regarding the practical tasks, modifications of the COTS controller were only those necessary to satisfy the needs of the aircraft configuration and to integrate the external sensors. Thus, no further investigation of control algorithms was carried out, and no attention was given to autonomous navigation capabilities. The aim was again to demonstrate the system functionality and to leave it ready for future integration of more complex control algorithms. Finally, flight testing was performed with small, simplified, remotely piloted models which were non dynamically- or geometrically-scaled from their respective full scale counterparts.

1.2 Methodology

In a nutshell, the methodology followed in this study can be outlined in three main branches, as illustrated in Figure 1.1. The first group comprises analytical tasks such as the development of the theoretical models and the development of the data analysis tools needed for both simulation and flight testing. Apart from the selection of the appropriate COTS components, the second group includes more practical tasks such as the integration, the set up, and the calibration of the different instruments and systems. The third group covers purely practical tasks such as the construction of the test-bed aircraft and the flight testing. Of course, all these branches converge in the end, where the models and control laws are experimentally evaluated by flight testing.
1.3 Thesis Outline

Apart from the general introduction presented in this chapter, the document consists of four main parts followed by final analysis and conclusion. Each one of these four parts includes a brief presentation of the discussed topic, an exposition of the work carried out, and its outcome. In the first part, a short review of the related fields is presented. The second part comprises the development of the theoretical models of the flying platforms and the analytical evaluation of different control systems. The third part contains a more detailed insight into the real flight control system implementation, the development of additional instruments, and the integration process in terms of both software and hardware. The fourth part presents a experimental evaluation of different systems carried out by flight testing. Finally, overall results analysis, recommendations, and some general conclusions are included in the last chapter.
2

Background

The purpose of this chapter is to offer the reader a brief review of the disciplines involved in the study. The exposition starts with a discussion about the role of subscale flight testing in aircraft design with special emphasis in its applications to flight control and some of its main limitations. A short comment on Linköping University's activities in this field is also included. Furthermore, it was considered appropriate to offer here an insight into contemporary unmanned aircraft systems technology and to describe how it is used in modern remotely piloted research aircraft. This includes some basic notions about the type of remote operation that is used in this investigation. Finally, some general knowledge of contemporary flight control strategies used in advanced aircraft is additionally included, with special attention to the control approaches relevant to this study.

2.1 Subscale Flight Testing in Aircraft Design

Testing physical prototypes constitutes a valuable tool in aircraft design since it can complement the sparse knowledge available in early design stages with critical data that could be very difficult or costly to obtain by other methods, as suggested by Chambers, Jouannet, Lundström, and others [1, 3, 4]. According to the same authors, the decreasing cost and turnaround time allow flight testing to be used nowadays as an excellent complement to the escalating cost and complexity of high-fidelity simulations. Maximising knowledge during early design stages is of critical importance since according to Roskam more than 80 percent of the life-cycle cost of an aircraft is already incurred before finishing the preliminary design [5], and these are precisely the stages at which subscale flight testing can provide with valuable data as symbolised in Figure 2.1. Additionally, other authors such as Walker [6] admit that there is a strong correlation between aircraft size and cost, agreeing that the use of subscale demonstrators represents in most cases a satisfactory trade-off between risk, cost, and fidelity of results, especially for evaluation of radical or unconventional designs for which no previous experience exists. Chambers states that free-flight subscale aircraft have been constantly used for more than 50 years by NASA and its predecessor NACA in order to gather important data and provide with confidence and risk reduction for new designs. Furthermore, he affirms that subscale flight testing has proved itself extremely valuable in research of critical stability and control characteristics for complex flight conditions that are not easily studied with conventional techniques, such as the investigation of dangerous manoeuvres outside the normal flight envelope like flight at high angles-of-attack, stall and spin [1].
Scaled models are designed according to certain scaling laws that ensure that the measured data are similar (i.e., correlated by known factors) to those of the full-scale counterpart, see Wolowicz [9]. At small scales it is usually impossible to satisfy all scaling parameters, and it is up to the designer to select the scaling method that best suits the investigation and to specify which characteristics of the design must be carefully reproduced and which ones can be simplified. Aerodynamic similarity is usually imperative for static wind tunnel models, while it is compromised in favour of inertial similarity for free-flight models in order to ensure that its motion is dynamically similar to that of the full-scale aircraft. The latter, known as dynamic scaling, is the most common scaling method in subscale flight testing according to Lundström [4].

The complexity of research flying models has evolved alongside available technology and it varies according to the nature of the investigation, from modified hobbyist-type gear to advance aerospace-grade systems and avionics. As reported by Chambers, relatively-simple unpowered drop models, both remotely controlled and uncontrolled, were used during the development of several aircraft programs such as the McDonnell Douglas F-4 and F-15, the Rockwell-MBB X-31, the Lockheed Martin F-22 [1], and also in scales up to 22 percent as for the McDonnell Douglas F/A-18E/F [10]. More advanced subscale models are frequently powered by their own internal propulsion systems. Electric propulsion and compressed air ejectors have often been used in dynamic free-flight model tests inside NASA Langley’s large Subsonic Wind Tunnel [1], while conventional gas turbine engines are usually fitted to larger remotely piloted vehicles such as NASA’s Highly Manoeuvrable Aircraft Technology (HiMAT) demonstrator [11], the 28 percent dynamically-scaled Boeing-NASA X-36 agility research aircraft [6], the 8.5 percent dynamically-scaled Boeing-NASA X-48 blended wing body test air-

Figure 2.1: Potential effect of subscale flight testing on contemporary aircraft development process. Elaborated with data from [5, 7, 8].
2.1 Subscale Flight Testing in Aircraft Design

craft [12], and academic projects such as the German-Polish Flexi-Bird IEP research aircraft platform [13]. However, subscale aircraft are not always remotely piloted: an excellent example can be found during the development of the Saab J35 Draken double-delta aircraft, in which a sub-scaled manned demonstrator was built and successfully tested, namely the Saab 210 “Lilldraken” [14].

Although traditionally flying aircraft models carried less extensive instrumentation than that used for the static wind tunnel models [1], this trend has changed in recent years due to the miniaturization of electronics and sensors, as shown by NASA’s Free-flying Aircraft for Subscale Experimental Research (FASER) project [15]. A good example of the current state of the art in subscale flight testing is the NASA’s Airborne Subscale Transport Aircraft Research (AirSTAR) project, in which turbine-powered, 5.5 percent dynamically-scaled remotely piloted vehicles with a tremendous suite of instrumentation and telemetry links are used to validate modelling methods, flight dynamics characteristics, and control system designs for large transport aircraft in high-risk flight conditions [16, 17, 18]. Considering the complex avionics, the system architecture, and the operation techniques of the aerial platforms, this project is also a perfect example of how subscale testing can nowadays take advantage of the emerging unmanned aircraft technology to improve flight testing operations and research scope. This particular topic will be discussed in more detail later in section 2.2.

2.1.1 Applications to Flight Control

According to Rizzi, modern aircraft design trends towards augmented stability and expanded flight envelopes demand an increased knowledge about stability and control as early as possible in the development process in order to design the correct flight control system (FCS) architecture [8]. Since prediction errors related to stability and control, according to the same author, are responsible for costly and risky fly-and-try fixes during the flight test program, it is crucial to start the FCS design with an acceptable knowledge already in the conceptual design phase. Although simulations have traditionally been favoured [4], subscale flight testing has also been used to investigate and develop advanced FCS’s during the last decades. Chambers gives again a good description of NASA’s activities in this field and the following information has been mostly extracted from his work [1]. Free-flying dynamically-scaled models have been extensively used since the 1980s by NASA and its partners to research on “supermanoeuvrability” capabilities at high angles-of-attack. Subscale testing has been extremely useful not only to investigate the non-linear stability characteristics of certain aircraft, but also for development of advanced FCS’s. Special attention was given to the integration of pitch- and yaw-vectoring vanes in combination with additional lifting surfaces such as canards and deflectable nose- or tail-strakes. Free-flying models equipped with these capabilities obtained valuable data during the development of new experimental designs such as the Grumman X-29 and the Rockwell-MBB X-31; and they were also utilized to evaluate the benefits of this technology when retrofitted to existing aircraft configurations such as the F/A-18 prior to its full-scale tests [19]. Furthermore, Chambers mentions that preliminary free-flight evaluation of the dynamic stability and control of the YF-22 fighter prototype at high angles-of-attack was
carried out with a powered subscale model, which included thrust-vectoring nozzles and an noseboom equipped with flow angle vanes and sensors for implementation of critical FCS elements of the full-scale aircraft. This ability to implement these full-scale FCS elements into subscale demonstrators has proved to be particularly interesting for practical study and verification of control laws in unconventional configurations, as shown first by NASA's Highly Manoeuvrable Aircraft Technology (HiMAT) programme [11], and more recently in tailless concept demonstrators such as the Boeing-NASA X-36 agility research aircraft programme [6], and the Boeing-NASA X-48 blended wing body test aircraft [12]. The previously mentioned AirSTAR project by NASA is also a perfect example of the application of subscale flight testing for research and development of adaptive guidance and control laws, which in this case focuses on large transport aircraft [16, 17, 18].

2.1.2 Some Limitations

Although at first sight the possibilities of subscale flight testing may seem extremely attractive, it is important to notice that both testing and results analysis are tightly constrained by certain factors. Wolowicz offers an extensive description of different scaling methods and their respective problems in [9]. Regarding the prevalent method of dynamic scaling, aero-elastic effects are usually neglected due to their complexity, and key scaling factors include geometric similarity, aerodynamic similarity (Reynolds number, compressibility), inertia scaling, and Froude number; see Chambers [1] for the detailed mathematical expressions. As this author comments, it is necessary to carry out numerous preliminary studies to assess if the weight/payload/strength compatibility issues can be met, and to determine a feasible model scale for the desired simulated flight conditions. According to various authors, scaling laws make dynamically-scaled free-flight models weight significantly more than conventional hobbyist-type, radio-controlled models for the same geometric scale, which therefore increases the flight speed. This problem becomes even worse when the subscale model flying at sea level must simulate a full-scale aircraft flying at high altitude [1, 4, 18]. Scaling factors also introduce other challenges, such as complicated structural design, problems with onboard space for instrumentation and other devices, and selection of control actuators in the case of small models with rapid angular motions. As Chambers points out, since it is not possible to satisfy all similitude requirements, it is critical to be aware of the limitations of the subscale test. Therefore, the results must be interpreted carefully and keeping in mind that they should not be extended beyond their intended areas of application.

The principal problem in dynamic subscale testing is the aerodynamic similarity between the model and the full-scale aircraft. As stated by Chambers and Grafton [20], even though compressibility effects can be included in the subscale analysis, there is always a significant discrepancy in the Reynolds number values which can play an important role when analysing viscosity-dependent phenomena such as flow separation at high angles-of-attack, departure modes, and maximum lift conditions. Bertin and Cummings give a detailed review of these aerodynamic phenomena in [21] and will...
not be included here. A good illustration of this issue is given in Figure 2.2, adapted from a report by Vicroy [22] and showing the multiple tests carried out at different scales in order to investigate thoroughly the characteristics of the Boeing-NASA X-48B Blended Wing Body demonstrator. However, when adverse scale effects are not significant, Chambers mentions that the aerodynamic characteristics of NASA's subscale models have been found to agree very well with data from other wind tunnel tests and theoretical analyses [1]. After all, NASA's ability to conduct free-flights and precise wind tunnel measurements with the same aircraft model is a key advantage for interpreting the results.

Figure 2.2: Reynolds number comparison between the region of interest and the different subscale tests of the Boeing-NASA X-48B Blended Wing Body demonstrator program, adapted from Vicroy [22].

Finally, awareness of the appropriate areas of application of dynamic subscale testing should be a priority. Chambers comments that results obtained with free-flight dynamic models should not be extrapolated to other issues that are best analysed using other engineering tools. For example, NASA's experience has shown that free-flight dynamic models are not as appropriate to assess quantitatively handling qualities of an aircraft as full-scale cockpit simulators are [1].

2.1.3 Subscale Flight Testing at Linköping University

Building and flight testing sub-scaled demonstrators is an important part of the aircraft design education at Linköping University, and according to Jouannet et al [2], it provides aeronautical students with a fundamental holistic view of the entire design cycle of an aircraft and a valuable portion of practical work. At a research level, possibilities of using rapid prototyping and subscale flight testing as early-design analysis tools are being investigated and developed, see Staack and Lundström [23]. Apart from various student projects, the research team has currently access to the four advanced
subscales aircraft shown in Figure 2.3. However, for the time being none of them has been equipped with automatic FCS.

(a) Raven business jet, in-house project [24]  
(b) The Midjet single-seat jet, in-house project

(c) Dassault Rafale fighter, test-bed [23]  
(d) Generic Future Fighter (GFF) [3, 25]

**Figure 2.3:** Main subscale research aircraft and concept demonstrators at Linköping University. Images courtesy of the division of Fluid and Mechatronic Systems.

From the flight control point of view, the Generic Future Fighter (GFF) project deserves more attention: this research aircraft is a concept demonstrator of a fifth generation Generic Future Fighter with stealth capabilities. The design of the full-scale system was carried out by Saab AB and the Swedish Defence Research Agency (FOI), while the manufacturing and operation of a 13 percent jet-powered research demonstrator was assigned to Linköping University, shown in Figure 2.4. See references [3] and [25] for detailed information. Even for the demonstrator, the advanced configuration with multiple lifting and control surfaces requires complex control mixing and calls for adaptive control laws. Furthermore, certain aircraft design characteristics (such as relaxed longitudinal stability) cannot be properly analysed without an electronic flight control system onboard, which is the fundamental motivation of this thesis.

### 2.2 Unmanned Aircraft Systems Technology

As commented earlier, over the last decades subscale free-flight research aircraft have evolved from rudimentary remotely piloted devices and data loggers to highly advanced interconnected systems which can be perfectly included into the industry’s thriving field of Unmanned Aircraft Systems (UAS). In order to provide with a better understanding of the systems evaluated throughout this thesis, it is considered convenient to offer here a brief introduction to the UAS field and the related technology
2.2 Unmanned Aircraft Systems Technology

Figure 2.4: General full-scale layout (left), demonstrator CAD model (centre) and finished demonstrator (right) of the Generic Future Fighter (GFF). Images reproduced from [25], courtesy of Linköping University and Saab AB.

that can be used in subscale flight testing.

Although unmanned aircraft are as old as aviation itself, nowadays, the media gives special attention to this expanding sector with a still-young regulation. Confusing information about the commonly, but inappropriately, called “drones” can be heard almost everywhere. The terms Unmanned Aerial Vehicle or Unpiloted Aerial Vehicle (UAV) are widely accepted to define an aircraft designed to operate with no human pilot onboard [26]. However, in line with Barnhart [27] and also with the principal aviation authorities, the general denomination Unmanned Aircraft Systems (UAS) was preferred here since it emphasises the inclusion of all the necessary elements of the system beyond the flying vehicle itself, such as command and control elements, launch and recovery elements, and human elements, among others. See the European regulation in [28] and specialised literature such as [27] for further information.

Regarding the operation of the system, a straightforward classification is presented in Figure 2.5 according to the different operation principles. If the aircraft has the ability to fly and navigate without any human intervention and it has received full authority (types 4 and 5), it is known as an autonomous system. This type of system has no direct application in subscale flight testing other than to serve as a backup security feature in case contact is lost. However, if the aircraft is directed with navigation commands or piloted directly by a remote human operator through a radio communication system, it is referred to as a Remotely Piloted Aircraft System (RPAS), also according to the last international standards [29]. The flying platform alone is then known as a Remotely Piloted Aircraft (RPA) or Remotely Piloted Vehicle (RPV). The control by navigation commands such as waypoints and vectors (type 3) has also little application in subscale flight testing and it can only serve as a way of alleviating the pilot workload at certain flight phases. Consequently, the operation methods of interest are here the types 1 and 2, in which the aircraft attitude is under direct control of the pilot. The main difference between these two is that in type 1 the FCS (and therefore all flight data real-time processing) is located at the ground station and only the final commands for the control surface actuators are transmitted to the aircraft. This method allows using larger computational resources and thus, testing much more complex control laws and signal filters. This is the approach chosen for example by NASA in the AirSTAR
2 Background

project, see [16]. On the other hand, this system relies upon a solid data link that was not reachable with the limited resources available for this study. Therefore, an operation of the type 2 with the FCS onboard the aircraft was preferred. Nevertheless, the miniaturisation of powerful processors in recent years has substantially increased the capabilities of onboard FCS even in small scales, as seen in complex systems such as in the flapless fluidic flight control of the DEMON UAV demonstrator in the BAE FLAVIIR project [30, 31]. Provided the actual evolution of these electronic systems, it is reasonable to expect that the capability advantages of the ground-based FCS will soon become minor.

![Diagram of aircraft systems classification]

**Figure 2.5**: Basic classification of aircraft systems according to the operation method.

2.2.1 Generic Radio Control Systems for Aircraft Models

Hobby-type radio-controlled (R/C) aircraft models for hobby can be classified according to its operation as of type 1 in Figure 2.5: the control surface deflections are commanded directly from the ground, where the only FCS would be the (usually computerised) transmitter of the pilot which applies the basic control mixing, trim, or settings needed. Hence, the pilot cannot delegate the control since the aircraft does not have the ability to resume autonomous flight. Although there is usually no control augmentation, the whole system could be technically considered a fly-by-wire (FBW) system since there is no mechanical connection between pilot and control surfaces.

Taking advantage of the miniaturisation of the avionics used by autonomous UAS, in recent years it has been increasingly usual to equip enthusiast-level aircraft models with relatively advanced FCS that, in some cases, allow not only operations of the type 2, but also 3 and 4 with autonomous navigation and landing. Therefore, it is
sometimes difficult to tell between advanced amateur R/C systems and professional or military UAS apart from their significant differences equipment quality and complexity of the missions. Indeed, the purpose of this work was to explore the possibilities that this approach opens for low-cost research flight testing at education institutions, which has been traditionally carried out with standard direct control from the ground. The interest was not the autonomous navigation of the model, but rather the possibility to include control augmentation systems able to support different control laws and stability configurations.

2.3 Flight Control Techniques

This purpose of this section is to introduce briefly the main flight control techniques used in modern fixed-wing aircraft, rather than to offer an exhaustive review that would definitely be out of the scope of this document. Although according to Balas [32] advanced flight control design has been a very active research area during recent decades, very little documentation is openly available in literature to be used as a reference handbook by designers. In fact, as this author comments, the know-how required to design advanced flight control systems is not easily transferred and some companies consider it to be intellectual property. However, some general concepts and common control design approaches that are relevant to this study will be outlined.

First, it is important to mention that FCS can be designed for different types of control objectives. Indeed, in advanced aircraft, the control objective may be changed during operation depending on the flight condition, according to authors such as Stevens and Lewis [33]. Leaving apart automatic navigation or special operations and focusing on aircraft attitude control, Härkegård states in [34] that for general manoeuvring in the longitudinal direction both normal acceleration $n$ and pitch rate $q$ are suitable control variables: the latter is usually very intuitive for the pilot since it correlates approximately with the traditional direct command of the elevator deflection. On the other hand, normal acceleration or load factor is commonly used as a control objective in high performance military aircraft and commercial transports such as the Airbus A320 and A340 [32]. This is especially interesting at higher flight speeds since it is directly correlated with the acceleration experienced by the crew and the loads on the structure, see [33] for more details. As stated by Härkegård [34], normal acceleration is closely coupled to the angle-of-attack, which is a better alternative for slow flight speeds and non-linear approaches. Therefore, angle-of-attack command control designs are also common, and they might be more convenient for subscale models. In fact, according to the same author, modern fighters usually combine both load factor and angle-of-attack command control designs depending on the flight condition. Although control in the lateral directions is not discussed in this study, it is convenient to mention that roll rate and sideslip command control systems are the most common approaches conforming also to Härkegård [34].

Moreover, according to Balas [32], the use of multivariable control techniques to design the flight control laws is standard in modern aircraft, and not only in the military
2 Background

sector: for example, in large airliners such as the Airbus A340 and A380, it is common to incorporate a structural-mode suppression controller in order to reduce structural mode vibrations and fuselage response to turbulence [32]. Although most of these considerations are irrelevant for a first approach in subscale FCS design, there are some advanced features that could be important to include: modern flight control laws incorporate automatic command limiters in order to prevent the aircraft from ending up in an out-of-control situation or over-stressing the structure. Examples of this are the angle-of-attack (alpha) limiter of the F-16 fighter [33], the manoeuvre load limiter (MLL) of the Saab Gripen fighter [35], and the manual pitch limiter (MPL) of the F-35 fighter.

It is important to notice that differently from helicopters and other powered-lift aircraft, the control response of an aeroplane is dependent on the control surfaces lift, and thus, on the square of the airspeed [36]. Modern FCS vary the controller gains according to the airspeed in order to maintain a regular response across the entire flight envelope [33, 35]. A common approach is to use dynamic interpolation of reference values stored in look-up tables. This is also an important feature that should be included in the control system design for subscale aircraft, especially for the case of responsive unstable models at relatively high speeds. More advanced systems could even include adaptive control designs able to account for malfunctions or inoperative actuators and to modify the flight control laws accordingly. A good example of such system applied to a subscale research aircraft is given by Gregory et al in [37], but however, adaptive control designs are out of the scope of this thesis work.

Regarding control distribution, in traditional aircraft configurations the attitude in roll, pitch and yaw is controlled by the typical aerodynamic control surfaces: ailerons, elevator, and rudder, respectively; see, e.g., Gudmundsson [36]. However, in modern advanced configurations it is usual to find more control surfaces than the traditional three due to performance and redundancy aspects, as stated by Härkegård in [34]: in a delta-canard configuration, pitch control is achieved by combining symmetric elevon (portmanteau of “elevator” and “aileron”) deflection and canard deflection, as shown later in Figure 3.21. While the elevon deflection generates certain phase-response due to aerodynamic couplings, the canard deflection generates an immediate response in the commanded direction. Hence, both are usually combined in a similar magnitude for pitch attitude changes demanded by the pilot, as reported by the same author. However, in order to minimise the significant drag generated by the canard surfaces, these are usually aligned with the airflow during steady flight and the pitch stabilisation and longitudinal trim are carried out only by the elevons. Although this is commonly used during cruise segments according to this author, it might not be appropriate at lower speeds or high angles-of-attack. Due to the lack of more detailed documentation, this approach is however used as a starting point in this study.
3

Theoretical Models

A theoretical study was carried out in order to investigate dynamics of the small and light flying platforms and to evaluate if analytic results could be useful for estimating and adjusting the response of the real control system. This was done by analysing the different test-bed aircraft using common aircraft-design methods and conventional preliminary-design tools. This chapter presents the development process of the theoretical models and how they were assembled together in a flight dynamics analysis program written in MATLAB.

Various test-bed aircraft with different configurations were used in testing different integration phases as will be shown later in chapter 5. In order to limit the extent of this report, the complete theoretical study is presented here only for one of the platforms: a delta-canard, pusher configuration which resembles a Saab JAS 39 Gripen aircraft at approximately nine percent scale, shown in Figure 3.1. As a general rule, the study considers a relaxed longitudinal static stability setting with a negative static margin of 10 percent of the mean aerodynamic chord (MAC). The basic, control-fixed neutral point (NP) of the aircraft is taken as reference for these computations, i.e., the NP computed with both canard and elevons fixed at the neutral position. Notice that in general, references to static stability throughout this study are assumed to be control-fixed since all control surfaces are tightly connected to servo-actuators that in this scale will not move freely due to aerodynamic forces. One exception to this is the study of the neutral point and static stability margin alteration when the canard surfaces are deliberately disconnected and free to rotate, leaving the aircraft in a pure delta configuration.

![General views of the delta-canard test-bed aircraft modelled in OpenVSP software. This aircraft is a very simplified model of the Saab JAS 39 Gripen fighter at approximately nine percent scale.](image)

**Figure 3.1:** General views of the delta-canard test-bed aircraft modelled in OpenVSP software. This aircraft is a very simplified model of the Saab JAS 39 Gripen fighter at approximately nine percent scale.

Detailed characteristics of this and the other test-bed aircraft can be consulted in chap-
3 Theoretical Models

Nevertheless, the analysis process described here is essentially identical to that applied to the other test-bed aircraft, and the development of the analytical models was also considerably similar.

3.1 Mass and Inertia Analysis

The small size of subscale aircraft is a great advantage when it comes to study their mass properties since it is normally easy to measure accurately the mass of the airframe and all components onboard, or to compute its centre of gravity (CG). This was the case for the test-bed models and their components, which where weighed easily with a high-precision scale. The measurement of the inertia characteristics is however more challenging. Thanks again to the small size, inertial testing can provide with accurate experimental estimations for subscale aircraft with mid or high wing loadings, as described by Lundstrom in [4] and Jordan et al in [18]: the aircraft is suspended by wires and set in pendulum motion. Its moments of inertia can be later derived from the observed period, averaged over a certain number of cycles. Nevertheless, this method requires to take into account the air damping effects and to subtract them from the initial values. For example, this was achieved by Jordan et al by measuring the characteristics of an extremely light air damping model made with the same geometry, and then obtaining the correction factors from the differences observed between the CAD model prediction and the test measurements. Anyhow, inertial testing could not be used here since the test-bed models are non-dynamically-scaled and they have a very low wing loading, which makes the air damping effects significant enough to invalidate measurements. Alternatively, moments and products of inertia were computed by using CAD tools together with analytical analysis. A summary of the main mass and inertia properties and their respective estimation methods can be found in Table 3.1.

<table>
<thead>
<tr>
<th>Property</th>
<th>Symbol</th>
<th>Estimation method</th>
</tr>
</thead>
<tbody>
<tr>
<td>Total mass of the aircraft at a specific condition</td>
<td>( m_{AC} )</td>
<td>Direct measurement</td>
</tr>
<tr>
<td>CG location in space</td>
<td>( x_{CG}, y_{CG}, z_{CG} )</td>
<td>Direct measurement</td>
</tr>
<tr>
<td>CG location in percentage of the MAC</td>
<td>( x_{CG} )</td>
<td>Direct measurement</td>
</tr>
<tr>
<td>Moment of inertia about the ( x-, y-, ) and ( z- ) axes</td>
<td>( I_{xx}, I_{yy}, I_{zz} )</td>
<td>CAD and analytical</td>
</tr>
<tr>
<td>Product of inertia in the ( xy-, xz-, ) and ( yz- ) planes</td>
<td>( I_{xy}, I_{xz}, I_{yz} )</td>
<td>CAD and analytical</td>
</tr>
</tbody>
</table>

Two CAD models of the Gripen test-bed aircraft airframe were created using Dassault Systèmes CATIA V5 and NASA’s OpenVSP software respectively. An homogeneous density of the foam material was assumed, as well as a uniform distribution of extra airframe mass such as adhesive and joints. The inertia characteristics of the airframe
3.1 Mass and Inertia Analysis

were obtained directly from these CAD models while, on the other hand, individual components such as battery, motor, avionics, and servos were approximated by discrete point loads and they were located in three-dimensional space according to their respective CG. Gudmundsson offers in [36] a complete review of this technique. The discrete point loads were then added to the total moments and products of inertia by applying analytical expressions adapted from Gudmundsson:

\[
I_{xx} = \iiint (y^2 + z^2) \, dm \Rightarrow I_{xx} = \sum_{i=1}^{N} (y_i^2 + z_i^2) m_i + I_{xx_{airframe}}
\]

\[
I_{xz} = \iiint xz \, dm \Rightarrow I_{xz} = \sum_{i=1}^{N} x_i z_i m_i + I_{xz_{airframe}}
\]

where the first expression is an example of the mass moments of inertia about the \(x\)-axis; and the second shows the product of inertia with respect to the \(x\)- and \(z\)-planes. Computations were carried out automatically by the main MATLAB application previously loaded with the measured coordinates and masses of the aircraft components, located with respect to the fixed aircraft reference datum.

Notice that even though the points were initially located with respect to a fixed aircraft datum, the code first computed the aircraft CG and then translated all references so the CG was the origin of coordinates about which all moments and products of inertia were computed. Same process was followed for the change of \(x\)- and \(z\)-axes.

**Figure 3.2:** Gripen test-bed aircraft modelled in MATLAB showing airframe structure imported from CAD, and the system of discrete point loads corresponding to different components.
orientation between the airframe analysis and flight dynamics standards. The application of this technique to the test-bed aircraft can be seen graphically in the output from the MATLAB application shown in Figure 3.2, and detailed numerical results can be consulted in the appendices. The low inertia values were consistent with the low weight of the aircraft and suggested that there was no big resistance to motion even for the pitch axis. As expected from the aircraft symmetry according to Nelson [38], results also confirmed that $I_{xy} = I_{yz} = 0$.

### 3.2 Aerodynamic Model

Estimating the aerodynamic characteristics of the aircraft is always a trade between accuracy, available resources and time. Semi-empirical models such as DATCOM were not suitable in this case due to the extremely low Reynolds number values, see [39], and therefore analytical methods had to be followed. If the analysis is focused on small deviations from straight and level flight conditions and non-linear regions are avoided, simple mathematical models such as lifting-line theory or inexpensive computational fluid dynamic (CFD) methods such as Weissinger’s method and vortex lattice method (VLM) may be applicable with success depending on the level of simplification and the suitability of the aircraft configuration. If the aircraft configuration is not appropriate for these methods or if there is interest in exploring non-linear flight conditions, it would be necessary to use of higher-order numerical methods such as CFD Navier-Stokes solvers, with the consequent increase in computational requirements and time. However, as Kroo points out in [40] aerodynamic analysis for dynamics simulation and control system design does not require a detailed view of the flow field properties, but rather the integrated effect of the flow on the aircraft forces and moments at many different flight conditions, which sometimes makes CFD Navier-Stokes codes not necessary or even desirable for control system design. Considering that for this study time and computational resources were both very limited, these high-fidelity methods were discarded and the analysis was carried out by using a low-fidelity VLM despite the fact that the aircraft configuration did not fit well the requirements of this method.

Since this study is limited to longitudinal stability and control, only longitudinal aerodynamic characteristics were analysed. This leads to some further assumptions such as symmetric deflection of the canard surfaces in order to avoid asymmetric induced lift or asymmetric vortex wake, which can lead to aerodynamic cross-coupling between the longitudinal and lateral equations of motion, as stated by Nelson in [38]. The complete aerodynamic analysis process that was followed here is outlined in Figure 3.3. Although most processes were fully integrated in the main MATLAB flight dynamics simulator code, the VLM computations were performed out of the loop in a discrete manner for a selected number of different conditions in order to save development time. However, it would be straightforward to integrate in-the-loop aerodynamic computations in future versions of the program, or to create a complete database of aerodynamic data at numerous conditions that could be used for non-linear simulations.
Figure 3.3: Aerodynamic analysis process followed in this study.

### 3.2.1 Vortex Lattice Method

Aerodynamic coefficients were estimated by using the vortex lattice method (VLM) by virtue of its low computation time. In this simple panel method the wing is represented by a surface on which a grid of horseshoe vortices is superimposed. A complete review of the VLM can be found in Bertin and Cummings [21] and it will not be repeated here. However, it is important to mention several identified issues that arose with the application of the VLM for this particular case:

**Satisfactory results limited to low angles-of-attack**: according to Nelson [38], detached flow is of critical importance for the aerodynamics of a delta wing at high angle-of-attack (also referred to as AOA or alpha, $\alpha$), but the VLM cannot model these non-linearities and therefore any separated vortices. The computation could however be improved by, for example, introducing experimental corrections based on wind tunnel data for similar delta geometries.

**Some design features are neglected**: since the VLM cannot compute detached flow it is pointless to model certain vortex-related design characteristics such as the “dog-tooth” leading edge, which in this aircraft play a significant role at higher AOs. The wing geometry was therefore simplified in order to avoid eventual mesh-induced errors. Additionally, all airfoils were approximated by flat plates.

**Effect of the fuselage is significant**: the influence of the fuselage volume in the aerodynamic coefficients is expected to be significant for this configuration and therefore should be taken into account, see Bertin and Cummings [21]. The approximation of the complex three-dimensional body shape by thin plates can be inaccurate and may introduce errors, especially regarding the effect of the nose section on the canard surfaces (at different height), and these two over the main wing. The solution applied here was a compromise between geometrical fidelity and error margin assumption.

**Propulsion system effects**: flow velocity induced by thrust over the rear-inner area of the aircraft may be significant at slow speeds and may affect the stability.
3 Theoretical Models

derivatives, especially for the pusher-propeller configuration of the Gripen test-bed model. Effects on the flight control system may be expected, such as increased control authority along the inner sections of the elevons while outer sections, away from the induced stream, would have very limited or null control authority. It could be possible to improve the VLM code by modelling the helical-wake of the propeller using cylindrical vortices, but this idea has not been yet implemented.

The open-source Tornado VLM code for MATLAB [41] was used for computations. The numerical parameters needed to build the aircraft geometry with flat lifting surfaces were obtained by degenerating the 3D model using OpenVSP. The aircraft was divided into separate lifting surfaces in a similar manner to that presented by Staack et al for a F-16 fighter in [42], which in this case consisted of nose, fore-body, canard and main body section, main wing including elevons, and aft fuselage, as shown in Figure 3.4. However, since the analysis here is only longitudinal all vertical surfaces were neglected in order to save computer resources. An additional version without canard was also modelled with the purpose of studying the floating (free) canard setting. A comparison between the geometry modelled in Tornado VLM and the accurate 3D model is shown in Figure 3.5. Mesh convergence was checked by increasing the panel density with no major change in the results considering the overall low accuracy of the method. The analysis was executed for different flight conditions assuming always inviscid flow, small AOAs, and similarly to the physical test-bed aircraft the lifting surfaces were modelled as flat plates without camber. After computation, VLM results for each flight condition were automatically loaded by the main flight dynamics application.

Figure 3.4: Tornado VLM model of the Gripen test-bed aircraft showing lifting surfaces layout, eventual deflection of the elevons, and including the mean aerodynamic chord.
3.2 Aerodynamic Model

3.2.2 Neutral Point Estimation

An accurate estimation of the neutral point of the aircraft is here of critical importance since it is used as the reference for quantifying the longitudinal static stability, the so-called static margin. This and other useful definitions can be found in Nelson [38] and Gudmundsson [36], among other authors. In this paper the basic, control-fixed neutral point (NP) of the aircraft is normally taken as reference, i.e., the NP computed with both canard and elevons fixed at the neutral position. This choice was motivated by the fact that all control surfaces are tightly connected to servo-actuators that at this scale will not move freely due to aerodynamic forces, as observed in other model aircraft of similar scale. However, as Claréus describes for the real Gripen aircraft in [43], a good characteristic of this delta-canard configuration is that the canard surfaces can be deliberately disconnected and freed to rotate: the aircraft would then become a pure delta configuration and the neutral point location would be altered to the extent that the negative (unstable) static margin would be transformed into a neutral or marginally positive static margin. Even though the added mechanical complications made difficult to test this characteristic with the scaled model, the free-canard configuration was also included in this theoretical study.

The Tornado VLM code includes a numerical computation of the neutral point location based on the derivation of the aerodynamic moment coefficient. Although this seemed a reasonable solution, especially considering the complex configuration and the possible aerodynamic effects of the body, there was interest in investigating the agreement between this VLM and the simplified analytical equations often used in conceptual design. Therefore, the control-fixed neutral point location along the longitudinal axis was computed by using both the VLM and the analytical expressions given by Gudmundsson in [36] for all the test-bed models. Results are listed in Table 3.2 and show a very good agreement between both methods for the models with conventional con-
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configuration (Balsa and Cub), as well as for the elementary Rafale delta-canard model. However, in the case of the Gripen model the differences grow up to 13 percent of the MAC (approximately 5 cm). This can be caused by the significant effect of the body, which is modelled as a lifting surface in the VLM while it is neglected in the analytical equations, see Gudmundsson [36].

Table 3.2: Computation of the neutral point (NP) location using both analytical method and VLM for the different model aircraft tested. Neutral point is control-fixed except for the last case where the canard surfaces are free to rotate.

<table>
<thead>
<tr>
<th>Model</th>
<th>MAC m</th>
<th>Analytic NP</th>
<th>VLM NP</th>
</tr>
</thead>
<tbody>
<tr>
<td>Balsa</td>
<td>0.159</td>
<td>53%</td>
<td>55%</td>
</tr>
<tr>
<td>Cub</td>
<td>0.227</td>
<td>43%</td>
<td>48%</td>
</tr>
<tr>
<td>Rafale</td>
<td>0.309</td>
<td>11%</td>
<td>9%</td>
</tr>
<tr>
<td>Gripen</td>
<td>0.381</td>
<td>6%</td>
<td>19%</td>
</tr>
<tr>
<td>Gripen free-canard</td>
<td>0.381</td>
<td>41%</td>
<td>32%</td>
</tr>
</tbody>
</table>

Even though differences were in general sufficiently small, it was decided to use the values provided with the VLM code since this did not increase the computation time and they were expected to be somewhat more accurate, especially when the body contribution was significant. Thus, the computed NP location values for both standard and free-canard cases were directly transferred to the main flight dynamics MATLAB program together with other relevant geometric and aerodynamic parameters. Figure 3.6 is an extract from this program that shows the longitudinal neutral point location on the aircraft compared with the MAC and the CG location.

![Figure 3.6: Gripen test-bed aircraft modelled in MATLAB showing the computed longitudinal location of the control-fixed and canard-free neutral points. Notice that while the current CG location leads to a negative static stability margin of $-10$ percent of the MAC, this becomes 3 percent positive if the canard surfaces are freed.](image)
3.2 Aerodynamic Model

3.2.3 Parasite Drag Estimation

Bertin and Cummings comment in [21] that as the size of the vehicle decreases the parasite drag and especially the skin-friction contribution becomes dominant over other parameters. Since the VLM assumes inviscid flow, the parasite or zero-lift drag needs to be estimated by an alternative method. Experimental techniques were not available and advanced CFD Navier-Stokes solvers present additional complications in this case: according to data by Bertin and Cummings included also in [21], most parts of the test-bed aircraft would present Reynolds number values corresponding to the turbulent transition region, which has a significant effect on parasite drag and it is considerably difficult and expensive to simulate using CFD methods, as Versteeg and Malalasekera state in [44]. Once again, practicality was favoured and a traditional drag bookkeeping method was used.

The straightforward method proposed by Bertin and Cummings in [21] was chosen. Similarly to other drag estimation methods for preliminary design, the basic approach consisted of the following steps:

1. Estimation of an equivalent flat-plate skin-friction coefficient for each component: The Prandtl-Schlichting formula was applied to both lifting surfaces and bodies,

\[ \tilde{C}_f = \frac{0.455}{(\log_{10}Re)^{2.58}} \]  

(3.2)

Notice that this formula does not include the correction for laminar flow and it is given in the form proposed by Kroo in [45]. Since in this case turbulent transition was difficult to predict, it was decided to estimate the “worst-case scenario” for parasite drag and to assume complete turbulent flow. Reynolds number values were computed at sea level conditions according to the mean aerodynamic chord for lifting surfaces and to the stream-wise length for other bodies.

2. Correction of the skin-friction coefficient for surface roughness: the previous values were corrected according to the correlation given in [21] assuming an equivalent sand grain roughness of \( k_{foam} = 0.1 \) mm for the foam material.

3. Application of a form factor correction, \( K \), to take into account super-velocities and pressure effects: form factor values were estimated separately for lifting surfaces and bodies according to their respective thickness or fitness ratio, following the relations given in [21].

4. Conversion of the corrected skin-friction coefficient of each component into an aircraft drag coefficient and sum of all components, i.e.

\[ C_{DP} = \sum_{i=1}^{N} \frac{K_i \tilde{C}_f S_{wet_i}}{S_{ref}} \]  

(3.3)

where the precise wetted area value for each component was obtained from the CAD model using OpenVSP, as shown in Figure 3.7.
5. Additional increase: the total value was increased by 10 percent to take into account interference effects and other miscellaneous terms.

Figure 3.7: Accurate mesh-based computation of the wetted area for the Gripen test-bed aircraft using OpenVSP.

All operations were integrated in the MATLAB application and were carried out automatically according to the selected flight conditions. Since Reynolds number values depend on the airspeed, drag results vary slightly at different flight conditions. However, the illustrative estimations listed in Table 3.3 for $V = 20 \text{ m s}^{-1}$ are a good example of common flight conditions.

Table 3.3: Parasite drag estimations for the Gripen test-bed aircraft at sea level, $V = 20 \text{ m s}^{-1}$.

<table>
<thead>
<tr>
<th>Item</th>
<th>$Re \times 10^{-5}$</th>
<th>$C_f$</th>
<th>$K$</th>
<th>$S_{wet} \text{ m}^2$</th>
<th>$C_{D_p}$</th>
</tr>
</thead>
<tbody>
<tr>
<td>Main wing</td>
<td>5.223</td>
<td>0.0051</td>
<td>1.02</td>
<td>0.3568</td>
<td>0.0074</td>
</tr>
<tr>
<td>Canard</td>
<td>1.670</td>
<td>0.0064</td>
<td>1.05</td>
<td>0.0581</td>
<td>0.0016</td>
</tr>
<tr>
<td>Vertical tail</td>
<td>3.190</td>
<td>0.0056</td>
<td>1.03</td>
<td>0.0774</td>
<td>0.0018</td>
</tr>
<tr>
<td>Spine</td>
<td>4.381</td>
<td>0.0052</td>
<td>1.05</td>
<td>0.0086</td>
<td>0.0002</td>
</tr>
<tr>
<td>Rear strakes</td>
<td>2.054</td>
<td>0.0061</td>
<td>1.08</td>
<td>0.0090</td>
<td>0.0002</td>
</tr>
<tr>
<td>Wingtip rails</td>
<td>2.533</td>
<td>0.0059</td>
<td>1.03</td>
<td>0.0151</td>
<td>0.0004</td>
</tr>
<tr>
<td>Fuselage</td>
<td>17.53</td>
<td>0.0040</td>
<td>1.13</td>
<td>0.4095</td>
<td>0.0075</td>
</tr>
<tr>
<td>Canopy</td>
<td>5.477</td>
<td>0.0050</td>
<td>1.26</td>
<td>0.0380</td>
<td>0.0010</td>
</tr>
<tr>
<td>Antenna</td>
<td>2.328</td>
<td>0.0060</td>
<td>1.10</td>
<td>0.0068</td>
<td>0.0002</td>
</tr>
<tr>
<td>Miscellanea</td>
<td>0.0020</td>
<td></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td>Total</td>
<td></td>
<td></td>
<td></td>
<td>0.0223</td>
<td></td>
</tr>
</tbody>
</table>

3.2.4 Propulsion System Effects

Among others, Nelson in [38] and Gudmundsson in [36] state that propulsion can have a significant effect on the aerodynamics of the aircraft and it can affect significantly both longitudinal trim and stability derivatives of the aircraft. Leaving force
effects due to thrust-line offset aside, it is clear from Nelson that it is needed some estimation of the propeller influence in the aerodynamic derivatives, and especially in the pitching moment \[38\]. Nelson continues stating that although it is possible to derive a simple expression to account for the propeller contribution to \(C_{m_{\alpha}}\), the actual influence of the propulsion system in the aircraft stability is extremely difficult to estimate analytically due to the indirect aerodynamic effects, such as the modification of the stream near the propeller which can alter the control surfaces efficiency as mentioned earlier. Although these effects can be significant for conventional aircraft configurations in which the propulsion system interacts strongly with all surfaces downstream, they were expected to be minor in the case of the Gripen test-bed aircraft due to the pusher propeller located far back. Therefore, no further investigation was carried out and no direct corrections were applied to the aerodynamic model.

However, a basic estimation of the thrust and power requirements is essential for the analysis of performance characteristics. The thrust and power required for straight level flight were estimated in a fairly straightforward manner based on the previous drag estimations. Details can be found in any aircraft performance primer such as Anderson [46]. The horizontal thrust required is equal to the total drag and therefore total thrust required can be estimated by using the relation:

\[
T_{\text{req}} = \frac{D}{\cos \alpha} = \frac{Q S_{\text{ref}} C_D}{\cos \alpha} \tag{3.4}
\]

where \(Q\) is the flight dynamic pressure, as usual:

\[
Q = \frac{1}{2} \rho_{\infty} V_{\infty}^2 \tag{3.5}
\]

However, for small AOAs the correction for alpha can be neglected.

The fixed-pitch pusher propeller used in the Gripen test-bed aircraft was 7 × 6 inches, i.e. a diameter of 0.178 m. With the aim of simplifying calculations a constant propeller efficiency of \(\eta_p = 0.5\) was assumed, value which seemed reasonable according to the low Reynolds number tests for similar propellers carried out by Brandt and Selig in [47]. Hence, in line with Anderson [46], the required power computation was included in the program in the form:

\[
P_{\text{req}} = \frac{T_{\text{req}} V}{\eta_p} \tag{3.6}
\]

### 3.2.5 Some Numerical Results

The estimated aerodynamic coefficients and characteristics of the Gripen test-bed aircraft were obtained after computing together all previous steps. The main MATLAB application scheduled the operations, gathered the necessary data from the different tools, and interpolated the values for the desired airspeed. Although detailed numerical values can be consulted in the appendices, for the sake of a better comprehension some relevant aerodynamic estimations for the Gripen test-bed aircraft are graphed in Figure 3.8 for different airspeeds. However, stability coefficients will be discussed in more detail later in section 3.4.
3.3 Mechatronic System Model

While aerodynamic and inertia models were developed to represent the dynamics of the test-bed aircraft, it is essential in this study to develop also appropriate equations or transfer functions to represent the elements that make up the control system. As introduced earlier in section 2.2.1, the complete control system of a remotely piloted aircraft is essentially a FBW system on its own, and it comprises a complex mix of electronic devices, data transmission systems, and mechanical components, as illustrated in Figure 3.9. Each one of these components can be extremely complex to model accurately if the behaviour of the internal electronics and processors are taken into account. However, a more straightforward approach was followed here for most of the components since the main goal was again to evaluate the functionality at a big scale. In addition, the extremely low latency levels and the high precision achieved by hobbyist-type modern R/C systems permit neglecting all components from the pilot mechanical input until the FCS electronic signal input, which here were considered equal and instantaneous. Precise data and characteristics of R/C transmitter systems are usually available from manufacturers. On the other hand, Nelson states in [38] that although it is reasonable to assume that the transfer functions of some elements such as gyroscope sensors and amplifiers can be represented by simple gains $k$, appropriate models must be defined for the rest of the control system components. Regarding stability and control augmentation systems, different controllers were modelled trying to replicate the real devices used for the practical testing.
3.3 Mechatronic System Model

3.3.1 Control Surface Actuator

In the test-bed aircraft each control surface is actuated directly by a common electric R/C servo, which is a complex automatic mechatronic system on its own. Therefore, the development of a transfer function for the entire servo requires to have a closer look to its components. First, according to Nelson [38] the torque produced by the electric motor of a common servo is proportional to the control voltage $v_c$ by a motor constant $k_m$, so the relation between the angular position of the motor shaft and the control voltage can be approximated by the transfer function

$$\frac{\theta_{shaft}}{v_c} = \frac{k_m}{Is^2}$$ (3.7)

where $I$ is the electric current. Moreover, according to the same author servo motors normally incorporate a simple rate feedback as shown in Figure 3.10. The transfer function for this system can be defined as

$$\frac{\theta_{shaft}}{v_c} = \frac{k}{s(\tau_ms + 1)}$$ (3.8)

where $k = \frac{1}{B_m}$ and $\tau_m = \frac{I}{k_mB_m}$ is the motor time constant which measures the motor response to voltage changes. However, according also to Nelson this motor time constant is very small for fast electric motors like the one considered here and it can be neglected [38]. Thus, the transfer function of the servo motor with rate feedback is reduced to

$$\frac{\theta_{shaft}}{v_c} = \frac{1}{B_ms}$$ (3.9)

Following Nelson, a simple position-control servo system could be developed from the electric servo motor model as a first order control system, which was considered enough for the purpose of this study taking into account the limited knowledge of the real servo components. Such first order system shown in Figure 3.11 and it assumes
that the control surface is firmly linked to the servo arm with no multiplication or deformation, $\theta_{arm} = \delta_E$. In line with Nelson, the relation between the commanded signal and the actual control surface deflection can therefore be approximated by the transfer function

$$\frac{\delta_E}{e_c} = \frac{1}{\tau s + 1} \quad (3.10)$$

where the time constant of the servo is given by $\tau = \frac{B_m}{k_f k_a}$. For common electric servo actuators the time constant values are usually between $\tau = 0.05$ and $\tau = 0.25$, being $\tau = 0.1$ a typically chosen value when little information is available [38]. This servo model was also complemented with a limiter function (or saturation block) at the output in order to account for the deflection limits of the real actuator.

**Figure 3.10:** Diagram of an electric servo motor with rate feedback, adapted from Nelson [38].

**Figure 3.11:** Diagram of a positional feedback control servo system for control surface deflection, adapted from Nelson [38]. It is assumed that the control links do not deform and the surface deflection angle is proportional to the servo arm angle.

### 3.3.2 Pitch Rate Gyroscopic Stability Augmentation System

Pitch rate feedback was the first flight control law (FCL) considered here, and it is probably the most straightforward of all. Pitch rate controllers based on closed-loop, single-axis gyroscope sensors are widely available, inexpensive, and easy to implement in small-scale aircraft since they are extensively used to stabilize the directional axis (tail rotor pitch) of R/C helicopters. Although some advanced gyroscopic controllers incorporate complex control algorithms, the most basic and inexpensive models have a simple structure fairly easy to model. Such is the case of the Assan GA-250 MEMS gyroscope [48] used here for the preliminary tests, and described in more detail in section 4.1. This, as most R/C helicopter gyroscopes, has two different FCL modes that can be switched at any time: rate control mode, and angular vector control system (AVCS) mode or “heading hold”. The former is the one of interest here, while the
latter will be used for the second approach.

The controller structure for the rate mode FCL was modelled in MATLAB Simulink environment according to the extensive investigations of this gyroscope carried out by some advanced users such as in reference [49]. As shown in Figure 3.12, the pilot commands the desired angular rate which only goes through a very simple P controller. Then, the response of the system is reported directly by the rate gyroscope sensor and sent back through an analog-to-digital converter (ADC). The response of the controller is therefore proportional to the change in pitch rate, regardless of what the initial or final attitude are. Hence, it acts like a traditional stability augmentation system (SAS) and it only cancels pitch perturbations without maintaining a fixed pitch attitude over time. As a traditional SAS, it could present one problem commented by Stinton in [50]: the feedback loop could provide a command that opposes pilot control inputs and therefore the SAS authority needs to be limited.

\[
\theta_{\text{ref}} = q_{\text{ref}}, P \text{ controller } k_p, \quad e_c, \quad \delta_e, \quad \text{Servo}, \quad \text{Aircraft Dynamics}, \quad \dot{q} = q
\]

Figure 3.12: Diagram of a pitch rate gyroscopic stability augmentation system, based on the Assan GA-250 MEMS gyroscope structure.

### 3.3.3 Pitch Attitude Control System with AVCS Gyroscope

In this second approach the pilot still commands the desired pitch rate, but in absence of input, the controller will make the corrections needed to maintain the current pitch attitude. In this FCL mode, the system behaves more like a control augmentation system (CAS), see [50]. As mentioned before, this is a common directional control method in R/C helicopters known as AVCS or “heading hold”. Here, the pitch attitude FCL was tested with the same single-axis MEMS gyroscope controller Assan GA-250 [48], which only required a radio command to switch to the AVCS-attitude mode. Although there are no changes in hardware, the controller software works in a significantly different way, as depicted in Figure 3.13 based on the investigations found in reference [49]: since the pilot still wants to control the angular rate but the controller now evaluates total angle, the input signal has to be integrated before passing through a PD controller. In addition, the gyroscope sensor registers only angular rate and its signal has to be integrated as well.

However, by applying some mathematical manipulations the integration processes can be joined into a single integration process after the subtraction, and moreover, this can be “absorbed” by the controller turning it into a PI controller. The resulting structure, shown in Figure 3.14, was therefore easier to implement in the MATLAB Simulink model.
3 Theoretical Models

Figure 3.13: Control structure changes for the gyroscope AVCS mode with respect to the rate mode.

Figure 3.14: Pitch attitude control system, based on the Assan GA-250 MEMS gyroscope structure in AVCS mode after some mathematical manipulation.

3.3.4 Advanced Pitch Attitude Control Augmentation System

The single-axis gyroscope controller used in the two previous approaches presents important limitations, especially for three-dimensional free flight as will be discussed in the experimental evaluation, chapter 5. In addition to this experimental observations, authors such as Nelson [38] also claim that including certain supplementary features in the pitch control structure is necessary to obtain satisfactory response through different flight conditions. According to this author, in a pitch attitude controller it is highly recommendable to add an internal loop with pitch rate feedback in order to soften the response of the system. This concept is depicted in Figure 3.15. Moreover, as discussed in section 2.3, FCS in real aircraft often vary the value of the controller gains according to the airspeed. Differently from helicopters, multicopters, and other powered-lift aircraft, the response of an aeroplane is dependent on the control surfaces lift, and thus, on the square of the airspeed [36]. Although in complex systems this is often solved by using dynamic interpolation of values stored in look-up tables, this requires an extensive knowledge and previous testing of the aircraft that may result not practical for small subscale models. Here, a system able to vary the gains according to certain airspeed ratio was considered satisfactory enough to solve this issue.

Figure 3.15: Pitch attitude control system with inner rate feedback loop, based on Nelson [38].
Given this requirements a more advanced COTS controller with a complete three-axis inertial measurement unit (IMU) was adopted for the study of more complex FCL. Although detailed characteristics of the hardware and firmware of this platform will be presented later in chapter 4, it is important to notice here that its open-source software allowed not only a complete analysis of the controller structure but also any custom modification. In fact, the version of the APM-Plane firmware installed in the platform already incorporated the advanced characteristics commented earlier and no modifications were needed in order to obtain a satisfactory pitch attitude CAS with internal rate feedback and airspeed-dependent PID gains. The FCL modes corresponding to direct attitude angle control were denominated FBW A/B in the firmware. Based on its code, available in [51], the control system structure could be depicted as in Figure 3.16.

![Figure 3.16: Structure of the pitch control augmentation system based on the attitude mode FCL of the APM-Plane firmware, version 3.2.1.](image)

### 3.3.5 Advanced Pitch Rate Control Augmentation System

The APM-Plane controller firmware presented earlier could also be switched to a rate mode FCL. In this mode, referred to as ACRO in the firmware, the pitch angle feedback disappears and the integration of the measured pitch rate is no longer needed. By contrast, the input commanded by the pilot is now the desired angular rate, as depicted in Figure 3.17.

### 3.3.6 Proposed Angle-Of-Attack Control Augmentation System

While the pitch rate can be used as a good indicator of the short period motion of the aircraft, the pitch attitude (angle) with respect to the Earth-fixed coordinate frame seems not very relevant during flight phases other than take-off and landing. The use of pitch angle controllers like the ones described before is in most cases only motivated by their simplicity and their excellent performance in combination with a pitch rate feedback loop. Indeed, they are rarely used in full-scale aircraft except in very basic autopilots for some light general aviation aeroplanes. As commented in section...
2.3, during free-flight other parameters offer much better information about the state and performance of the aircraft, such as the angle-of-attack. The relevance of this parameter for the aircraft aerodynamics is extensively explained by Gudmundsson in [36] and will not be commented here. There is strong interest in implementing the AOA as a functional variable in the FCS of the subscale models, since doing so would allow to study and test more realistic FCL. However, as stated by Stinton in [50], this requires an accurate measurement of the AOA, which is in fact difficult to achieve in small models as will be commented later in chapter 4.

Regarding the design of an pitch-axis CAS based on AOA, it was not clear that in this case a simple AOA feedback loop could stabilise successfully the highly responsive aircraft for short-period motion: first, measuring and feedbacking the same first-order variable that it is to be controlled can lead to an unstable system according to basic control theory. In addition, in this particular case the AOA transducer was expected to suffer from certain delay due to the rotating sensor’s own inertia, which could make the situation worse. Third, the AOA measurements were not expected to be sufficiently

**Figure 3.17:** Structure of the pitch control augmentation system based on the rate mode FCL of the APM-Plane firmware, version 3.2.1.

**Figure 3.18:** Proposed design for an angle-of-attack control augmentation system with pitch rate feedback and stall limitation.
“clean” and steady considering the small-scale aerodynamics and the turbulent atmospheric conditions. As a consequence of all this, the preferred solution was to include a pitch rate feedback loop in combination with the AOA loop, since the approximation $\dot{q} \approx \dot{\alpha}$ is valid for very short-period pitch motion. The proposed angle-off-attack CAS structure can be seen in Figure 3.18, and it is based on that presented by Murch for NASA’s AirSTAR platform [52].

3.4 Flight Dynamics Model

The previous models were assembled together with a longitudinal flight dynamics model of the test-bed aircraft in order to evaluate both stability and control parameters. In addition to the static stability analysis (derived from the combination of the mass properties and aerodynamic models) dynamic stability was also analysed here by studying the time history of the motion of the aircraft after it is disturbed from its equilibrium conditions. Good descriptions of aircraft static and dynamic stability are given by Nelson in [38] and Stevens and Lewis in [33]. Since this study was focused in an aircraft with negative longitudinal static stability, the addition of a SAS or CAS in the FCS model was mandatory from the beginning in order to avoid motion divergence, as explained also by Nelson. However, due to time constraints and the limitations of the linear aerodynamics model, the simulator program was only prepared for small deviations from straight level flight. Additionally, the aircraft was modelled as a rigid body and the Earth was considered a flat, inertial reference system.

3.4.1 Definition of Coordinate Frames and Aircraft Variables

Before moving forward, it is convenient to describe the notation and reference systems that were used in the analysis. Following conventional practices described by Nelson [38], Caughey [53], and Härkegård [34], the aircraft motion was characterised by using different coordinate frames depending on the circumstances. An Earth-fixed coordinate frame with its origin at an arbitrary location on the ground is denoted here with the subscript $f$, and it is most useful for describing the position, orientation, and trajectory of the aircraft. A body-fixed reference frame where the origin is the CG of the aircraft is denoted with the subscript $b$, and it is particularly convenient for describing angular motions of the aircraft since its inertia matrix remains constant [34]. However, the body-fixed frame is sometimes rotated about the $y$-axis so that the orientation of the $x$-axis is parallel to the velocity vector $V$ for an initial equilibrium state. These, known as stability axes, are denoted with the subscript $s$, and they present the advantage being aligned with aerodynamic forces, which makes this frame particularly useful for analysing flight dynamics [53]. The change between these different coordinate frames is often done by using Euler angles and it is thoroughly described by both Nelson [38] and [53]. Figure 3.19 shows graphically these different coordinate frames and it also defines the positive sign for the main longitudinal forces and moments applied to the aircraft, namely forces $X$ and $Z$ along the $x_b$- and $z_b$-axes respectively, and the pitching moment $M$ about the $y_b$-axis.
Figure 3.19: Definition of the inertial Earth-fixed coordinate frame $f$, the body-fixed frame $b$ which translates and rotates with the aircraft, and the stability axes $s$ which does the same but it is aligned with the velocity vector $V$ for an initial equilibrium state. Forces and moments relevant to longitudinal motion are shown in the figure and defined positive as indicated.

Although coordinate frames are presented in three dimensions, only two of these are involved in the longitudinal motion analysis and therefore all lateral forces and moments are neglected. The longitudinal relations between the velocity vector of the aircraft, the body-fixed, and Earth-fixed frames are depicted in Figure 3.20. The velocity vector is expressed in terms of the body-fixed frame by the velocity components $u$ and $w$, from Neslon [38]:

$$V = \sqrt{u^2 + w^2} \tag{3.11}$$

Velocity components are related to the angle-of-attack $\alpha$ by

$$\alpha = \tan^{-1} \frac{w}{u} \tag{3.12}$$

which for small $\alpha$ values can be further simplified as

$$\alpha = \frac{w}{u} \tag{3.13}$$

The orientation of the aircraft relative to the Earth-fixed reference frame is determined by the pitch angle $\theta$. The angular velocity about the pitch axis is known as pitch rate and it is denoted by $q$, which according to [34] is given for level flight by

$$\dot{\theta} = q \tag{3.14}$$
3.4 Flight Dynamics Model

Figure 3.20: Definition of the longitudinal velocity components $u$ and $w$, the aircraft orientation pitch angle $\theta$, the aerodynamic angle-of-attack $\alpha$, and the pitch angular rate $q$ for a generic flight state. The figure shows positive values according to the standard convention described by Nelson in [38].

In this delta-canard configuration the longitudinal control variables consist of two canard surfaces (left and right), two elevons (left and right), and the engine throttle setting. In order to avoid lateral forces, deflection of both canards and elevons was considered symmetrical at all situations throughout the analysis. Figure 3.21 shows the deflection sign definitions, similar to those described by Nelson in [38]. Regarding the longitudinal control distribution, the approach explained earlier in section 2.3 for real aircraft with the same configuration was also applied to the model. This means that the trim and stabilisation tasks were carried out only by the elevon as in a standard delta configuration, but the pilot pitch commands for manoeuvres were divided equally between canard and elevon. Therefore, only elevon deflection was included in the SAS model.

Figure 3.21: Longitudinal control variables for the Gripen test-bed aircraft. Control surface deflections are symmetric and positive as indicated.

3.4.2 Forces and Moments

Forces and moments are defined in terms of dimensionless coefficients, the flight dynamic pressure $Q$, and the reference area $S_{ref}$, see Nelson [38]. Consequently, in the
body-fixed coordinate frame forces are expressed as

\[
X = QS_{ref} C_x \\
Z = QS_{ref} C_z
\]  
(3.15)

and in a similar manner, the pitching moment

\[
M = QS_{ref} c_{ref} C_m
\]  
(3.16)

However it is known that aerodynamic forces are often expressed in the stability-axes coordinate frame [34], i.e.

\[
D = QS_{ref} C_D \\
L = QS_{ref} C_L
\]  
(3.17)

which relate to the body-fixed frame by the AOA as follows:

\[
D = -X \cos \alpha - Z \sin \alpha \\
L = X \sin \alpha - Z \cos \alpha
\]  
(3.18)

On the other hand, it is necessary to take into account the gravitational force, which is considered here independent of the flight altitude, acts in the CG, and it is always parallel to the \(z\)-axis of the Earth-fixed coordinate frame. Therefore, according to Nelson [38] its contribution to the longitudinal motion can be expressed in the body-fixed frame as

\[
X_{gravity} = -mg \sin \theta \\
Z_{gravity} = mg \cos \theta
\]  
(3.19)

Finally, the thrust force due to the propulsion system must be included. In this case the propulsion system configuration creates a force aligned with the \(x\)-axis of the body-fixed coordinate frame, namely

\[
X_{thrust} = T
\]  
(3.20)

which is assumed to be equal to the total drag force produced by the aircraft at the initial trimmed flight condition. In the Gripen test-bed aircraft the thrust line is approximately aligned with the CG and no additional contribution to the pitching moment is to be expected according to the same reference, i.e., \(M_{thrust} = 0\). Since no attention is given to lateral motion, other effects such as propeller roll torque are also neglected.

By combining the rigid body dynamic equations and the previous forces and moments it is possible to obtain the equations that govern the motion of the aircraft. Nelson offers in [38] a detailed derivation of the rigid body equations of motion and this will not be repeated here. The resulting kinematic and dynamic equations relevant to this longitudinal motion study are summarized in Table 3.4.

### 3.4.3 Linearised Small-Disturbance Longitudinal Motion

As Nelson points out in [38], the dynamic characteristics of the system can be easily studied if the system is linearised. The linear model is based on the assumption that the aerodynamic characteristics are linear and can be represented by stability derivatives,
Table 3.4: Summary of kinematic and dynamic differential equations used for describing the aircraft longitudinal motion, based on Nelson [38].

<table>
<thead>
<tr>
<th>Equation</th>
<th>Description</th>
</tr>
</thead>
<tbody>
<tr>
<td>( \dot{u} = -qw - g \sin \theta + X/m )</td>
<td>Force equations</td>
</tr>
<tr>
<td>( \dot{w} = qu + g \cos \theta + Z/m )</td>
<td></td>
</tr>
<tr>
<td>( \dot{q} = M/I_{yy} )</td>
<td>Moment equations</td>
</tr>
<tr>
<td>( \dot{\theta} = q )</td>
<td>Body angular velocities</td>
</tr>
<tr>
<td>( \dot{x} = u \cos \theta + w \sin \theta )</td>
<td>Velocity of aircraft in the fixed frame</td>
</tr>
<tr>
<td>( \dot{z} = -u \sin \theta + w \cos \theta )</td>
<td></td>
</tr>
</tbody>
</table>

which is an acceptable assumption for low AOA as according to the same author. The small-disturbance theory, also called perturbation theory, can be used to linearise the dynamic equations presented earlier in Table 3.4 by assuming that the aircraft motion consists of small deviations about an initial equilibrium flight condition about which stability derivatives are evaluated, as described by Nelson [38] and Caughey [53]. This initial flight condition is defined here as a longitudinal equilibrium state, i.e., a steady, unaccelerated, trimmed flight with constant propulsive force where the velocity and gravity vectors are perpendicular. In the body fixed coordinate frame, this means that

\[
\begin{align*}
    u_0 &= V \cos \alpha_{trim}, \\
    w_0 &= V \sin \alpha_{trim}, \\
    q_0 &= 0, \\
    \dot{q}_0 &= 0, \\
    X_0 &\neq 0, \\
    \dot{X}_0 &= 0, \\
    \theta_0 &= \alpha_{trim}, \\
    \dot{\theta}_0 &= 0, \\
    Z_0 &\neq 0, \\
    \dot{Z}_0 &= 0,
\end{align*}
\]

(3.21)

where the subscript 0 denote the original equilibrium state. Notice that it would be possible to further simplify the equations with \( u_0 = V \) and \( w_0 = 0 \) if the stability axes are used. Furthermore, all state variables can be expressed by a reference value plus a disturbance:

\[
\begin{align*}
    u &= u_0 + \Delta u, \\
    q &= +\Delta q, \\
    w &= w_0 + \Delta w, \\
    \theta &= \theta_0 + \Delta \theta,
\end{align*}
\]

(3.22)

Hence, by making certain approximations and neglecting high order perturbation terms as indicated by Nelson and by Caughey, the force and moment equations from Table 3.4 can be linearised and rewritten as:

\[
\begin{align*}
    \Delta \dot{u} &= -\Delta qw_0 - \Delta \theta g \cos \theta_0 + \Delta X/m, \\
    \Delta \dot{w} &= \Delta qu_0 - \Delta \theta g \sin \theta_0 + \Delta Z/m, \\
    \Delta \dot{q} &= \Delta M/I_{yy},
\end{align*}
\]

(3.23)

where the perturbations in aerodynamic forces and moments \( \Delta X, \Delta Z, \) and \( \Delta M \) are functions of both the state variables and control inputs [53]. However, some of the factors contribute very little to the aircraft response and can be neglected following Nelson’s criteria [38], which added to the assumption of fixed thrust setting \( \Delta \delta_T = 0 \).
yields

\[
\begin{align*}
\Delta X &= \frac{\partial X}{\partial u} \Delta u + \frac{\partial X}{\partial w} \Delta w \\
\Delta Z &= \frac{\partial Z}{\partial u} \Delta u + \frac{\partial Z}{\partial w} \Delta w + \frac{\partial Z}{\partial \delta_E} \Delta \delta_E \\
\Delta M &= \frac{\partial M}{\partial u} \Delta u + \frac{\partial M}{\partial w} \Delta w + \frac{\partial M}{\partial q} \Delta q + \frac{\partial M}{\partial \delta_E} \Delta \delta_E
\end{align*}
\]

(3.24)

As shown above, the result of the linearisation is a set of simple, ordinary linear differential equations with constant coefficients made up of the aerodynamic stability derivatives, mass, and inertia characteristics of the aircraft, which is often called state-space [38]. This system is more conveniently represented in matrix form as

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

(3.25)

where \( x \) is the state vector, \( u \) is the control vector, and matrices \( A \) and \( B \) contain dimensional stability derivatives which account for the force and moment perturbations already divided by the mass or the moment of inertia of the aircraft. The assumptions taken here by Nelson seemed too confident for small aircraft, so Caughey’s criteria [53] was applied to write the complete state-space as follows:

\[
\begin{bmatrix}
\Delta \dot{u} \\
\Delta \dot{w} \\
\Delta \dot{q} \\
\Delta \dot{\theta}
\end{bmatrix} =
\begin{bmatrix}
X_u & X_w & -w_0 & -g \cos \theta_0 \\
Z_u & Z_w & Z_q + u_0 & -g \sin \theta_0 \\
M_u + M_w Z_u & M_w + M_w Z_w & M_q + M_u u_0 & -g M_w \sin \theta_0 \\
0 & 0 & 1 & 0
\end{bmatrix}
\begin{bmatrix}
\Delta u \\
\Delta w \\
\Delta q \\
\Delta \theta
\end{bmatrix}
+ \begin{bmatrix}
0 \\
Z_{\delta_E} \\
M_{\delta_E} + M_u Z_{\delta_E} \\
0
\end{bmatrix}
\begin{bmatrix}
\Delta \delta_E
\end{bmatrix}
\]

(3.26)

which is the final system of ordinary differential equations that was implemented in the flight dynamics MATLAB program. The equations used to compute the value of the stability derivatives from the aerodynamic stability coefficients obtained from the VLM analysis are listed below and they are based on Nelson [38] and Caughey [53]. The speed derivatives \( X_u \) and \( Z_u \) have been specifically derived for the case of a propulsion system with a constant-speed propeller following Caughey, and effects of the Mach

38
number have been neglected:

\[
\begin{align*}
X_u &\equiv \frac{1}{m} \frac{\partial X}{\partial u} = \frac{-(3C_{D_0} + C_{L_0} \tan \theta_0) SQ_{ref}}{mu_0} [s^{-1}], \\
Z_u &\equiv \frac{1}{m} \frac{\partial Z}{\partial u} = \frac{-2C_{L_0} SQ_{ref}}{mu_0} [s^{-1}], \\
Z_{\delta_E} &\equiv \frac{1}{m} \frac{\partial Z}{\partial \delta_E} = \frac{-C_{Zq} SQ_{ref}[m s^{-2}]}{m}, \\
M_u &\equiv \frac{1}{m} \frac{\partial M}{\partial u} = \frac{C_{mu} SQ_{ref} c_{ref}}{u_0 I_{yy}} [m^{-1} s^{-1}], \\
M_{\delta_E} &\equiv \frac{1}{m} \frac{\partial M}{\partial \delta_E} = \frac{C_{msq} SQ_{ref} c_{ref}}{I_{yy}} [s^{-2}], \\
X_w &\equiv \frac{1}{m} \frac{\partial X}{\partial w} = \frac{C_{Xu} SQ_{ref}}{mu_0} [s^{-1}], \\
Z_w &\equiv \frac{1}{m} \frac{\partial Z}{\partial w} = \frac{C_{Zw} SQ_{ref}}{mu_0} [s^{-1}], \\
Z_q &\equiv \frac{1}{m} \frac{\partial Z}{\partial q} = \frac{C_{Zq} SQ_{ref} c_{ref}}{2u_0 I_{yy}} [m s^{-1}], \\
M_q &\equiv \frac{1}{m} \frac{\partial M}{\partial q} = \frac{C_{mq} SQ_{ref} c_{ref}^2}{2u_0 I_{yy}} [s^{-1}], \\
M_{\delta_E} &\equiv \frac{1}{m} \frac{\partial M}{\partial \delta_E} = \frac{C_{mS_{\delta_E}} SQ_{ref} c_{ref}}{I_{yy}} [s^{-2}].
\end{align*}
\]

(3.27)

According to Nelson [38], the homogeneous (open-loop) system can be written in the form

\[ \dot{x} = Ax \]

(3.28)

and the homogeneous part of the solution can be obtained by assuming a solution on the form

\[ x(t) = C_r x_r e^{\lambda_r t} \]

(3.29)

where \( C_r \) are constants and \( \lambda_r \) are the roots of the characteristic equation, also known as eigenvalues of the stability matrix \( A \). These eigenvalues were computed through the MATLAB script by solving the characteristic equation

\[ |\lambda I - A| = 0 \]

(3.30)

which yields the following roots for the given statically unstable configuration of the Gripen test-bed aircraft trimmed at an airspeed of 20 m s\(^{-1}\):

\[
\begin{align*}
\lambda_1 &= -12.18 + i0.00 \\
\lambda_2 &= 3.02 + i0.00 \\
\lambda_3 &= -0.26 + i0.85 \\
\lambda_4 &= -0.26 - i0.85
\end{align*}
\]

(3.31)

Figure 3.22 was extracted from the MATLAB script output and it shows these roots in the complex plane. Notice that instead of obtaining the usual two pairs of conjugate complex roots for conventional configurations with large positive static margin (see Nelson), here two roots are real and one of those is positive. Babister offers in [54] a complete review of the effects or the longitudinal static margin in the characteristic equation roots: the positive real root obtained here corresponds to the divergence that is expected from aircraft with small to moderate negative static margin. If the static margin was zero, the characteristic equation would be expected to have one zero root and three real negative roots. On the contrary, if the static margin decreased still
3 Theoretical Models

Further, two of the three negative roots would coalesce giving a new damped oscillation, according to this author. The results obtained here verified that, as expected, a closed-loop system (controller) needed to be added in order to provide apparent static stability, moving the unstable root into the stable region of the complex plane, i.e. left side of the imaginary axis.

Figure 3.22: Representation of the longitudinal characteristic equation (open-loop) roots in the complex plane for the Gripen test-bed aircraft in a statically unstable configuration and \( V = 20 \, m \, s^{-1} \)

3.4.4 Assembly of the Complete Flight Dynamics Model

All sub-models were integrated around the main script written in MATLAB. This script gathered the non-preloaded data from the external tools automatically, it scheduled and performed the computations, and it displayed the results in both numeric and graphic format. Although the program computed the linearised flight dynamics system, it also included written functions able to solve non-linear motion and an in-built example with a simple pitch rate controller. However, it was decided to establish a link between the main script and MATLAB Simulink simulation environment and to model the complete control system structure in this application. Despite the fact that only linear aerodynamic data were available, the Simulink model included the complete set of non-linear longitudinal motion equations shown previously in Table 3.4, which were introduced similarly to the three-degrees-of-freedom model available in the application’s aerospace blockset library, see Figure 3.23. This was done with the purpose of leaving the model ready for a straightforward implementation of a non-linear aerodynamics database in future.

An overview of the complete flight dynamics model together with the rest of the control system components can be seen in Figure 3.24. The linear aerodynamics subsystem contained the respective stability derivatives multiplied by the difference between the actual and the reference flight condition, as in Equation 3.24. The servo-actuator sub-system was modelled according to section 3.3.1. Furthermore, the correct pitch controller structure was loaded depending on the controller selected in the main script.
3.5 Study of the Aircraft Dynamics

Figure 3.23: Longitudinal motion equations of Table 3.4 modelled in MATLAB Simulink environment, modified from the three-degrees-of-freedom model available in the application’s aerospace blockset library.

The controllers were modelled according to the theoretical structures proposed in section 3.3, as shown in Figure 3.25 for the APM-Plane FBW A/B pitch angle controller. Results of the simulation were sent again to the main script for analysis and visualisation.

Figure 3.24: Overview of the complete flight dynamics model including an automatic pitch control system, modelled in MATLAB Simulink environment.

3.5 Study of the Aircraft Dynamics

The flight dynamics program written in MATLAB integrated all the models discussed before, and it was especially tuned to represent two of the test-bed aircraft: the delta-canard platform shown in the previous examples, and the traditional Cub model shown in Figure 3.26. The program was used to investigate the response of the aircraft with different pitch control systems under flight conditions similar to those found in the experimental flight testing. For example, Figure 3.27 shows a trajectory plot of a pitch-step response testing with the simulated Cub model, very similar to the real
3 Theoretical Models

Figure 3.25: Pitch angle control structure of the APM-Plane firmware modelled in MATLAB Simulink environment, as seen in section 3.3.4.

flight testing manoeuvres that will be described later in chapter 5.

Figure 3.26: Cub test-bed aircraft modelled in the flight dynamics MATLAB program.

Tests carried out with the simulator program were also useful for evaluating the influence of the diverse components on the aircraft response. One of the critical parameters was found to be the response-speed of the elevator servo, as shown in the example of Figure 3.28. Regarding the evaluation of the controller gains, the PID gain tuning tool provided with the MATLAB Simulink software was used to linearise the entire system and to explore the optimum gain values for each controller according to the flight conditions. Although it was possible to obtain a qualitative assessment of the effect of each gain in the overall response, no clear quantitative correlation with the flight-test data could be established, as will be described in section 5.7.
3.5 Study of the Aircraft Dynamics

**Figure 3.27:** Trajectory plot from the flight dynamics program showing the simulated Cub model going through pitch step response testing.

(a) Fast elevator servo

(b) Servo time-constant increased ten times

**Figure 3.28:** Difference in response to a pitch disturbance for a fast and a slow elevator servo with the same $-10$ percent unstable model, at $V = 30 \text{ m s}^{-1}$. 

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3 Theoretical Models
Flight Control System Integration

The recent boom of the unmanned aircraft market has led to the appearance of numerous electronic flight control systems (FCS) designed for small-scale vehicles and even hobbyist-type model aircraft. The complexity of these systems ranges from basic closed-loop gyroscope controllers to very advanced systems able to compute complex navigation algorithms and process data collected by different sensors such as inertial measurement units (IMU), GPS receivers, pressure transducers, and magnetometers, among others. Some semi-professional systems are also available to (or developed by) enthusiasts, and in some cases they offer a surprising performance comparable to those used in the industry and scientific research but at a fraction of the cost, see for example [51] and [55]. Therefore, the purpose of this study was not to develop a new FCS from scratch, but rather to take advantage of the available technology and to examine the performance of different commercial off-the-shelf (COTS) low-cost systems in statically unstable aircraft.

4.1 Basic Stability Augmentation System

A basic pitch stability augmentation system (SAS) of the types presented in 3.3.2 and 3.3.3 could be achieved with relatively simple equipment: angular rate closed-loop controllers based on single-axis gyroscope sensors are very common in the R/C market since they are extensively used to stabilize the directional axis (tail rotor pitch) of R/C helicopters. Vibrating structure gyroscopes (or Coriolis vibratory gyroscopes) developed for R/C applications are extremely small and inexpensive regardless the transducer manufacturing technology, which is often divided in piezoelectric (crystal) and silicon micro-electro-mechanical system (MEMS). In addition, they are encapsulated in a perfect format for this application and output a signal directly compatible with the servo.

An inexpensive, single-axis, Assan GA-250 MEMS gyroscope [48] was used here for the SAS. This gyroscope, shown in Figure 4.1 has two different operating modes that can be switched remotely at any time: a rate control mode used for the SAS of the type 3.3.2, and an angular vector control system (AVCS) mode (also known in R/C helicopters as “heading hold”) that was used for the SAS of the type 3.3.3. The in-built controller firmware also allowed remote adjustment of one parameter: while in the rate mode it corresponded to the main proportional gain, in the AVCS mode the exact correlation between this parameter and the P and D gains was unknown. The gyroscope sensor was installed on anti-vibration pads inside the fuselage in the Balsa
Figure 4.1: Single-axis Assan GA-250 MEMS gyroscope [48], commonly used in R/C model helicopters. and Cub models (see 5.1), as close as possible to the neutral point, as depicted in Figure 4.2. In all cases it was oriented parallel to the symmetry plane of the aircraft in order to measure rotation about the y-axis.

Figure 4.2: MEMS gyroscope sensor installed in the Cub model near the neutral point, and the corresponding elevator servo location.

### 4.2 Advanced Flight Control System

A much more capable COTS flight controller was used for configuring an advanced FCS. After a thorough market research focused on open-source projects, two platforms based on 32 bit controllers stood out: Paparazzi firmware combined with Apogee v1.00 hardware [55], and PX4/APM firmware combined with Pixhawk hardware [56]. The latter solution was finally chosen and a set of hardware parts were acquired from the official manufacturer, 3D Robotics (United States). The Pixhawk flight controller was the latest release of the open-hardware PX4 autopilot project started by Lorenz Meier and others at the Computer Vision and Geometry Lab in ETH Zürich, supported by the Linux Foundation Dronecode community and the private company 3D Robotics, see reference [56]. This advanced platform was designed around a powerful board equipped with 32 bit processors and a complete suite of sensors. It run an embedded NuttX real-time operating system (RTOS) and a specifically developed PX4 middleware layer. On top, the user was free to choose the flight control software between
the PX4 autopilot from the same authors, and the well-known APM multiplatform autopilot. The fixed-wing version of the APM software, known as APM-Plane [51], was chosen here. All detailed characteristics and technical documents are openly available from the cited references and only relevant features will be discussed throughout this chapter.

The main controller board was designed to be complemented with additional peripheral devices such as a GPS receiver or an external magnetic compass in order to take advantage of the autonomous navigation capabilities. Although autonomous navigation is not concerned in this study, these additional devices were installed and used for detailed flight-path analysis. Figure 4.3 shows the layout of the main components and instruments of the flight control system in the Gripen and Cub test-bed aircraft. Achieving a good distribution that optimises performance is far from simple and it requires some experience and detailed knowledge of the components: the solution is a compromise between the ideal installation requirements of certain sensors, the electromagnetic disturbances between devices, the reduction of vibrations, the length of critical cables, and the tight constraints of available space and weight balance.

Figure 4.3: Layout of the main components and instruments of the flight control system in the Gripen and Cub test-bed aircraft.
4 Flight Control System Integration

4.3 Instrumentation

The principal reasons for selecting the Pixhawk hardware were its complete suite of built-in sensors and its excellent connectivity options for external devices. Figure 4.4 shows a conceptual diagram with some of the main inputs and outputs of the controller board. Only the built-in sensors and instruments that are relevant to this study will be commented briefly since no modifications were performed, while more attention will be given to the specifically developed angle-of-attack sensor.

![Figure 4.4: Conceptual diagram showing the main I/O connections of the FCS and their respective refresh rates. Modified from an original diagram by the PX4 developers community [56].](image)

4.3.1 Embedded Sensors

Inertial Measurement Unit

The core of the controller board comprised two different inertial measurement units with different architecture, different manufacturer, and operating at different frequencies: a STM L3GD20 three-axis gyroscope [57] together with a STM LSM303D three-axis accelerometer and magnetometer [58] on one side, and a Invensense MPU 6000 three-axis accelerometer and gyroscope [59] on the other. Data were blended by the firmware which decided the usage ratio depending on the bias levels detected on each IMU. This was an excellent feature that is not often found in low-cost systems, and it is extremely useful not only for increasing robustness and redundancy, but also for frequency analysis and error suppression in highly vibrating environments like a model aircraft. Apart from a built-in temperature compensation, the drift and deviations of the IMU were continuously monitored and corrected by the firmware using data from other sensors through an extended Kalman filter (EKF). No modification or further development were performed to the IMU system.
4.3 Instrumentation

Barometer

The main board included an embedded barometric pressure sensor, Measurement Specialties MS5611 [60] with temperature compensation, digital output (24 bit), and an altitude resolution of 10 cm according to the manufacturer. It also worked as a digital thermometer with the same digital output, which was used as a good indicator of the main PCB temperature. Altitude data from this sensor were also blended with the readings of analogous instruments through an EKF algorithm. No modifications were performed to this sensor or its firmware other than offset calibrations at the beginning of each flight.

GPS Receiver

Even though the GPS receiver itself was not an embedded instrument, it was decided to include it here because of its strong relation with the main board and its sensors. Although the FCS firmware does not require a GPS device compulsorily, it was designed assuming that these data were available and many functions depended on it. Here, there was no interest in the autonomous navigation capabilities, but since the GPS data were essential for the full EKF functionality and flight trajectory analysis, it was decided to implement it. The COTS solution proposed by the manufacturer of the FCS, 3D Robotics, was one of the best low-cost options available and it was the selected device. It consisted of an external PCB with a ceramic patch antenna, a u-blox NEO-7 GPS/GNSS module [61], a dedicated processor, and a backup battery. It also integrated a Honeywell HMC5883L three-axis digital magnetometer [62] in order to supersede the magnetometer of the main board, more contaminated by electromagnetic interferences. The total weight of the device was 17 g and it provided with a GPS position refresh rate of 5 Hz. No modifications were done in the device or its firmware.

4.3.2 Airspeed Sensor

The airspeed was measured with a conventional pitot-static system, i.e., a probe with static and total pressure orifices. The difference between these is measured with pressure transducers and used to compute the indicated airspeed (IAS) by using Bernoulli’s equation, see Nelson [38].

Layout

Aiming for simplicity and direct compatibility with the FCS, it was decided to purchase directly a low-cost COTS airspeed system manufactured by 3D Robotics. This kit included a metallic probe with the corresponding orifices, silicon tubes, and a separate PCB with two pressure sensors, as shown in Figure 4.5. The pressure sensors were Measurement Specialties 4525DO ceramic-based pressure transducers with digital output (14 bit) and temperature compensation (11 bit temperature transducer). According to the manufacturer, these have an average accuracy of ±0.25% over the operating pressure from 6.9 to 690 KPa, providing with a total airspeed measurement
of roughly up to 100 $ms^{-1}$. It is also relevant to mention that the probe had four radial static pressure orifices which were aligned up-down-left-right in order to minimise the error in the static measurement due to asymmetric pressure at non-neutral flow inclinations, a problem pointed out by various authors in [38] and [63].

![Digital airspeed sensor kit](image)

**Figure 4.5**: Digital airspeed sensor kit as supplied by the manufacturer, 3D Robotics.

### Installation

Since there was no suitable wind-tunnel facilities for investigating the probe position errors, additional care was put to perform an appropriate installation on clean flow locations. As described by Moes and Whitmore in [63], airdata probes are typically mounted on a noseboom ahead of the fuselage nose or on the outer section of the wing in order to minimise the flow disturbances produced by the aircraft. The first option provides the best installation and minimum error according to the results of Gracey in [64]. Hence, the noseboom location was chosen for the Gripen aircraft, but the tractor propeller configuration of the Cub imposed a wing installation. Although lift-induced wingtip disturbances were avoided (the probe was located at about two-thirds of the span) the proximity of the airfoil was expected to cause more error than that of the noseboom installation [64]. As shown in Figure 4.6, flexible pneumatic tubes were routed from the static and total pressure orifices to the pressure transducers, which were located inside the wing approximately 20 $cm$ from the wing-mounted pitot tube in the Cub, and inside the nose approximately 25 $cm$ from the noseboom-mounted pitot tube in the Gripen model. The embedded installation of the transducers in both cases was intended to insulate them from sudden temperature variations and to protect them in case of crash. Data transfer to the FCS was done through a cable connection using $I^2C$ bus protocol.

### Calibration

Without going into source code modifications, the FCS firmware allowed control of the calibrated airspeed (CAS) computation by two different parameters: an initial pressure offset and a dimensionless ratio. The former accounted for the static atmospheric pressure at the airfield and it could be either entered manually or refreshed automatically during the FCS initialisation process by placing a loose fitting cover over the pitot-static probe in order to avoid wind interference. The ratio accounted for the probe characteristics and the installation errors. It was attempted to determine this parameter by testing the instrument in a very small open-section wind tunnel and comparing the readings with a hand-held digital anemometer, but later field tests
showed that this technique accumulated significant errors. On the other hand, the FCS firmware presented a function for automatic in-flight re-calibration based on the measured ground speed and the wind estimates obtained by the EKF. This approach was tested with acceptable results, but nevertheless, in the end it was decided to compute manually this correction ratio by analysing the logged data after circular-pattern flight at high altitude and low, near-constant wind, as depicted in Figure 4.7. Compressibility effects were neglected due to the low flight speeds. Although it can also be argued that altitude variations during flight were small and air density could be considered constant, the FCS firmware already had an algorithm to estimate the true airspeed (TAS) from the calibrated airspeed (CAS) based on the altitude readings, and consequently no modifications were done. In the end, the good calibration achieved could be corroborated by the excellent agreement between ground speed and airspeed for flights carried out in total absence of wind. Data shown in Figure 4.8 correspond to an approach and landing in such zero-wind conditions.

**Figure 4.6:** Fixed pitot-static system installation on the wing of the Cub test-bed model.

**Figure 4.7:** Extract from a manual calibration of the airspeed sensor parameters by analysing the averaged difference between corrected ground speed and measured airspeed, after flying circular patterns at high altitude and nearly constant wind.
Figure 4.8: Airspeed and ground speed measured during an approach and landing in total absence of wind. The good agreement indicates a correct calibration of the airspeed sensor. Notice that the noisy values after landing are normal for near-zero airspeed in this kind of sensor.

4.3.3 Angle-of-Attack Sensor

In order to avoid complex and expensive sensors, it was decided to measure the angle-of-attack with a conventional vane-type flow-direction sensor. This is basically a mass-balanced, pivoted wind vane free to align itself with the direction of the local airflow, which is a robust and reliable method extensively used in flight testing as described thoroughly by Gracey in [64]. In agreement with this author, flow-direction vanes can be directly attached to the body of the aircraft if the precise local flow conditions are known and the instrument can be calibrated accordingly, but however, for flight testing they are usually more effective when mounted on a boom support ahead of the aircraft body, method which was also chosen in this study.

Design

Provided that COTS flow-direction measurement systems are rarely suitable to very small models, it was decided to design and manufacture the instrument following the work of Lundström in [4] and trying to minimise weight and dimensions. Here, a single flow-direction vane was mounted on a boom support and included a pitot-static tube to complete standalone airdata system that could be interchangeable between models. Similarly to Lundström’s design, the angle measurement was done by a magnetic-induction rotary encoder which was extracted together with it’s power-supply regulator from an inexpensive hobbyist-type R/C servo HK28013DMG. The angular position was then computed by measuring the linear analog output of this encoder through an ADC processor in the FCS. Furthermore, the minimum diameter of the boom support was determined by the dimensions of the magnetic encoder and its PCB, resulting in a somewhat bulky layout. The vane was mounted on a traverse shaft approximately two boom diameters from the boom axis in order to minimise the effects of the boom flow distortion, as seen in the investigations carried out by Gracey in [64]. The entire airdata system was designed using CAD as shown in Figure 4.9, which was a useful way of testing the design and verifying the inertia and mass-balance of the vane. The main components were 3D-printed in ABS plastic, while the boom consisted of a 8.8x8 mm carbon fibre tube. The main advantages of the 3D-
printing technique are not only the low manufacturing time and custom-made design, but also the inexpensive and fast replacement of parts in case of damage.

Figure 4.9: CAD model of the airdata probe, combining a pitot-static tube and a flow-direction transducer. The sensor housing, the vane arm, and the vane itself were 3D-printed in ABS plastic.

Installation

Similarly to airspeed instruments, flow direction sensors are best mounted on nosebooms in order to minimise the flow disturbance of the aircraft as Moes and Whithmore describe in [63]. However, according to these authors noseboom-mounted systems can have a significant effect on the forebody aerodynamics at high AOAs and thus other locations are sometimes preferred. Flow angle errors are larger for wingtip installations because of the lifting characteristic of the wing (upwash, sideward and vortex) according to Gracey [64], and additionally wing bending and torsion modes may introduce further noise in the measurements [63]. Nevertheless, the tractor propeller configuration of the Cub motivated a mid-span wing installation, while the noseboom mounting was again chosen for the Gripen aircraft, as seen in Figure 4.3. Data transfer to the FCS was done through a cable connection using \( I^2C \) bus protocol.

Calibration

Estimating exact calibration parameters for this instrument can be extremely complex since it is not only affected by design and installation errors, but also by the dynamic manoeuvres of the aircraft according to Gracey [64]. As pointed out by Richardson in [65], a wind tunnel study is essential to determine the error caused by the vane design and the flow distortion in the vicinity of the instrument, but unfortunately this was not possible here: the correct function of the instrument was tested in a very small open-section wind tunnel, see Figure 4.10, but the facility was not appropriate for accurate measurements with the instrument installed in the aircraft. Nevertheless,
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it was enough to verify that the same readings were obtained in upright and inverted positions, which is a good indicator of low manufacturing imperfections according to the same author. Due to the relatively high stiffness of the components at this small scale, deflection and bending effects of the boom and the fuselage were neglected.

Figure 4.10: Basic functionality test of the airdata probe using a small open-section wind tunnel.

Moreover, Gracey describes in [64] several in-flight calibration methods but most of them are not suitable for small model aircraft, and only three of them could be used here: the attitude—angle method tries to compare the measured angle-of-attack with the total pitch angle (attitude) of the aircraft in level unaccelerated flight and in the absence of vertical air currents, situation in which both angles should be equal. However, it was nearly impossible to maintain this flight condition without a very precise variometer and flying in the turbulent conditions that are found close to the ground, as shown in Figure 4.11. The second, known as attitude—flight-path angle method, was also used during post-flight analysis. The angle-of-attack was estimated from the attitude, the flight-path and the airspeed through the mathematical relations explained in section 3.4.1. Still, this method was only valid for wings-level flight and it required an accurate measurement of the vertical speed, which here was again difficult to obtain due to turbulent conditions, rapid changes in trajectory, and the correspondent measurement fluctuations. Figure 4.11 also includes this estimation, and it is easy to see strong deviations from the measured AOA (graphed here after off-line low-pass filtering).

In the third method, an onboard camera was used to capture the motion of the vane with respect to the horizon and compare it to the measured values, as shown in Figure 4.12. Nevertheless, high fidelity observations were impractical due to the rapid fluctuations of the vane and the difficulties of maintaining a steady attitude in the aircraft. Thus, only a general assessment of the instrument behaviour was obtained. As a consequence, correct calibration of the instrument could not be guaranteed by any of these methods, even though the observations made with the camera suggested that the error should not be excessive and the measured data seemed to agree well with the expected theoretical values, such as the near-zero AOA during the high-speed dive reproduced in Figure 4.13.
4.3 Instrumentation

Figure 4.11: Extract from a test flight showing the difficulties of maintaining level unaccelerated flight. The pitch angle is compared to the AOA measured with the air-data probe, and to the AOA estimated indirectly from attitude, trajectory and airspeed values. Notice that the measured AOA is shown here after off-line low-pass filtering.

Figure 4.12: Evaluation of the AOA transducer performance by comparing data with visual references during a flight test.

4.3.4 Flight Data Recording

Telemetry

The FCS onboard the aircraft was continuously connected to the ground station by a bi-directional radio link. A COTS wireless transmission system in the 433 MHz band from the same FCS manufacturer, 3D Robotics, was used here due to its acceptable performance and low cost. This bi-directional connection was extremely useful for the flight tests and it allowed not only commanding actions and changing FCS parameters during flight, but also downloading flight data and visualising the state of the aircraft in real-time. Although flight data are reported through the wireless telemetry at a lower sample rate than that recorded internally by the FCS, it can be a useful backup in cases where the aircraft has been lost and no other flight data are available for analysis.
4 Flight Control System Integration

Figure 4.13: Extract from a flight test showing a high-speed dive, where the measured AOA agreed well with the near-zero AOA expected. This may indicate a tolerable offset calibration. The estimated AOA seemed to suffer serious deviations.

Data Logger

The Pixhawk board has a built-in flash card data logger that has been essential for recording the data from the sensors and evaluating the flight tests. Even though all telemetry data received at the ground station were recorded, the sample rates were significantly lower and varied according to the available transmission bandwidth. On the contrary, the onboard data logger provided with higher and much more constant sample rates. These were firmware-dependant but tightly constrained by the hardware limitations, far away from the internal algorithm refresh rate. However, there was enough computing power to record all parameters available in the firmware at a sample rate that ranged from 5 Hz for the GPS receiver to 50 Hz for the IMU data and some external sensors, as seen in Figure 4.4.

Onboard Camera

As seen earlier in Figure 4.12, a micro-camera was mounted onboard the test-bed aircraft in certain occasions in order to obtain direct visual references of the performance of instruments or control surfaces. No wireless video transmission or streaming devices were used. The images were captured in an internal storage, and only used for post-flight analysis.

4.4 Flight Control Laws

The APM-Plane flight control software allowed remote switching between the different FCL modes. These were mutually exclusive (only one could be active at a time) and the transition was instantaneous, without blending progressively the eventually different commands. The sequence and selection procedure of the FCL modes were
freely configurable. Figure 4.14 shows the configuration used during the flight testing: the pilot could override any active FCL and regain manual control by flicking only one switch of the radio transmitter. The augmented FCL modes as well as alternative functions required two switches to engage. The CAS modes denominated FBW A/B and FBW C corresponded to the FCL modelled in 3.3.4 and 3.3.5 respectively.

Figure 4.14: Diagram of the different FCL modes as configured in the FCS during the flight tests. Modes were mutually exclusive and the transition between them was commanded by two switches in the radio transmitter.
4 Flight Control System Integration
The experimental validation of the theoretical estimations and the investigation of the relaxed static stability effects through flight testing comprise an important part of this work. Although from the theoretical point of view there is no big issue in handling instability with a properly designed controller, this needs to be empirically assessed by including the real (imperfect) components and sensors into the loop, and by subjecting the system to real conditions with the complex unsteady aerodynamics that dominate small scale flight in a usually turbulent atmospheric layer near the ground. This chapter covers the practical work carried out, from the manufacturing of the test-bed aircraft to the flight test verification. Furthermore, a brief assessment of the relaxed stability limits for manual is also included in the last section.

5.1 Test-Bed Aircraft

The experimental evaluation was carried out with simple and light remotely piloted aircraft with electric propulsion. The main requirements for the test-bed platforms were low-cost, to be highly modifiable and easy to repair, to be able to carry extra payload, and yet small enough to be transported to the airfield by one person without a car. At least one model had to present a conventional configuration while other had to be a tailless delta design, in order to explore its possibly different flight dynamics. Two models fitting to some extent these characteristics were already available at the division: an old conventional balsa wood aeroplane from a previous fuel-cell project (referred to as Balsa model in this text), and a very elementary electric ducted-fan jet in delta-canard configuration which resembled a Dassault Rafale aircraft. Another two more suitable models were built: a Multiplex FunCub model aircraft kit similar to a Piper J-3 Cub, and a delta-canard pusher-propeller fighter which resembled a Saab JAS 39 Gripen aircraft. All these four models are shown in Figure 5.1 and will be briefly commented one by one. Detailed characteristics can be found in appendix A.

(a) Cub

This model, referred to as Cub throughout this paper, was built from a COTS kit FunCub of the German R/C manufacturer Multiplex. It was similar to a Piper J-3 Cub aircraft, although it was not geometrically or dynamically scaled. Despite its reduced dimensions and its foam structure, it was a very durable and capable test-bed able to carry almost the double of its nominal take-off weight in extra payload. Several modifications were done during the construction in order to adapt the aircraft to the
study needs, to re-use the available components, and to improve durability and flexi-
bility of the airframe towards other future projects. Among these modifications, were
the manufacturing of a new brushless-motor mount, a new nose structure, fuselage
wood reinforcements, a modular equipment tray, a new battery compartment, new
cockpit lock mechanism, and new control surface servo-actuators placement far back
in the tail, see Figure 5.2. The default landing gear wheels where also substituted with
smaller ones that were less likely to affect significantly the general aerodynamics.

(b) Balsa

Following the low-cost philosophy and with the idea of re-using existing materials as
far as possible, an old conventional balsa wood aeroplane from a previous fuel-cell
project was repaired and modified in order to use it for the first “risky” tests before
the more appropriate test-bed aircraft were available. The work done on this model
included fuselage and landing gear repair and reinforcement, control surfaces restora-
tion, nose modification and reinforcement, battery tray installation, and wing root
repair, among others.

(c) Rafale

A very simplified model of a Dassault Rafale aircraft equipped with an electric ducted-
fan engine was also available and almost ready to fly as tailless delta test-bed until a
more advanced, multi-control-surface model was built. Despite its small size, a very
5.2 Flight Test Methodology

Despite the reader may think that the design of the tests and procedures for subscale remotely piloted aircraft are substantially different from those used in full-scale flight testing, some of the commonly used techniques and procedures are also suitable here.
5 Experimental Evaluation

Figure 5.3: Some details of the building process of the Gripen model.

A complete introduction to flight test engineering can be found in the RTO/NATO report referenced in [67], while other references such as [1], [6], and [17] describe very advanced testing techniques for research subscale aircraft. Considering the scope, the knowledge, and the resources available during this research, the approach was here considerably more modest, although not necessarily less methodical.

5.2.1 Flight Tests

Due to the small weight of these electric-powered platforms, it was possible to perform the flight tests under the standard model aircraft hobby license and insurance provided by the Swedish Aeromodelling Federation (SMFF). Flights were carried out in the airfield of the Malmens Modellflygklubb in Linköping, and all operations were limited to strict visual line-of-sight rules (VLOS). As commented by Cox et al in [17], the main drawback of remotely-piloted flight testing under VLOS is that the range limitation causes the test execution to be highly compressed: the need for constant turns reduce the “on-condition flight” to only 10-20 seconds in each straight leg, and consequently all tests had to be planned in such way that they can be split into multiple segments. Experience shows that, ideally, this kind of flight test should be monitored
5.2 Flight Test Methodology

by minimum two people apart from the pilot: while the latter focuses only in flying the required manoeuvres as accurately as possible, a second person should verify the flight data and the status of the systems in real time, and a third person should take the responsibility to manage the sequence of tests and indicate the pilot when to proceed to the next manoeuvre. Here, all these tasks were carried out by the pilot, a situation which required a great deal of previous planning, methodical procedures, and still some extra time to repeat test flights after non-successful manoeuvres.

A digital flight logbook in Microsoft Excel was used to keep track of each and every one of the test flights. The annotations in this logbook included the configuration of the aircraft, the duration of the flight, the atmospheric conditions, and any additional comments and observations. The most significant atmospheric parameter was of course the wind speed and gustiness: although a hand-held digital anemometer was used in several occasions, wind speed was evaluated using the Beaufort Scale as published by the Royal Meteorological Society in [68], with additional notes regarding surface gustiness. Regarding the aircraft configuration, accurate measurements of the CG can be difficult in the airfield. In order to save time, practical visual marks and stickers were used during the tests and the precise CG values were later measured in the laboratory by using a precise balance device as shown in Figure 5.4.

In the aircraft equipped with the advanced FCS, numerical measurements were recorded by the data logger and available for post analysis. However, when the aircraft were non-instrumented or the test was based on pilot rating, the corresponding comments were written in the logbook right after landing. When possible, tests involving pilot rating were performed during the same session in order to keep the criteria as similar as possible. The Cooper-Harper rating scale, available in various references such as Nelson [38], was used here to evaluate the flying qualities of the aircraft. This scale will be referred to as CHR throughout this text.

![Figure 5.4: Measurement of the CG longitudinal location by using a precision balancer device, in the image mounted on the Cub test-bed aircraft.](image)
5 Experimental Evaluation

5.2.2 Post-flight Analysis

After a session of flight tests all data logged in the FCS were downloaded to a computer and classified together with the ground station recordings and the respective flight logbook notes. As commented before, the accurate values of the aircraft CG were checked with the balance device and corrected if necessary. The Python algorithms included with the open-source ground station software Mission Planner, by Michael Oborne [69], were used to create KML and MATLAB MAT-files from the binary files recorded by the data logger. The resulting KML files, compatible with any three-dimensional Earth browsers such as Google Earth, were suitable for a quick analysis of the flight trajectory and they contained not only the flight path but also the attitude of the aircraft at discrete points and information about the active control mode. On the other hand, the MAT-files contained all variables and system parameters as recorded by the data logger. A MATLAB script denominated FLOGA v1.00 was specifically developed to analyse thoroughly these files. This script, which is delivered together with this thesis, allows visualisation and comparison of parameters, detection of errors, sampling frequency analysis, and filter design for certain variables.

5.3 Validation of the Estimated Neutral Point

The first flight tests of each test-bed model were intended to empirically validate the control-fixed neutral point (NP) estimation obtained from the VLM computations. This was done by performing series of flights under direct manual control (no SAS engaged) in which the CG of the aircraft was progressively shifted backwards, from the nominal positive static stability margin until the estimated NP location, or even beyond this point if the aircraft longitudinal stability was not seriously degraded. The qualitative assessment was based on the remote pilot “feeling” and the handling qualities of the aircraft, which for a neutrally stable aircraft could be compared to a rating 7-8 in the Cooper-Harper scale (CHR), see Nelson [38]. Therefore, it was not possible to define accurate validation boundaries. In fact, the accuracy of this method is questionable since NASA’s experiences already shown that remotely controlled models do not provide optimum cues for the pilot to effectively rate flying qualities, as stated by Chambers in [1]. A more rigorous method to determine the control-fixed NP from flight data is described by Nelson in [38], which is based on the derivation of the the elevator angle needed to trim the aircraft with respect to the lift coefficient. However, this method requires to measure accurately the elevator angle of trim at various speeds for different CG locations and therefore could not be applied here due to the COTS servo-actuators without position-signal output that were used in the test-bed platforms.

For the Balsa model, it was surprisingly clear that the estimated NP location was accurate: the aircraft handling qualities degraded rapidly when decreasing the static stability margin below 5 percent of the MAC (8 \text{ mm}), and lack of natural stability was evident when tested at zero static margin. While at this point the aircraft was still flyable with manual control, any perturbation caused by wind gusts or control inputs was clearly not damped and pilot intervention was required to restore the attitude.
5.4 Evaluation of the Gyroscope-Based Flight Control System

Attempts to pass this point to a negative static margin (−5 percent) resulted in the aircraft feeling certainly unstable, suffering constant oscillations and being very difficult to fly straight and level without any SAS.

In the case of the Cub, this validation did not result in obvious: the aircraft behaviour and trim changed progressively when approaching zero static margin but its flying qualities did not degrade sharply. With the CG located near the NP, the aircraft became definitely more responsive and less predictable, although still flyable in direct manual control mode. It may be assumed that an additional stabilising effect provided by the low-lying centre of gravity (with respect to the high wing) together with a more “forgiving” airfoil characteristics could contribute to this phenomenon. Thus, although it was confirmed that the NP lay certainly close to the estimated location, it could not be verified with more accuracy by using this technique.

A similar behaviour was observed in the Rafale electric jet and no accurate validation was achieved: longitudinal trim variation and increased response to control inputs were noticed as the static margin was reduced towards the estimated NP, but no sharp degradation of the flying qualities occurred. In fact, the control sensitivity increased significantly more in the roll axis, while the aircraft remained relatively controllable in pitch even for slightly negative static margin values past the estimated NP. The aircraft presented limited pitch attitude divergence and it had enough control authority to perform correct recovery even for severe nose-up disturbances, aspects which contributed to a general feeling of controllability (7-8 CHR). Taking into account the low inertia of this model, it may be assumed that the aerodynamic characteristics of this delta-canard configuration played the main role in the gradual deterioration of the longitudinal stability and control. Special aerodynamic features of the delta wing can be consulted in Bertin and Cummings [21], and some of them such as the low lift curve slope, the strong vortex-lift and transient interactions might be behind the observed phenomenon. These effects will be soon investigated further with the Gripen model.

5.4 Evaluation of the Gyroscope-Based FCS

The experimental evaluation of the basic inertial FCS approach with the single-axis gyroscope was considered first step in the investigation of relaxed stability. As explained in section 4.1, taking into account the original application of these gyroscope controllers it seemed clear from the beginning that stabilising the rapid dynamics of small aircraft models was not supposed to be a big problem. There was however no previous experience with statically unstable aircraft, and it was still not clear how the single-axis gyroscope, if successful, could affect the control of the vehicle during different phases of the flight. The MEMS gyroscope GA-250 [48] was installed in the Balsa, Cub, and Rafale models. Due to the possibility to switch remotely between different flight control laws (FCL), both the pitch rate FCL modelled in 3.3.2 and the pitch attitude FCL (mode AVCS of the gyroscope) modelled in 3.3.3 were tested simultaneously. Since these flights were performed without any data logger onboard,
results are summarised here in the form of pilot comments and rating of the flying qualities.

The Balsa model was pushed to its limits in a series of test flights where the CG was progressively moved backwards, evaluating the performance of both FCL modes while varying the only adjustable gyroscope gain at different airspeeds. The SAS began to be indispensable at a negative static margin of approximately $-5\%$ of the MAC, where the aircraft felt highly unpredictable (rating 8-9 CHR) and entered easily pilot-induced oscillations (PIO). Both rate and AVCS-attitude FCL modes performed effectively and restored good handling qualities (rating 1-3 CHR). At each step, the available gain of both FCL modes was progressively incremented since no oscillations were observed from the ground, even at high speeds. At a negative static margin of $-13\%$ of the MAC, the aircraft was totally unflyable under manual control at any time (rating 10 CHR), and the use of the SAS was imperative from take-off to landing. The performance of both FCL modes was such that basic aerobatic manoeuvres could be performed without problems. However, the AVCS-attitude mode FCL (3.3.3) began to show some advantages over the rate mode FCL (3.3.2), such as the automatic regulation of the elevator trim needed to compensate the significantly tail-heavy balance of the aircraft. At this point the effectiveness of the elevator started to be compromised at low airspeeds, and lateral stability also started to feel slightly marginal. The aircraft was finally flown with a negative static margin of $-21\%$ of the MAC, level at which manual control could not be engaged at any time due to instantaneous loss of control. Surprisingly, flight was still possible with good handling qualities when the stabilisation system was active. The AVCS-attitude mode FCL started to offer a significant advantage over the rate mode FCL due to the constant attitude control, especially critical during take-off and landing manoeuvres. The performance of the elevator became critical at slow speeds and it was observed that its stall could lead to a total loss of control. Therefore, approach manoeuvres felt safer when performed at higher airspeed and steeper angle than usual. The aircraft was significantly more responsive to control inputs, but oscillations did not appear at high gain settings, even though wind gusts were often overcorrected for short periods of time. The poor speed of the standard analog servo used for the elevator in this model seemed to be the bottleneck of the control system, and thus it could be responsible for the limited gain effect.

In the Cub model, this issue was addressed by installing a digital high-speed servo, although the performance of the particular servo used in the test resulted disappointing and not far away from that of the analog servo. Nevertheless, the Cub featured a larger and more effective elevator with aerofoil shape and no articulation gap. The procedure was similar to that of the Balsa model, although it was flown only up to a negative static margin of $-11\%$ of the MAC due to the urgent need of the aircraft for the advanced FCS testing. Up to that point, it was verified that both FCL modes performed exceptionally well (rating 1-3 CHR) and the elevator authority did not feel as compromised as before at low speeds. Demanding manoeuvres such as aerobatics or controlled stalls could be performed without any signs of instability. Again, the AVCS-attitude mode FCL felt especially comfortable: no elevator trim was needed, and smooth take-offs or approaches at constant pitch angle were achieved effortlessly.
On the other hand, differently from the Balsa model case, the gyroscope gain presented in both FCL modes an upper boundary where sharp correction of wind gusts started to cause short oscillatory cycles, especially at high speeds. The slightly faster response of the servo and the significantly higher elevator authority could be behind this effect.

Since no sharp degradation of the flying qualities in the Rafale model, it was difficult to define a point were the SAS became necessary. However, both modes performed well and high gains were not needed to achieve good flying qualities (1-3 CHR) even at a negative static margin of $-7$ percent of the MAC. At this point, the seriously weakened lateral stability together with the lack of rudder control in this model became a serious complication for coordinated turns, and this prevented the testing from continuing towards further negative stability settings. It would be necessary to increase the vertical tail area of this model in order to compensate for the degradation of lateral-directional stability.

Nevertheless, in all models the single-axis gyroscope presented performance limitations during turns, in both rate and AVCS-attitude modes: the angular rate about the pitch axis detected by the gyroscope during normal banked turns was considered by the system as a non-desired response, and thus, it was actively compensated. Thanks to the limited authority of the SAS, this effect was not a serious problem during flight, but it required a continuous “pull-up” command from the pilot during turns, especially in the AVCS-attitude mode. This proved the need for more sophisticated FCL with pitch rate washout filters and inertial measurements in various axes.

### 5.5 Evaluation of the Advanced FCS

The advanced FCS has been thoroughly investigated using the Cub platform. Initial HIL ground tests and preliminary flight tests with nominal positive static stability were needed to get familiar with the complex adjustments and functions of the FCL firmware (APM-Plane [51]) and the ground station tools (Windows-based Mission Planner [69]), as well as with the data logger operation. The first flight tests with were focused on adjusting optimum values for the CAS in all pitch, roll and yaw, for the nominal stable configuration (a positive static margin of 8 percent of the MAC). The automatic gain-refinement function included in the FCS firmware, denominated “autotune” mode, was used during these tests as shown in Figure 5.5, although this system led to very conservative values that were later refined manually, especially for the integrator I and derivative D gains.

#### Controller Performance for Relaxed Stability

The same aircraft was flown repeatedly with the same set of PID gains (optimised for nominal positive stability) at various levels of static stability, ranging from a positive static margin of 8 percent of the MAC to a negative 10 percent, where the degradation of the pitch-angle controller performance was such that the integrity of the aircraft was seriously jeopardised. The aircraft response to pitch angle step inputs for three
5 Experimental Evaluation

Figure 5.5: Extract from a flight test showing automatic in-flight refinement of the pitch controller gains using the “autotune” function of the APM-Plane FCS firmware.

different stability levels is shown in Figure 5.6. Although some minor oscillations were experienced when approaching neutral stability, the aircraft remained predictable and could be rated CHR 3-5. However in the last case, the FCS performance was clearly unsatisfactory (CHR 7-8) and strong oscillations of about 1.5 s of period demanded all the attention of the pilot, especially when increasing the airspeed. Notice that these oscillations were convergent at lower speeds (at the top of a climb) but undamped for higher speeds (at the end of a dive). The same behaviour was obtained for both pitch rate (section 3.3.5) and pitch angle (section 3.3.4) FCL modes, which seemed consistent with the fact that their inner loop design is essentially the same.

Figure 5.6: Response to pitch angle step inputs of the Cub aircraft with pitch angle FCL engaged and the same set of PID gains, for: (a) 8 percent of positive static stability margin, (b) neutral static stability, (c) 10 percent of negative static stability.

The observations suggested that the damping of the short-period oscillations was the
critical factor to improve the performance of the pitch controller for relaxed stability settings. In successive flights with the same platform, the derivative (D) gain was progressively increased while maintaining all other parameters constant. The improvement of the performance was immediate, as shown in Figure 5.7. Nevertheless, the very-conservative limit of the maximum derivative gain allowed by the firmware was already reached at \(-10\) percent of static margin. This limit, which is supposed to protect conventional aircraft from suffering from severe oscillating over-corrections according to the developers, needs to be overridden through the code or directly deactivated to continue towards more negative static margin configurations. It is suspected that the mediocre response of the elevator servo played a significant role in the damping performance as well.

During all these tests, the integrator I gain was maintained constant and as low as possible (10 percent of its total range) since it shown appropriate elevator authority to trim the tail-heavy model even for the most unstable case. There was concern about the fact that an excessive integrator authority could lead to eventual loss of control in some situations, such as take-off and landing with a tail-dragger configuration or long period oscillations due to excessive trim sensitivity. However, none of these problems were detected during the flight tests.

Regarding the proportional (P) gain, the observations suggest that the lowest acceptable setting should be used in the beginning, provided that the response of the aircraft increased considerably when the static margin was reduced. However, no further decrease of the gain was required here since, for this aircraft, the increase in the response was somewhat compensated with the increase in the damping gain required to eliminate unstable oscillations. Moreover, it may be assumed that the proportional gain could be decreased slightly in order to alleviate the work of the damping loop if initial overshoots are detected and the rate of response is still satisfactory.

Figure 5.7: Response to pitch angle step inputs of the Cub aircraft with 10 percent negative static stability margin and pitch-angle FCL engaged: (a) PID gains set for nominal positive stability, (b) Gains P and I without change, and derivative gain D increased by a factor of three.
5 Experimental Evaluation

High Angle-Of-Attack Testing

The response of the FCS in high-AOA conditions was also evaluated with the Cub model. The system presented no relevant problems in the pitch-angle and pitch rate FCL even for a large negative static margin, although it could be suggested to let the velocity-scaler parameter increasing the gains a bit more for low airspeeds. On the other hand, stalls and departures presented more problems. Figure 5.8 is an example of a high-AOA low-speed stall with wings level carried out with the Cub model in a negative static margin configuration, and with the pitch-angle FCL engaged: it is possible to see that as the airspeed decreases, the AOA increases up to a point were the main wing starts to stall and the attitude cannot be maintained. The natural reaction of the aircraft is to pitch nose-down regardless of the tail-heavy configuration, which may be caused by the fact that the horizontal tail has not stalled yet (apart from its geometry, it is directly submerged in the prop wash). This desirable response is however counteracted by the FCS which tries to keep the previously commanded pitch attitude by deflecting the elevator and pulling-up, making the situation worse and letting the AOA increase even more. The pilot, which in this case is aware of the situation, is forced to intervene with full throttle and commands a new pitch attitude out of the stall.

![Figure 5.8](image-url)  
**Figure 5.8**: High-AOA low-speed stall and recovery of the Cub model with a negative static margin of $-10\%$, and with the pitch-angle FCL active. Notice that the pilot intervention was needed to ensure correct recovery.

As seen in this example, it would be critical to correct the FCS automatic response during stalls and departures since there is no possibility of a fully-manual recovery of a model with relaxed stability. However, it was seen that overriding manually the FCS commands without disengaging it was enough, provided that the FCS authority
is limited to some extent. It was observed that the recovery of the unstable model was possible and not significantly different from the statically stable configuration as long as the horizontal tail did not stall. Consequently, in the delta-canard configuration the all-moving canard surfaces are expected to have an advantage over the elevons, and therefore it is suggested to give them an important role in the attitude control loop for high-AOA conditions.

### 5.5.1 Angle-Of-Attack in the Attitude Control Loop

Although the implementation of a CAS based on AOA (see section 3.3.6) has not been completed by the time this report is being written, some necessary previous steps have been completed.

#### Signal Processing

The quality of the AOA transducer signal and its suitability as an input for the attitude control loop was carefully analysed. The standalone instrument showed a good performance and a clean analog output which allowed the computation of the AOA with a noise in the order of 0.6 degrees. However, the onboard-camera and the logged data suggested that during flight the flow-direction vane experienced significant aerodynamic buffeting and quick oscillations. These were not significant in the case of a very calm atmosphere and zero-wind conditions as shown in Figure 5.9(a), and no filtering would be needed. On the contrary, in light winds with slight gustiness the "aerodynamic noise" started to be relevant and hindered the accurate measurements needed for the control loop, as shown in 5.9(b).

![Figure 5.9](image-url)

**Figure 5.9:** Measured AOA during a take-off and climb manoeuvre, shown before and after low-pass off-line digital filtering for (a) absolutely calm wind, (b) light wind and gustiness. Notice that the high frequency noise comes from aerodynamic turbulence and not from electromagnetic noise or sensor malfunction.
The noisy transducer signal was therefore studied in order to find the frequencies of interest: Figure 5.10 shows its power spectral density (PSD) using fast Fourier transform. It can be observed that most of the noise is spread equally from 2 to 10 Hz, decreasing gradually towards higher frequencies. Considering the inertial measurements, the frequencies of interest for the test-bed models were between 0 and 2 Hz. Hence, a continuous-time low-pass filter was designed with a corner frequency of 2 Hz. This filter was transformed into a digital (time-discrete) form by using a bilinear transform, also known as Tustin’s method. For the 50 Hz sampling frequency of the AOA transducer, this filter takes the form:

\[
DLP = \frac{0.1482s + 0.1482}{z - 0.7035}
\]  

(5.1)

See Shenoi [70] for extended explanations on signal processing. All data shown in the figures was processed with a zero-phase (off-line) digital filtering, i.e. processed both the forward and reverse directions, but it may be assumed that any digital filter function implemented on-line in the FCS will have some phase-delay.

**Figure 5.10:** Frequency analysis of the AOA transducer signal of Figure 5.9(b), before and after low-pass off-line digital filtering. Power spectral density using fast Fourier transform.

**Transducer Response**

There were initial doubts about both the delay in the AOA transducer response due to the inertia of the flow-direction vane, which could be significant for the fast dynamics of these models. There was also concern about the ability of such instrument to capture appropriately the attitude changes of the model and serve as a reliable input for the control loop. Flight test data suggest that there is definitely a slight delay in the AOA measurement with respect to the pitch rate detected by the IMU. However,
5.5 Evaluation of the Advanced Flight Control System

the AOA transducer seemed to perform generally well and to reflect appropriately the attitude variations even in difficult situations. A good example is given in Figure 5.11, where the pilot tries to control manually the Cub aircraft with a negative static margin of $-10$ percent of the MAC and the FCS disengaged. The slight delay of the AOA measurements with respect to the pitch rate can be clearly observed in the oscillations. Therefore, the importance of the inner rate-feedback loop proposed in section 3.3.6 for the AOA CAS can be demonstrated. Moes and Whithmore already stated in [63] that the degraded performance of the airdata systems during high rate manoeuvres motivates the enhancement of the measurements by using inertial data from the IMU through, for example, a Kalman filtering scheme. This could be a useful future implementation in the existing EKF code of the FCS.

![Figure 5.11](image)

**Figure 5.11:** Severe attitude oscillations occur when trying to manually control the Cub model with a negative static margin of $-10$ percent, with the FCS disengaged. Notice how the measured AOA reflects the variation of the pitch rate with a slight delay.

**Estimated vs Measured Values**

Figures 4.11, 4.13, 5.8, and 5.11 have already shown that the AOA estimated mathematically from the attitude, flight-path, and airspeed parameters, presents large deviations in some circumstances and it is not appropriate for its use in the control loop. However, a higher fidelity in these parameters, together with a more complete three-dimensional mathematical transformation might provide with results good enough to be used as a backup for the AOA transducer.
5.6 Relaxed Stability Limits for Manual Control

It is difficult to establish a boundary between controllable and uncontrollable aircraft based only on its static longitudinal stability. First of all, the limit would depend on how controllability is defined. Indeed, according to Nelson, the definition of acceptable flying qualities is often determined by the intended mission. Here, it was decided to take as a reference the rating 10 CHR, and to consider uncontrollable an aircraft that, even with all the effort of the pilot, departs from the intended flight path long enough to put in danger the integrity of the aircraft during precision manoeuvres such as landing. This definition depends, of course, on other factors like the atmospheric conditions and the inherent characteristics of the airframe. The experiences related here should therefore be take only as examples, and it could not be appropriate to extrapolate a general rule.

The balsa model started to feel highly unpredictable and difficult to fly under manual control at a negative static margin of approximately $-5$ percent of the MAC. At this point, flying qualities could be rated 8-9 CHR, and the response was significantly prone to PIO. Beyond this point, the aircraft was totally unflyable under manual control at any time, and the use of the SAS was imperative from take-off to landing.

![Figure 5.12: Extract from a test flight of the Cub model with neutral static stability, showing the difficulties of maintaining a zero-pitch constant attitude (rating 8-9 CHR). The positive steady-state pitch rate observed towards the end of the figure corresponds to a turning manoeuvre.](image)

Although the flying qualities of the Cub model did not degrade as sharply, at neutral static stability the aircraft became sensibly more responsive and less predictable, with a rating of 7-9 CHR depending on other factors such as wind gusts, airspeed, and maximum surface deflection settings. Figure 5.12 shows an extract of a test flight with the Cub model in a neutrally stable configuration. As can be seen, despite the aircraft is still flyable in direct manual control mode, the pilot struggles to avoid departures and produces much stronger elevator deflections than the automatic system. It is also in-
5.7 Correlation with the Flight Dynamics Simulator

Interesting to notice that at certain points inaccurate corrections couple with the aircraft response leading directly to PIO. A more extreme case with a negative static margin of $-10\%$ and uncontrollable flying qualities (rating 10 CHR) has been already shown earlier in Figure 5.11, where these oscillations are considerably faster.

In the delta-canard Rafale model no sudden degradation of the flying qualities was observed, and therefore no precise limitation could be defined. In fact, the aircraft remained relatively controllable in pitch even for slightly negative static margin values past the estimated NP (7-8 CHR). Although relatively unpredictable, at $-7\%$ of negative static margin the aircraft could still be controlled at slow speeds, but it required constant compensations and it was not possible to follow a precise flight path. The weakened lateral stability added more complications and the aircraft was therefore rated as unflyable (10 CHR). This also warned that longitudinal stability and lateral stability cannot always be decoupled in reality. However, the observations suggest that the delta-canard configuration may have some beneficial characteristics in statically unstable settings, such as a “natural limitation” in extreme departures which makes the aircraft easier to recover. This has not been yet verified with the Gripen model by the time this paper is being written, but it will be further investigated.

5.7 Correlation with the Flight Dynamics Simulator

It was only possible to investigate thoroughly the correlation between flight testing data and the flight dynamics simulator program for one of the two test-bed aircraft equipped with data logger: the traditional Cub model. Interesting manoeuvres performed during flight tests were selected and reproduced in the simulator program, as shown earlier in section 3.5. A good example is given in Figure 5.13, where the simulated aircraft was subjected to pitch angle steps similar to those recorded during flight, with the advanced pitch angle FCL engaged and identical settings: $-10\%$ of negative static stability and inappropriate gains in this case. It is evident from the figure that the simulator presents excessively stable responses and it fails predicting the oscillation characteristics.

However, by analysing more thoroughly small fractions of the flight and visualising all parameters, it soon becomes clear that the simulator is not severely off from the registered responses in most situations. Figure 5.14 shows in detail a pitch angle step input, or doublet, flown with the Cub aircraft in the same configuration, but in this case with more appropriate controller gains. It is possible to see that all parameters of the simulated aircraft behave approximately as those recorded in flight, although the exact magnitudes differ.

In line with this, the flight dynamics simulator was definitely more useful when focusing in short-term motions, and evaluating the qualitative effect of the gains and parameters rather than optimising their values. The somewhat similar responses achieved may indicate that the model worked correctly, but also that more effort should be put into refining the estimation of the aircraft characteristics.
5 Experimental Evaluation

![Diagram](image)

**Figure 5.13:** Response of the Cub aircraft to pitch angle steps, with 10 percent negative static stability margin, pitch-angle FCL engaged, and inappropriate gains, for: (a) real data recorded in flight testing, and (b) simulated flight using the flight dynamics program.

![Diagram](image)

**Figure 5.14:** Detailed response of the Cub aircraft to a pitch doublet input, with 10 percent negative static stability margin and pitch-angle FCL engaged, for: (a) real data recorded in flight testing, and (b) simulated flight using the flight dynamics program.
6

Discussion and Conclusions

In spite of the fact that the results have been already commented throughout the respective sections, some general concluding remarks will be discussed here.

6.1 Systems Modelling

A simple flight dynamics simulator program and several complementary analysis tools have been developed, evaluated, and are delivered together with this thesis. Although the poor correlation with the experimental results only allows to use it for qualitative analysis, it may be assumed that its fidelity can improve significantly by enhancing the estimation methods of the aircraft characteristics. The simulator program is open for further development, and it will support the replacement of several modules by more advanced analysis tools.

Moreover, the models could be adjusted by system identification. Although a wind tunnel would be the best tool, system identification by free-flight testing the test-bed aircraft may be possible even with the basic low-cost systems that have been used in this study. It would be however a challenge to develop robust identification methods able to suppress the significant atmospheric influence on the models.

Regarding the validation of the neutral point estimation method, it appeared to be satisfactory for all tested platforms, even though a precise evaluation was not possible with the experimental technique used here.

The findings suggest that the servo-actuator may play a critical role in the system response and it would be convenient to increase the fidelity of its theoretical model. For other electronic systems, it seems worthless to increase the complexity of their models as long as the fidelity of the aircraft dynamics is low.

If a complete aircraft system identification is carried out and enough aerodynamic data is available, it could be convenient to use a more advanced flight dynamics simulator such as the Swedish FOI ADMIRE model, the open-source JSBSim, or even carry out complete hardware-in-the-loop simulations using the commercial Laminar Research’s X-Plane environment.
6 Discussion and Conclusions

6.2 Flight Testing

The methodology used during the flight testing was in general satisfactory taking into account the available resources. However, the efficiency of the flight tests could be significantly improved by splitting the tasks between one or two assistants apart from the pilot. This could be especially important in flight testing within VLOS. Regarding pilot rating, it would be convenient to have more than one pilot flying the same test and rate the aircraft according to an unanimous opinion. Other specific observations and conclusions for each FCS are included below.

Basic Single-Axis Gyroscope Sensor Approach

It has proved successful for all the unstable platforms in both pitch rate and pitch angle (AVCS) FCL modes. Its performance was, however, connected to the aerodynamic effectiveness of the elevator. Regarding the physical limitations of the system, the main constraint appeared to be the performance of the elevator servo rather than the capabilities of the gyroscopic sensor.

Even though the system was extremely robust and easy to adjust, there was a clear lack of flexibility and more advanced settings, which become especially important for canard-delta configurations. The fixed gains independent to airspeed are also a disadvantage, although the system was not prone to saturation in most conditions.

Since the pitch rate FCL did not trim the aircraft, it became difficult to fly with highly unstable, tail-heavy configurations. The pitch angle FCL resulted more convenient in most cases. However, both FCL modes suffer from deviations in banked turns due to the constant pitch rate produced. A washout filter might solve this issue.

Another issue is that the system operated as a SAS, independently of the pilot commands. Hence, it compensated also the desired inputs and the overall response of the aircraft is slightly reduced as the gyroscope gain increases.

Advanced Flight Control System

The FCS tested has proved extremely versatile and effective. This flexibility allowed its use with multiple platforms, and it should not be a problem to adapt it to more advanced multi-surface configurations.

It solved the problems that the single-axis gyroscope presented, and both pitch rate and pitch angle FCL modes performed satisfactorily. On the other hand, it was much more complex to adjust and tune appropriately. Observations suggest that the tuning process for unstable models is critical, and it must be performed carefully and progressively from a configuration with positive static stability. The dynamics are again correlated with the aircraft configuration and its elevator effectiveness. In general,
it appears that as the aircraft becomes more responsive a slight decrease of the pro-
portional gain may be needed, together with a considerable increase of the derivative
gain in order to suppress the short-period oscillations.

Among the proposed FCS modifications are: increase the 50 \( \text{Hz} \) refresh rate of the
PWM signal output to the longitudinal control servo-actuators, since fast-response
digital units could take advantage of higher frequencies. This hardware has already
proved to support satisfactorily output rates up to 400 \( \text{Hz} \) with alternative software
versions [56]. Moreover, it could be convenient to smooth the transfers between FCL
modes, for example by using linear faders. Additionally, the implementation of a
stall-detection system or an AOA limiter may be necessary. In the case of designing a
completely new FCL based on AOA, inertial augmentation would be needed, possibly
by using a Kalman filter.

Different longitudinal control distributions in multi-surface configurations are to be
investigated in the near future by using this FCS in the manufactured delta-canard
test-bed.

**Limits for Manual Control**

It can be concluded that it is not possible to determine an exact boundary between con-
trollable and uncontrollable configurations. The observations suggest that the degra-
dation of the flying qualities is highly airframe-dependent, and other factors such as
airspeed and wind turbulence also affect strongly the outcome. From a conservative
point of view and assuming an average skilled pilot, most platforms could be flown
manually up to the neutral static stability margin, where they present significantly
degraded flying qualities that could be rated as level 8-9 in the Cooper-Harper scale.
It was possible to continue further into negative static margin with certain platforms,
although it was not advisable to go beyond minus five percent in any of the cases.

Furthermore, it appeared that the conventional configurations presented a somewhat
sharper deterioration of the flying qualities than that of the delta-canard platforms
regardless of their inertia, phenomenon which could be related to the aerodynamic
characteristics of the delta wing.

**6.3 Instrumentation and Data Analysis**

The suite of low-cost sensors used in the advanced FCS demonstrated a surprisingly
good performance for this level of research, and their resolution was acceptable con-
sidering the price tag. The redundant IMU system and the automatic monitoring of the
drift and bias of the inertial sensors were remarkably useful features. Apart from the
moderate sensor quality, the challenge lies in the correct calibration of some airdata
sensors, due to the fact that a wind tunnel was not available and that some common
calibration methods are not appropriate for small-scale models.
The appropriate installation of the instruments resulted of critical importance for a correct performance. An optimum layout was certainly complicated to achieve in some of the platforms. Apart from accounting for the respective installation errors, some inconspicuous interactions resulted especially problematic, such as airflow circulation inside the fuselage affecting the barometric sensor, or the electromagnetic interference of the power systems on the long sensor cables routed along the aircraft.

The EKF algorithm implemented in the FCS software proved essential for filtering and correcting the deviations of some sensors. Observations suggest that, although computationally expensive, its use enhances significantly the performance of the FCS even with low-quality sensors.

The developed AOA transducer has shown to work exceptionally well, although it was very sensitive to atmospheric disturbances and the own aircraft motion. The implementation of a low-pass filter similar to that designed in this study would be necessary to use AOA data for control purposes. Furthermore, observations indicate that a FCL based on the AOA would need inertial augmentation, which as mentioned before could be done through the EKF.

Regarding flight-test data analysis, the analysis tool FLOGA v1.00 has been specifically developed and it is also delivered with this thesis. The sample rate provided by the FCS built-in flash card data logger ranged from 5 Hz for the GPS receiver to 50 Hz for the IMU data and some external sensors, which are considered sufficient for enthusiast-level projects and higher education in flight dynamics. However, for scientific research and professional applications it would be necessary to increase the sample rates in order to allow more advanced post-processing.

### 6.4 Some Suggestions for Flight Testing Unstable Models

It is possible to outline some general suggestions regarding flight testing longitudinally unstable models based on the experience accumulated throughout this work. Although these are not intended to be categorical guidelines, they may be applicable to platforms similar to those used here. Items are listed in no particular order.

1. The correct performance of the flight control system is vital and there is almost no margin for recovery from any malfunction. Therefore, it is especially important to minimise human error by following methodical procedures during all phases of the field test.

2. The pilot must always be aware of the trim settings demanded by the aircraft when switching between assisted and manual control modes. This is especially relevant when flying tail-heavy aircraft or when the centre of gravity varies during flight. If the flight controller firmware has a function that allows automatic
6.4 Some Suggestions for Flight Testing Unstable Models

“learning” of the trim state, it can be very practical to use it. It is also recommended to check that the controller is allowed to use enough elevator deflection to trim the aircraft even at very low speeds.

3. During the take-off roll, the aircraft should be allowed to accelerate to a sufficiently high rotation speed in order to ensure full elevator authority. Short take-offs at steep angles are not recommended for severely unstable aircraft.

4. In the case of hand-launched aircraft, the launch must be performed at horizontal or slightly nose-down angle with sufficient initial speed because of the same reason. This is particularly important for pusher aircraft and configurations where the thrust flow does not hit the elevator surfaces.

5. A similar situation occurs in final approach and landing. Since the effectiveness of the elevator is critical, the approach must be performed with sufficient airspeed and the flare manoeuvre should never be excessive. It must be taken into account that a tail-heavy unstable aircraft can easily continue pitching nose-up by itself if the elevator stalls.

6. The aircraft should not be stalled at any time when flying in assisted or automatic control modes unless the algorithm has the ability to detect the stall and perform a recovery. Furthermore, it should be verified that the pilot’s recovery inputs can override the deflections commanded by the autopilot in case of departure, as long as switching to manual control mode is not a valid option for unstable aircraft. Considering that the feedback to the pilot is particularly limited in control-assisted, R/C aircraft, an AOA limiter onboard or a stall awareness system at the ground station can be effective ways of preventing the pilot from entering a dangerous situation.

7. Since relaxed longitudinal stability will in most cases degrade also the directional stability, it is convenient to increase the rudder deflection during coordinated turns to avoid sideslip. It has been observed that, in some extreme cases, excessive sideslip during a banked turn can develop into a lateral stall. It must be checked that the vertical tail area of the aircraft is sufficient for the desired level of negative longitudinal static stability.

8. Gain auto-tuning functions of the autopilot will in most cases not work properly with unstable aircraft. The controller gains must be adjusted manually. It is however recommended to start from well-proven, soft settings in a stable configuration, then reduce progressively the static margin and make the corresponding gain adjustments until the static margin is zero, and continue from there in small steps by applying conservative changes. If the aircraft has the possibility of changing the static margin in-flight (by for example shifting its centre of gravity or its neutral point) more aggressive steps could be taken.

9. Appropriate damping of high-frequency pitch oscillations becomes critical and, in general, a significant increase of the pitch controller derivative gain may be needed. Therefore, high-speed, powerful and reliable servos must be used for
the elevator surfaces, although it has been observed that standard analog R/C servos can perform satisfactorily in some platforms.

10. It should be taken into account that as the static stability margin is reduced, not only natural damping disappears, but also the aircraft response becomes more violent. This is especially dangerous at high speeds if inappropriate airspeed-driven gain scaling is applied. The proportional gain of the pitch controller, together with pitch rate limiters, response-time factors, and any other parameter affecting the direct response signal path must be revised in every step in order to avoid high load factors that could burst the airframe.


Bibliography


Appendix A

Figure A.1: General view of the Cub test-bed aircraft.

Table A.1: Technical specifications of the Cub test-bed aircraft.

<table>
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<th>Specification</th>
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Appendix A: Characteristics of the Test-Bed Aircraft

Figure A.2: General view of the Balsa test-bed aircraft.

Table A.2: Technical specifications of the Balsa test-bed aircraft.

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**Figure A.3:** General view of the Rafale test-bed aircraft.

**Table A.3:** Technical specifications of the Rafale test-bed aircraft.

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**A Appendix A: Characteristics of the Test-Bed Aircraft**

![General view of the Gripen test-bed aircraft.](image)

**Figure A.4:** General view of the Gripen test-bed aircraft.

**Table A.4:** Technical specifications of the Gripen test-bed aircraft.

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