Design of a Ground Static Load Test for a 3 m Joined Wing Flexible Sensorcraft

Francisco Maria Pimenta de Castro de Almeida

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Supervisor: Prof. Afzal Suleman

Examination Committee
Chairperson: Prof. Fernando José Parracho Lau
Supervisor: Prof. Afzal Suleman
Member of the Committee: Prof. Filipa Andreia de Matos Moleiro Duarte

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I want to dedicate this work to my mother, father and sister who have been my biggest supporters.
Acknowledgments

This thesis mirrors the amazing experience I had in Canada working with exceptional and tremendously professional people at the Centre for Aerospace Research (CIAR). This journey made me grow up not only professionally but also personally, which is really important in this stage of my life.

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Abstract

Operational requirements for High Altitude, Long Endurance (HALE) aircraft capable of carrying advanced sensor arrays over large territorial areas can be achieved by light weight and geometrically unconventional aerostructures. The Boeing Joined Wing SensorCraft (JWSC) concept is one such example. However, the joined wing configuration presents design challenges due to its non-linear aeroelastic behaviour.

Computational models to predict the non-linear aeroelastic behaviour have been developed and reported in the open literature, however experimental data to benchmark analytical predictions is still non-existent.

The objective of this thesis is to design a Ground Static Load Test (GSLT), its apparatus and procedures to evaluate the structural response of a flexible joined wing sensorcraft, to understand its static behaviour in terms of displacements, strains and aft wing twist angles when subject to a $2.25 \, g$ pull-up maneuver.

A careful test plan matrix was designed including the appropriate boundary conditions, finite element solvers, ground static test rigs and data acquisition systems. The structural computational model was updated based on the linear deflections. Finally, the validity of the model updating process of a non-linear structure using linear displacements was assessed.

Keywords: sensorcraft, ground static load test, non-linear, finite element model, model update.
Resumo

A nova exigência por aeronaves de alta altitude e longa autonomia com a capacidade de transportar avançados sensores sobre extensas áreas territoriais, originou um novo foco de projecto em aeronaves leves com estruturas e geometrias pouco convencionais. O conceito de *joined wing sensorcraft* da Boeing é um exemplo. Esta configuração possui benefícios mas também desafios. Uma das maiores preocupações é o comportamento não linear da asa posterior. Enquanto modelos computacionais têm sido desenvolvidos ao longo dos anos para estudar estes efeitos, dados experimentais, para validar os cálculos analíticos, são ainda uma escassa matéria.

O objectivo desta tese é conceber um teste de cargas estático para um *joined wing sensorcraft* flexível por forma a perceber o seu comportamento (medindo deslocamentos, deformações e ângulos de torção) quando submetido a cargas estáticas que pretendem representar o equivalente a uma manobra de $2.25 \ g$ pull-up. O segundo objectivo é estudar a viabilidade de actualizar o modelo de elementos finitos, com base em deflexões ainda consideradas lineares, usando uma solução numérica linear.

Desta forma, um plano de teste foi elaborado, onde condições de fronteira, soluções numéricas, equipamentos e sistema de aquisição de dados podem ser analisados e escolhidos. O teste é levado a cabo, e os dados importantes são recolhidos. Por fim, com base nos dados recolhidos das deformações lineares, o modelo computacional é actualizado.

O intuito é elaborar e introduzir todo o processo inerente aos resultados obtidos, avaliando a viabilidade de actualizar um modelo não linear utilizando deflexões lineares.

**Palavras-chave:** sensorcraft, teste de cargas estático, não linear, modelo de elementos finitos, validação do modelo computacional.
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Nomenclature

Greek symbols

\[ \Delta z \quad \text{Displacement.} \]
\[ \theta \quad \text{Twist angle around } y \text{ axis.} \]
\[ \varepsilon_{xx} \quad \text{Longitudinal strain.} \]

Roman symbols

\[ x, y, z \quad \text{Cartesian Components.} \]

Subscripts

\[ FE \quad \text{Finite element.} \]
\[ GSLT \quad \text{Ground static load test.} \]
\[ O \quad \text{Original loads.} \]
\[ PE \quad \text{Pseudo equivalent loads.} \]
\[ wt \quad \text{Wing tip.} \]
\[ xx \quad \text{Longitudinal.} \]
# Glossary

<table>
<thead>
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<th>Term</th>
<th>Description</th>
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<tr>
<td>ASWing</td>
<td>Aerodynamic Structures Wing is a program of aerodynamic, structural and control-response analysis for aircraft with flexible wings of high aspect ratio</td>
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<tr>
<td>BC</td>
<td>Boundary Conditions</td>
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<td>BFP</td>
<td>Bifilar Pendulum</td>
</tr>
<tr>
<td>CBEAM</td>
<td>Is a FE beam element which is one-dimensional structural member which supports tension, compression, axial torsion and bending</td>
</tr>
<tr>
<td>CG</td>
<td>Centre of Gravity</td>
</tr>
<tr>
<td>CQUAD</td>
<td>Is an isoperimmetrical plate quadrilateral element with optional coupling of bending and membrane stiffnesses</td>
</tr>
<tr>
<td>DOF</td>
<td>Degree Of Freedom</td>
</tr>
<tr>
<td>EM</td>
<td>Elasticity Moduli</td>
</tr>
<tr>
<td>FEMtools</td>
<td>Is a solver and software program providing advanced analysis and scripting solutions for, among others, updating FE models for structural analysis</td>
</tr>
<tr>
<td>FE</td>
<td>Finite Element</td>
</tr>
<tr>
<td>GSLT</td>
<td>Ground Static Load Test</td>
</tr>
<tr>
<td>HALE</td>
<td>High Altitude Long Endurance</td>
</tr>
<tr>
<td>ISR</td>
<td>Intelligence, Surveillance and Reconnaissance</td>
</tr>
<tr>
<td>JWSC</td>
<td>Joined Wing SensorCraft</td>
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<tr>
<td>MATLAB</td>
<td>Is an iterative software used for numerical computing applications</td>
</tr>
<tr>
<td>NL</td>
<td>Non-Linear</td>
</tr>
<tr>
<td>PMI</td>
<td>Projection Moiré Interferometry</td>
</tr>
<tr>
<td>PUM</td>
<td>Pull-Up Maneuver</td>
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<tr>
<td>RE</td>
<td>Relative Error</td>
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<td>-----</td>
<td>----------------</td>
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<tr>
<td>UAV</td>
<td>Unmanned Aerial Vehicle</td>
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<tr>
<td>VIC</td>
<td>Visual Image Correlation</td>
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Chapter 1

Introduction

Ground static load tests of aircraft are typically performed very late in the development process. The main purpose of the test is to obtain experimental data of the whole aircraft structure for validating and improving its structural models. Among other things, these models are used to carefully plan the safety-critical in-flight tests. It is well known that finite element predictions are often called into question when they are in conflict with test results. The area known as model updating is concerned with the correction of finite element models by processing data related to the response from test structures.

GSLT requires a procedure so that enough data is collected to validate and update the finite element model. Several preliminary analyses must be done, scripts must be elaborated, the test must be planned and executed.

In this chapter, the background and motivation is presented. Afterwards, the contributions coming from this work are discussed and finally a detailed thesis outline is given.
1.1 Background and Motivation

In the aerospace industry, as one of the most technological and demanding fields in engineering, the need for new solutions and next generation ideas is always around the corner. Aircrafts are used, nowadays, for several and diversified applications, such as commercial transportation, military missions, humanitarian help, scientific investigation and so on. In an even more globalized world, the need for interconnection originates an increasing request for transportation of people and goods. Existing airplanes become inadequate and unsustainable. Also, the demand for smaller, more efficient, longer range and endurance Unmanned Aerial Vehicles (UAVs) has forced engineers and researchers to challenge the bounds of imagination and look towards non-conventional ways of thinking and configurations, to meet those requirements. As a consequence, conventional aircraft design is even more defied every day, with new and innovative materials, configurations and geometries [1].

Despite being some years old, the JWSC idea/concept is one such example, where there is usually an aft wing that connects to the front one (usually in a lower position), forming a complex over-constrained system, in a rhombus like form, creating thus additional design space and allowing more options in terms of aerodynamics, flight mechanics, engine integration and aero-elasticity [1], [2].

Studies done in the past decades made claims on the possible advantages or disadvantages, when compared to orthodox configurations. The absence of funding and commitment from the diversified aerospace agencies did not yet allow a deep study of this kind of design. Moreover, there are statements which contradict previous findings and a clear picture of the real potentials/challenges of the JWSC are still on debate. However, the need of addressing the previously referred goals makes the understanding of this concept a necessity [2].

Military applications were also a goal for joined wing configurations, when the sensorcraft was studied in the United States. The main goal for this configuration military wise, was to have an effective design for the antenna, directly integrated into the wing structure. This allows the achievement of superior high-altitude Intelligence, Surveillance and Reconnaissance (ISR) [1]. The point was enabling worldwide coverage without special deployment of the aircraft launch and recovery operations [3]. When talking about a traditional aircraft geometry, its design typically begins with the development of an airframe first, followed by the selection of several components and sensors that are chosen to integrate into it. A sensorcraft however, is designed in an opposite way, being the aircraft’s platform especially developed around an optimized group of sensors that are previously chosen based on the kind of mission to be carried out.

Boeing has a concept of their own illustrated in figure 1.1. Boeing’s concept is a solution proposed to the United States Air Force Research Laboratory’s Sensorcraft Request. The aim was to inspire innovation and high-end technology. The design goal was to integrate the sensing capabilities into an UAV capable of 30 h of endurance at 2000 nm range [3].

The joined wing concept was first suggested by Wolkovitch [4] in the 70s, with an overview being published in 1986. Being it a HALE UAV, and having potential structural and sensor carrying benefits, innovative capabilities over vast territorial areas can be easily achieved. However, it still has its own set
of challenges. The main benefits for this kind of geometry are:

- Full 360° coverage with an array antenna integrated into the wing panels.
- Potential reduction of wings’ structural weight with an even stiffer structure.
- Reduced induced drag which means better aerodynamic efficiency.

Regarding disadvantages, they might be related to elastic instability (buckling) on the aft wing due to providing support in compression to the forward wing [4].

A partnership was determined between the University of Victoria and Boeing so that several scale models could be designed, developed and built in-house, to thoroughly test and evaluate the feasibility (both in conception and flight) of this kind of design, delivering then results and pertinent conclusions to Boeing. As part of the whole sensorcraft project, several test models were developed and shall be explained as follows:

- Rigid 5 m span UAV
- Rigid 3 m span UAV
- Rigid 1.8 m span UAV
- Flexible 1.8 m span UAV

There are two different conceptions that should be explained. The sensorcraft, for any size, might be built into two different versions: rigid and flexible. For both, fuselage and boom which can be seen in figure 1.2 are considered and built as rigid components. The difference relies on the wings, flexible for one version, and rigid for the other one.

A rigid structure is built in composite materials (carbon fibre), which form a previously moulded structural skin, similar to a monocoque. The carbon fibre skin is reinforced with bulkheads inside. Therefore the structure is stiff enough to be considered as perfectly rigid. It does not bend or flex under an applied force, whereas a flexible structure (in this case the wings) has an inner flexible spar (i.e. that easily bends undergoing any kind of load) which is the only structural agent.

Thus, for a rigid version of the sensorcraft, every component is built as rigid, whereas for the flexible version, the aft and forward wings are built as flexible.
Fuselage is built as stiff as possible to be considered a rigid monocoque structure. It has a main transverse rib, one back bulkhead and two longitudinal ribs (one at each side) where wings are attached. It is covered with a composite material skin. The main transverse rib which might be seen in picture 1.3(b) is responsible for the structural rigidity of the fuselage. Rigid wings are built and designed in a conventional rib/spar with composite skin monocoque design, so that the majority of loads undergone by the wings are taken by the wing skins themselves. In this case, ribs are intended to act only as shear webs with most of the axial and bending loads taken by the upper and lower skins, exactly like a monocoque structure. Fuselage and boom present a similar design [1]. Flexible wings are designed in a slightly different way, now having a flexible aluminium spar that withstands all loads, and the skin divided into panels which are attached to each rib of the wing as it may be seen, again, on figure 1.3(a). This flexible version of wings intends to make them as we see nowadays in commercial airplanes, although its flexibility is excessive with the intention of having an aeroelastic scale model [1].

Regarding the work being done for this thesis, a GSLT is being designed around a newly built 3 m span flexible version, in order to investigate its highly Non-Linear (NL) behaviour, due to large deflections, and to characterize structure’s response. This version can be seen in figure 1.2 where flexible parts can be easily identified. A scale test model provides a low cost and effective way to investigate these non-linear aeroelastic responses and to validate/tune existing computational models. Beyond the work done here, the project alongside Boeing also intends to test additional technologies such as stability/control methods for highly flexible structures, dedicated to stress investigation and gust load
1.2 Contribution to the State of the Art

The main motivation in this thesis is to design a GSLT with the objective of updating the existing FE model, characterize the structure non-linear behaviour, assess the efficiency and precision of the available NL solution methods and evaluate the feasibility of using a linear procedure to update the computational model. The update of a Finite Element (FE) structural model is an iterative process where the used software (FEMtools) consecutively solves the structural FE analysis and changes the model properties, until the computational displacements (chosen target response for model update) match the experimental data. The update procedure might be done using any target quantity, like displacements, twist angle, strains, etc. In this work, displacements are chosen as the target responses. Therefore, if the update procedure is hypothetically being carried out with the non-linear solution method (as recommended for this kind of flexible structure) the update procedure becomes a hugely time-consuming task. The FE model might be solved a considerable amount of times and each time might take about 5 min. Besides, with the non-linear method, the update itself becomes even more trickier because the likelihood of divergence when changing properties of the model is considerably larger than when a linear one is used. Taking everything into consideration, the main goal is to study the feasibility of updating an extremely flexible FE framework, using small linear deformations as experimental results and linear solution method as a tool for the update procedure. A linear update procedure is less time consuming and simpler when compared to the non-linear one. In addition to this, the update shall be based on displacements as target responses. Other quantities will be measured throughout the test but in terms of the update itself, the linearity is investigated based on displacements primarily, as it can be seen ahead.

The static load test will be carried out using an imposed deformation that shall be an approximation to what the sensorcraft undergoes on flight during a specific 2.25 g Pull-Up Maneuver (PUM). Besides, it is important to gather data relative to this highly flexible structure and to analyse NL response, mainly on aft wing where it is more evident. Due to the high aspect ratio of this joined wing configuration, very flexible wings are part of the implications. Meaning we necessarily have to experience large deflections and linear assumptions may no longer be valid. Thus, NL analysis is essential both locally (skin panel buckling) and globally (front or aft wing buckling) [3]. That is to say, non-linearities in this sort of configuration might have important outcomes long before they are, in reality, apparent, meaning when considerable displacements take place. Whilst Lee et al [5] stated that non-linearities should be accounted for when the deformation is sufficiently large. Moreover, they also think that focus should be into relevant enough geometric non-linearities.

Linear and non-linear computational models have been developed to predict highly NL behaviours presented by the structure such as buckling in the aft wing or even flutter under gust conditions led by forces responsible for large deflections that occur on such geometries. Nevertheless, experimental ground and flight test data is still a lacking subject, regarding this particular concept. Ground test and alleviation schemes.
flight data are greatly advantageous to benchmark the analytical predictions and to allow an update to the FE model.

Resultant contributions from this work are the structural characterization of the JWSC configuration. This variety of design has been deeply studied but no experimental data exists that quantifies the highly NL response for a flexible version. The purpose is to provide data and independent analysis of a configuration that demonstrates significant structural non-linearities that aerospace community as a whole, can handle for validation of existing analyses, design tools and methodologies.

1.3 Thesis Outline/Test Design Plan

In this section, the procedure to carry out the GSLT is presented. Being this test conceived from a blank sheet, the work done entails not only the execution of the test itself but also a considerable amount of preliminary analyses to be done. Since this thesis is part of the Boeing JWSC project, the FE model framework was already available although it is constantly evolving and being updated. Regarding the desired behaviour of the sensorcraft, GSLT will be designed based on the deformation equivalent to a $2.25\,g$ PUM. The test plan procedure is illustrated in figure 1.4.

```
Aeroelastic Framework (FE Model)

Appropriate Aero Loads

Generate Simplified
Representative Load

Design Static Rig

Experimental Static Linear
Small Deflections

Update FE Model
with Linear Solution

Experimental Static Non-Linear
Large Deflections

Flight Testing

Compare FE Model
to Flight Results
```

Figure 1.4: GSLT’s schematic.

- **Aeroelastic Framework.** There is an aeroelastic framework which is a combination of several simulation and optimization tools which can be seen in figure 1.5. There is an interdependence
between structural and aero models. FE Nastran’s structural model needs to be validated and if required, updated. It is of vital importance because the whole project depends on it, given its direct influence on the ASWing’s (Aerodynamic Structures Wing) aero model.

- **Appropriate Aero Loads.** Appropriate aero loads were determined to produce the desired NL deformation, i.e. a 2.25 g PUM that will be carried out on flight later on.

- **Generate Simplified Representative Load.** In order to behold aircraft deformation on ground as it takes place on air, all the loads previously determined undergone by the structure, must be simplified into equivalent concentrated loads that might be easily applied on the aircraft, resulting in a pseudo equivalent NL behaviour.

- **Test Rig Design.** An apparatus including jigs, mass bags, and any other required equipment is designed at this stage.

- **Experimental Static Linear.** Small fractional loads of the maximum load case (2.25 g) were considered, previously investigated in Nastran to ensure linearity. Resulting deflections measured in the aft tail and main wing were used to update the FE model.

- **Update of FE Model with Linear Solution.** First, masses of components were updated on the computational model. Then inertias based on experimental data and finally Elasticity Moduli (EM) of structural elements based on linear deformation.

- **Experimental Static Non-Linear.** The load previously determined, comparable to the equivalent 2.25 g maneuver was applied. Strain, twist angle and displacements in the forward and aft wings are collected and compared to Nastran’s NL solution.

- **Flight Testing.** Work still to be done in the future. Flight test data will be compared to FE results and assessments will be made over the assumptions taken and how the 2.25 g maneuver was correctly executed on ground.

Figure 1.5: Integration of computational structural and aero tools [6].
1.4 State of the Art/Literature Review

As previously said, the joined wing concept was initially proposed by Wolkovitch [4], which also demonstrated that the wing bending axis is tilted, relatively to the horizontal, because there is an out-of-plane arrangement of wing surfaces. This allows the particular advantage of placing material near upper leading and lower trailing edges to increase structural rigidity of the structure. This is called the Wolkovitch effect [4].

More detailed studies and analyses by Kroo, et al [7], of NL FE models, showed the presence of buckling on the aft wing when the wings are subjected to very large deflections. Wings are highly flexible, hence large deflections are necessarily occurring, therefore linear assumptions will no longer be valid.

At some point, people have been only considering non-linear buckling response on the aft wing. Front wing buckling had been overlooked until the high-fidelity model and analysis of Canfield, et al [8]. The argument was that the wing bending up and forward allows both the aft and front wings to buckle whenever compression is present. Also, “the joined wing configuration exhibited large geometric non-linearity below the critical buckling eigenvalue. Thus, non-linear analysis was required to model correctly this joined wing configuration” [8]. The scope of this thesis is then, to confirm this experimentally and to better understand NL response for this particular geometry.

Lin, et al [9] investigated instabilities on the structure due to moment, shear and axial reaction loads at the joint, which may be divergence and flutter. Also, they studied joint fixity influence on the stiffness and strength characteristics of a joined wing geometry. A rigid joint was found to be the best compromise for mixed stiffness and strength benefits, which will be an important matter for this project as it will be seen ahead.

Rasmussen, et al [10], investigated several configurations and concluded that a lightweight joined wing concept (like CfAR’s aeroelastically scale test model) is possible to achieve, although being highly non-linear.

Also, recent studies carried out by Demasi, et al [11] [12] [13] showed how important a NL analysis is when designing a joined wing configuration (in this particular case, when designing a static test for the JWSC) as well as the snap buckling instability of such configuration. Snap buckling is a particular buckling phenomenon, which happens in structures undergoing axial compression. The structure usually snaps between two (often) symmetric buckle states, under some perturbation. They further concluded that the flexibility of wings, wing joint’s connectivity and anisotropy of the composites may all have a large repercussion on the NL response.

Since this is a GSLT where displacements need to be measured as precisely as possible, there is the need to investigate on such similar solutions, done previously. Ifju, et al [14], Albertani [15], and Stanford [16] were amongst the first investigators to use Visual Image Correlation technique (VIC) [17] for the full-field basis deformation measurement of rigid and flexible micro aerial vehicles’ wings.

Galvão, et al [18] tried using stereo photogrammetry for displacement measurements of a 12.9 cm by 5.9 cm membrane wing. There was a camera with a spatial resolution of 0.2 mm per pixel and direct
linear transformation to get marker’s coordinates. Therefore it was reported a measurement uncertainty of \( \pm 35 \mu m \) for in-plane measurements and \( \pm 40 \mu m \) for out-of-plane. Data is available at discrete markers placed throughout the wing. Jacob, et al [19] tried photogrammetry as well, to measure the deformed wing shape of inflatable wings. Fleming, et al [20] used Projection Moiré Interferometry (PMI) which does not demand the use of markers, i.e. a fringe pattern is projected onto the wing surface. However, with camera spatial resolution of 0.22 mm/pixel, the displacement’s resolutions accounted were about 250 \( \mu m \). For this work, as can be seen ahead, a Coordinate Measuring Machine (CMM) was chosen to measure displacements and twist angles. This scanner has an annunciated precision of 100 \( \mu m \) which is between both solutions found in previous references.

Regarding the update procedure to follow, some work was developed and tested for complex structures which had models to be updated. Its finite element model was divided into substructures, meaning that the design parameters are updated by regions, and proved to be a useful method for large and intricate structural finite element model updating [21].
Chapter 2

Computational Analysis

Regarding this section, the use of a FE approach is discussed as a preliminary computational structural analysis. Design methods have been upgrading, reaching unprecedented levels of efficiency. This has been possible due to a constant improvement of technology in the several engineering areas and the introduction of increasingly sophisticated computational models. Using FE methods turns out to be a very appropriate tool that gives designers and engineers a first look at the behaviour of a structure, under certain imposed conditions. It can be applied to a wide variety of engineering fields, such as aeronautical, mechanical, biomedical, construction and so on. The need for these numerical simulations relies on the fact that designed structures may be tested even before its own construction, which means that if a total change of paradigm is needed, no resources were spent in vain. Therefore, a big save both in time and money is accomplished.

Nevertheless, one must not base all the work and assumptions made on FE methods’ results. A FE analysis does not provide complete information about the behaviour of the desired structure, given the fact that it will always be an approximation of reality, either by the structure’s representation, the chosen mesh, elements to be used, their formulation, idealized connections and so on. To develop one, even more accurate, computational model, it must be updated with results determined from experimental testing. This is one main target of this work. In this work, FE analyses will provide useful information about static deflections and deformations, providing preliminary conclusions so that decisions may be taken in order to carry out the GSLT successfully. As previously said, the goal is to have an approximated deformation, undergone by the structure on ground, as it has when undergoing a $2.25 \, g$ PUM in flight.
2.1 Finite Element Model

When dealing with finite element modelling, it is important to bare in mind that there is a big likelihood of having differences between the final model and the as-built aircraft, inducing errors that should have been previously considered. Usually, results of numerical simulations are not in good agreement with the measured structural behaviour. This might happen for several reasons, namely material property randomness, construction deviation of the real structure and boundary conditions. Besides, mesh density, geometrical representation, chosen solver and its accuracy are all preponderant facts which influence how a FE analysis is done successfully. Masses must be estimated and inserted into the model. Finally, material properties, like Elastic Modulus (EM), are updated based on experimental results.

Apart from being used as prediction of reality, and being it the main target for this work, it is also used to predict aircraft’s behaviour in order to better enhance and carry out the test itself. The best load cases might be calculated as well as the useful data determined as such. The forehand analysis of the computational model allows also to determine the boundary conditions to be used.

As explained previously, the finite element model is useful not only for structural analysis in Nastran, but also for aerodynamic purposes. First of all, FE model is being explained, namely constitutive elements, materials, way of conception and Boundary Conditions (BC). The framework and the outside body and appearance of the as-built aircraft are represented in figure 2.1, where outside dimensions are given. Computational model can be seen in figure 2.2 where top view and perspective are presented. As described in chapter 1 regarding sensorcraft’s geometry, FE framework intends to represent as accurately as possible, its structural body, i.e. its skeleton.

Beam elements are the main components of the computational model, exception made for the aft wing which is modelled with plate elements, for additional precision. The goal is to study the behaviour of the sensorcraft as a whole and specifically analyse the rear wing. Thus, the model becomes fast to solve and assures a superior agreement with ASWing. CBEAM elements, which are one-dimensional structural members which support tension, compression, axial torsion and bending, are simple and fast to compute through any FE solver, and for this reason it is the main constituent of the FE framework. Although these elements may be simple, they still provide good and high fidelity results, even for a three-
dimensional structure. Therefore, the thinner lines that might be seen in figure 2.2, namely the ones that make up fuselage, forward wing (whose spars’ cross section is a regular rectangle), wing tip, boom, interface between forward and aft wings, and strake, consist of CBEAM elements. Aft wing’s spars have CQUAD plate elements as forming ones, given its different cross section which is represented in figure 2.2. These elements are bi-dimensional isoperimetrical quadrilateral plates with optional coupling of bending and membrane stiffnesses. Being aft spars a crucial part of the structure, and the ones that provide more data, they require a more rigorous modelling. There are also some rigid elements throughout the structure, which are used at the location of every rib with the purpose of placing point masses, which should account for the weight of aerodynamic panels and ribs themselves. Besides, rigid elements are also added to create additional nodes in order to have a better correlation between FE nodes and experimental points, later on.

Boundary conditions are perhaps one of the trickiest parts. One must be sure that BC match between flight, computational model and GSLT. If this can’t be achieved (as will be seen later on) other ways have to be found, to prevent this problem from changing results and induce one into wrong arguments and conclusions. It is known that in flight, specifically for a 2.25 g PUM, structure deforms about the centre of gravity. The biggest problem arises with the implementation of the GSLT. First, the way the sensorcraft was designed and built makes it impossible to hold it on ground exactly where centre of gravity is. The point in question is located where there is only a weak skin of composite material, where loads or supports of any kind can not be applied. Therefore, the closest to this point is the hole used for the hook that launches the UAV into flight. This hole is placed $d = 0.2168$ m ahead of the centre of gravity. This was the first idea, and the one that was used with the preliminary analysis done in section 3.2, i.e. fixed BC being applied at the hole in question. Distance between hook’s hole and centre of gravity is not negligible, however results do not change on a FE analysis for both different cases, as tested, because the aircraft’s fuselage is stiff enough to be considered rigid as it might be seen later on. This means that having these three cases (flight, FE model and GSLT) with slight differences in BC is not a severe problem. In short, we would have flight with known conditions being hypothetically fixed at the centre of gravity, FE model with fixed support where the hook for the launcher is placed and GSLT with the same BC as the FE model. Hence, boundary conditions can easily be matched between test and computational model, which is what matters the most for this work. Therefore and after analysing through some FE simulations, this slight difference can be considered negligible, and three cases can be acknowledged as having the same BC.

Regarding analysis itself, MSC Nastran was chosen to be the FE solver which has a solution method that accounts for the high non-linearity of the displacements for beam element structural models. As it will be seen ahead in section 4.1, the solver and solution method will be tested with a simpler experimental test, given the fact that this is a main concern: solver’s feasibility and efficiency. The solution method in question is number 400 (SOL 400).

Linear analyses assume a linear relationship between the load applied to a structure and its response. A number of important assumptions and limitations are inherent in linear static analysis. It assumes the supposition of a linear force-deformation and stress-strain response of the structure and
material. This limitation is imposed by the solution method chosen. The stiffness of a structure in a linear analysis depends on its initially non-deformed state. Linear static problems are solved in one step, by a single decomposition of the stiffness matrix. This type of analysis is restricted to small displacements, because otherwise the stiffness of the structures changes and must be accounted for by regenerating the stiffness matrix. As the structure deforms due to loading, the stiffness changes, and as the stiffness changes, the structure’s response changes. As a result, non-linear problems require incremental solution schemes that divide the problem up into steps calculating the displacement, then updating the stiffness. Each step uses the results from the previous step as a starting point. As a result, the stiffness matrix must be generated and decomposed many times during the analysis adding time and costs to it [22].

If any of these assumptions are no longer valid, once the load surpasses a determined level, a linear static solution is no longer accurate and a non-linear numerical method should be considered, to achieve a solution that embodies all of the physics of the problem. Although SOL 400 is effectual for use on linear problems, it has been designed to solve NL problems that include large deformation, material, and contact/boundary non-linearities [22].
Chapter 3

Test Planning

The desired operational limit for the sensorcraft is based on a $2.25$ g pull-up maneuver, which suggests extremely complex aerodynamic forces and moments, besides the inertial load, being applied into the structure. This being said, it is intended to determine and analyse structure's behaviour for a representative simplified load, with which the static load test shall be executed. One of the first challenges of a GSLT concerns loads simplification. Simulating structure's behaviour and deformation in the ground, accurately, entails representing faithfully the loads with which the body undergoes deformation on air. This being impossible to strictly recreate and, in order to attain reasonable results, a load simplification procedure must necessarily be carried out. In this chapter, the preliminary analyses done for the scope of this project are discussed. The goal is to determine how to apply the sensorcraft's deformation and structural response on the ground, as it is whilst flying, or at least reasonably close. The main focus regarding the structural behaviour, is a $2.25$ g PUM which should be the most extreme case when talking about structural deformation for the flight happening in the future. The aircraft is undergoing complex and distributed aerodynamic and inertial loads, being the target to simplify and replicate those loads as representative concentrated loads. It is important to underline that meanwhile the studies were being performed, the FE framework was being constantly updated. Therefore, different versions will be presented along this part of the work.

The planning of the test is still a considerable amount of work for the entire thesis. In this chapter, the simplified representative load will be firstly discussed and calculated. Then, several candidates shall be analysed, considering every important aspect of what is happening on the ground, namely boundary conditions. The test rig which includes supports and fixtures is afterwards designed and built, according to the decisions previously made. Lastly, data acquisition system is decided and the final test procedure explained.
3.1 Simplified Representative Load

A HALE UAV structural demonstrator was developed and used in reference [23] where a static load test was carried out. The test rig was complex and intricate, something out of reach for the scope of this project, specially regarding the available time. Therefore globally, the test will have a similar, although more simplified, approach. The aerodynamic loads and moments are known from the ASWing interface, for the desired maneuver. These loads and moments are simplified, through an explained procedure, into representative concentrated loads that shall yield a similar structural behaviour.

As explained in section 2.1, here, the model is being fixed at the centre, i.e. where the wings connect in the computational model. These fixed conditions entail fixing it in every single Degree Of Freedom (DOF) of the six possible. Translation in $x$, $y$ and $z$ and rotation in $x$, $y$ and $z$.

As represented in figure 1.5 the appropriate aero loads are known from the integration between structural and aero models. A relatively expeditious FE analysis is enough to start providing possibilities and conclusions regarding loads simplification illustrated in figure 3.1. The method behind problem’s resolution consists in constraining the points where concentrated loads shall be desired to be applied. These constraints are applied in the FE model that is undergoing all the appropriate complete aero-dynamic and inertial loads already known, associated to the desired $2.25 \, g$ PUM, represented on the second step of figure 1.4. An analysis is carried out and the respective reaction forces retrieved. Using the same framework, yet with no aerodynamic or inertial forces applied to it, each of the reaction forces already calculated are applied at each of the respective nodes. Structure’s deformation and behaviour is expected to be similar between the two cases given the fact that the framework is undergoing the same loads, only being applied differently, in one case with higher detail, and in the other one as pseudo equivalent. Regarding constraints, they are based on simple supports which only block vertical translation, in $z$ axis, i.e. a simple pin in $z$ translation coordinate. Thus, collected reaction forces are clean vertical force vectors, with no moments. This fact causes the applying forces to have only a vertical component instead of being vectors with components along every axis, which would be of an extremely troublesome to even impossible accomplishment. Besides, only forces are being locked and the same is not happening with moments, which would be extremely difficult to apply. Potential errors would be possibly induced in the experimental test. Hence, there is a much easier load implementation with smaller probability of additional experimental errors.

Being the procedure known, the next step is the choice of the place where to apply the equivalent concentrated forces on the JWSC. There will be several different cases to which one might call candidates. Being candidates’ analysis a repetitive process, a MATLAB script was developed to automate the data acquisition and make it more understandable. The MATLAB script parses the data resultant from Nastran’s structural FE analyses, analysing fundamental data like strain, twist angles and displacements, arranging it into plots, comparing the data between the appropriate original aerodynamic loads and the pseudo equivalent concentrated ones. Therefore, differences between each candidate can be easily detected and decisions be made. Candidates’ analysis will assess which one is the best case scenario to acceptably represent the desired deformation.
The goal is to collect three different main quantities during FE analyses to follow. These three measured units will be accompanied later on by the same ones, being acquired experimentally. They are:

- **Displacement.** This is being collected across the entire structure in \( z \) direction, as this is the most relevant one in terms of deformation. It is the axis where the sensorcraft deforms the most and where differences to exist might be more relevant. Also, this quantity is the one that will be utilized as reference for the FE model update. It shall be represented by \( \Delta z \).

- **Strain.** Both longitudinal and transverse strains are considered but the biggest focus is given to longitudinal one, which will be the only one to be measured experimentally, on both the aft and forward spars. It shall be represented by \( \varepsilon_{xx} \).

- **Twist Angle.** Is being considered around the \( y \) axis, in the streamline direction. The angle being measured is equivalent to how much the angle of attack of the wings changes from a non-deformed position to the deformed one being considered. It is represented by \( \theta \).

### 3.2 Preliminary Candidates Analysis

Every single case next presented regards the investigation of the framework undergoing the pseudo equivalent load, as represented in figure 3.1, on the fourth step, with no gravity being considered here. Results next presented shall be the strain profiles along aft wing’s span and a graphical representation of the whole structure’s deformation. In this section and beyond, three different Relative Errors (RE) will be subject matter of major importance, therefore clarification \textit{a priori} is essential. Each and every error follows the same reasoning, since the aim is to determine the difference between the original and the pseudo equivalent load cases. They will be given by the following equation:

\[
RE = \frac{|X_{PE} - X_O|}{|X_O|}
\]  

(3.1)

Where, \( X \) represents the quantity being analysed, \( PE \) stands for pseudo equivalent and \( O \) for original. These errors are being calculated for every FE model’s node along aft wing span, and they will be...
given as a summary where \( \Delta z \) RE is given for the maximum value achieved across aft wing span. \( \varepsilon_{xx} \) RE and \( \theta \) RE are presented as average numbers for all the nodes determined across the aft wing.

Firstly, for the analysis of every candidate, the RE calculated are for displacement on the aft wing, streamwise twist angles and aft wing strain. It is important to emphasize where strain is being measured. Figure 2.2(c), in previous chapter, displays the cross section of aft wing’s spar and, taking into consideration how the FE model was conceived (plate elements), we are able to measure strain in four different places in the cross section. Therefore for this set of results, strain is being measured on the top surface of the spar, on leading edge’s protuberance, i.e. point number 1. As previously explained, this first set of studies aims for a global overview of the candidates as well as to give a validation for the simplifying concept.

We shall start with the simplest case to be physically accomplished: loads being applied at each wing tip, shown in figure 3.2, to which we call candidate 1.

As shown herein in the following figures, when running analyses for both the structure undergoing the original aero loads and the pseudo equivalent ones, there is a significant difference between both deformed frameworks. We can easily detect the differences, thus results for displacements and strains are expected to be considerably off desired ones, being the errors around 14\% and 12\% for twist and strain respectively and there is a maximum error of around 8\% for displacement. These statements are supported by the data presented in figure 3.3. The strain is given along the non-dimensional span of the aft wing. Regarding structure’s deformation, the difference between original distributed (light green) and
pseudo equivalent (dark green) is observable. Also a first glance at the difference between linear (blue colours) and non-linear (green colours) solution methods might be taken.

![Whole Structure's Deformation](image)

(a) Candidate 1 structure's deformation

![Strain vs. Aft Wing's Length](image)

(b) Candidate 1 strain profile

Figure 3.3: Candidate 1 structure's deformation and strain profile along aft wing’s span.

In the next case we introduce candidate 2. Relatively to the previous one, a load is appended at the wings junction (where forward and aft wings attach to each other). Results change significantly, and a considerable improvement is achieved, being now the errors around 2% and 9% for twist and
strain respectively and there is a maximum error of around $1\%$ for displacement. This enhancement was tremendously unexpected given the fact that, using only four concentrated loads there is the ability to match quite well structure’s response to the one undergoing the original loads, exception made for strain, where an enhancement is achieved, but still with a high figure for the error. From second candidate onwards, results become steadier being the differences between each candidate not so easily recognizable. Therefore, although all candidates were analysed, not every single plot will be presented given its similarity between one another. Only plots for candidates 1, 2 and 8 are being shown.

Candidate 3 is an upgrade of the previous one. It has an additional load at the aft wing. This load is not applied randomly. $\Delta z$ RE along aft wing’s span was calculated and, for candidate 2, there is a peak where the value for the error is significantly larger than the rest, as can be seen in figures 3.12 and 3.13, later on. Taking this fact into consideration, the aim is to try to reduce this peak in the error plot, constraining that exact same point where it occurs. However, results for candidate 3, namely displacement error, show that placing the load at that critical point/area does not influence results, providing identical ones, with a similar peak at the same place. Errors are around $2\%$ and $10\%$ for $\theta$ and $\varepsilon_{xx}$ respectively and there is a maximum error of around $1\%$ for $\Delta z$.

A slightly different approach is carried out for candidate 4. Reference case still relies on candidate 2. Furthermore, two loads are placed on aft wing and one load on the inner clamp of forward wing’s spar. The inner clamp is the fixture for the forward spar to the wing’s rigid part. Instead of the criterion with which the load is placed for candidate 3, both loads applied on rear wing in this case, do not necessarily have a precise point where they should be. Loads are placed so that they can be equally spaced. Results are similar to the previous candidate, being errors around $2\%$ and $10\%$ for twist and strain respectively and there is a maximum error of around $1\%$ for displacement.

Although the point is to simplify all the complex and distributed loads that the sensorcraft is undergoing, there is the possibility of trying to reproduce them with as many application points as possible. The next cases presented are feasible and possible scenarios, however they lead to a more troublesome GSLT execution. Such examples are candidates 5, 7 and 8. The best achievable way to try to reproduce distributed loads (if we want to go deeper), in this case, is to apply loads at each of the sensorcraft’s structural ribs. Hence we have candidate 5 which has loads at each of the aft wing’s ribs, at forward wing tip and at the interface between both wings. Being the rear wing the biggest point of interest not only for the scope of this work but also for the whole JWSC project, it may be interesting to “constrain” each and every rib of the aft wing and conclude if this influences results, namely errors for displacement and strain. Whilst the main area of focus is the aft wing, there is the awareness that the majority of aerodynamic loads is being applied to the forward wing because this is the one with the biggest role aerodynamically wise. Lastly, candidate 6 is simply a different approach for candidate 2. It was implemented to investigate how important it is to have a load applied at the interface between aft and forward wing.

Results surprisingly show that two couples of loads, well placed, is enough to replicate a similar behaviour to the one observed with distributed loads. These preliminary analyses allow us to exclude, right away, some candidates which either do not provide acceptable results or are too much intricate to
be carried out. A quantitative summary is next presented, regarding every single option.

As presented in figure 3.6 we can immediately see how candidate 1 is not a satisfactory approach. For $\Delta z$ and $\theta$ RE, from candidate 2 onwards, results tend to be considerably similar, being the main difference between each candidate the RE for the average $\varepsilon_{xx}$ value along aft wing’s span. Also, candidates 2, 3 and 4 present similar results so we can conclude that adding some load points at the aft wing
Figure 3.5: Candidate 8 structure's deformation and strain profile along aft wing's span.

Regarding strain, results are considerably better as the number of loads increases, however the inherent complexity of applying many concentrated loads implies a trade-off between intricacy of application and consequent achieved results.

Candidate 1 was the case study to begin with, being the simplest one to carry out on ground. Results
were not satisfactory. Candidate 2 allows a big improvement in both displacements and twist angle, being it not as significant for strain. Results stagnate and remain similar for candidates 3 and 4. A small improvement is achieved for candidates 5, 6 and 7. Candidate 8 represents the best case scenario, however with the highest associated complexity.

The adopted procedure is valid based on the collected results, i.e. for candidate 8 for example, structural behaviour is indeed similar to the appropriate distributed loads. The plots previously shown demonstrate a clear difference between linear and non-linear solution methods which means that a NL solver has necessarily to be used. The next step is to choose a final candidate, the one that will be used to carry out the GSLT and to determine the final loads and boundary conditions to be used.

Next presented, in figure 3.7 is a plot that shows how much the three previous different errors are influenced by the number of loads applied on the framework.
3.3 Considering Gravity

Initial load’s simplifying procedure was successfully validated, i.e. the results derived from the used method were assessed, and turned out to be satisfactory. However the concentrated loads previously calculated, that yield a similar behaviour of the aircraft to the original distributed aerodynamic and inertial loads, are not the correct ones to be used on the GSLT. This is due to the fact that the structure on step 4 of figure 3.1 is only undergoing the simplified concentrated load and no static gravity is being considered. This is illustrated in the following figure 3.8, where the approach previously done is represented on the left side, and the reality of the GSLT (where the sensorcraft statically undergoes gravity) is shown on the right side. Therefore and according to figure 3.8, the true load to apply in the GSLT, represented on the right side must be determined. Equation 3.2 shows the relation between the true load to be applied and the one that was previously calculated.

\[ \text{GSLT Concentrated Load} = \text{Equivalent Concentrated Load} + \text{Gravity} \] \hspace{1cm} (3.2)

Figure 3.8: Difference between first approach and actual GSLT.

From now on, all conditions of the GSLT must be considered and carefully analysed. Large deflections are expected to happen, structure’s response must be considered as a NL behaviour. Two different ways of considering gravity into the simplification procedure will be presented. If the framework is undergoing only gravity, we may consider small deflections and thus accounting for linear simplifications, which will be the first approach. Second one shall be more careful when talking about non-linearities. Therefore, and to take gravity into consideration for the calculations of the final forces to apply, two different paths might be taken.

- **1st approach.** According to equation 3.2, the GSLT applied load can be found by subtracting gravity to the previously pseudo equivalent calculated load. However this can not be done directly because a distributed load can not be subtracted to a concentrated force. Therefore, the solution for this is to approximate gravity to a concentrated force as well, having then the possibility to carry out the linear subtraction. This is illustrated in figure 3.9 and equation 3.3.

- **2nd approach.** Although the approximation done previously must be true, because when undergoing only gravity the structure still behaves linearly, it is required that this is further investigated.
Therefore in this case, the GSLT applied load is calculated based on the diagram shown in figure 3.10. To the inertial loads known during the PUM, 1 g equivalent to the gravity undergone by the aircraft during the GSLT, is subtracted. Hence, using the same procedure as initially, simplifying the original aerodynamic and the inertial loads with 1 g less, it is possible to directly obtain the concentrated load to be applied in the GSLT.

\[ E_{\text{Equivalent Concentrated Load}} = GSLT_{\text{Concentrated Load}} - Gravity_{\text{Concentrated Force}} \]  

(3.3)

Simplification of gravity on first approach may induce additional errors and imprecisions, because linear relation 3.3 between loads may not be valid for larger deformations, given structure’s NL behaviour. However, both approaches provide similar results. Displacements are small when framework undergoes only gravity, meaning that it still behaves linearly, i.e. for both methods previously introduced, the GSLT applied load turns out to be similar.

Candidate 2 is the chosen one to test each one of previously presented approaches. Table 3.1 shows loads obtained for the pseudo equivalent cases. Loads are similar for each application point, to each one of the processes. Therefore the aircraft behaves similarly for both approaches and we can conclude that, in this case, non-linearities are not present. This means that when the sensorcraft is undergoing only gravity, it might still be considered linear. Furthermore, future analyses considering gravity will be done using the second approach.
3.4 Final Candidates Analysis

Final candidates are chosen and should be either candidate 2, which is the simplest one that can provide good results, or candidate 8 that clearly provides the best outcome, however with an inherent troublesome execution. Final analyses taking into consideration all the aspects like boundary conditions will be carried out so that a final decision may be taken regarding the candidate to be used for the static load test. Knowing that gravity is a major factor during the GSLT, there are two possible different ways of implementing the test. Since there is an optical table and a threaded ceiling, both options are valid to clamp the sensorcraft to, knowing that in the ceiling it would be turned upside down.

The structure is highly flexible, thus it deforms considerably when statically undergoing only gravity. The aircraft may be attached to the optical table, having in this case, gravity downwards relatively to it and loads applied in conformity, *i.e.* upwards, meaning that deformation would occur by means of only concentrated forces. The other feasible option is having the sensorcraft attached to the ceiling (being gravity upwards relatively to sensorcraft's referential). Theoretically, second case shall be the best one because, in flight, aircraft's deformation occurs upwards in aircraft referential. If we have it being held to the ceiling, gravity (which is distributed along all the structure) would contribute for deformation as a distributed load and therefore, possibly adding precision to the additional concentrated forces being applied, in the positive flight load direction. Hence, it is possible to say that a small amount of deformation would be caused by a distributed load (gravity) and everything else by concentrated loads. Therefore, this suggests that clamping the aircraft to ceiling would imply better results. Beyond this, loads would be applied downwards, by gravity, *i.e.* instead of turnbuckles, bags with masses shall be used, allowing better precision when applying forces. Also, loads have necessarily lower magnitude comparing to the case where aircraft is sitting on the table. As follows, FE analyses are being carried out to evaluate results for both scenarios and conclude if this statement was right.

Next presented, in figure 3.11, is a plot that demonstrates results (namely errors) for both candidates with two different configurations (upside up clamped to table or upside down secured to ceiling). Regarding the dots shown on the image in question, their colours represent the directions according to which loads are applied, respectively to the referential, exterior to the sensorcraft. Yellow dots represent loads applied upwards, *i.e.* opposite direction of gravity, while blue dots mean loads applied downwards, meaning in the same direction as gravity. As might be seen, both situations are illustrated and further conclusions are straightforward.

For candidate 2 being fixed to the optical table, all loads would have to be applied upwards, which

### Table 3.1: Loads figures for two different approaches.

<table>
<thead>
<tr>
<th>Locations</th>
<th>1st Approach Load [N]</th>
<th>2nd Approach Load [N]</th>
</tr>
</thead>
<tbody>
<tr>
<td>Forward and Aft Wings Junction (Right Side)</td>
<td>80.465</td>
<td>80.431</td>
</tr>
<tr>
<td>Forward Wing Tip (Right Side)</td>
<td>0.972</td>
<td>0.988</td>
</tr>
<tr>
<td>Forward and Aft Wings Junction (Left Side)</td>
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<td>80.428</td>
</tr>
<tr>
<td>Forward Wing Tip (Left Side)</td>
<td>0.973</td>
<td>0.989</td>
</tr>
</tbody>
</table>
means with turnbuckles and load cells, and that does not allow as much precision as masses. Additionally, turnbuckles are of considerable troublesome execution when several are applied at the same time, because when load is implemented in one side, others tend to change and it becomes a slow trial error process. For the same candidate but instead clamped to the ceiling, one pair of loads would be implemented with masses (downwards) and the other pair with turnbuckles, being this a substantial improvement in precision and ease. For candidate 8 sitting on the table, the majority of loads would have to be taken with turnbuckles and some with masses which would be almost impossible to achieve given the fact that there’s no room enough between the bottom surfaces of the sensorcraft and the table. For the last situation, most loads are applied with weights and others with turnbuckles. Aside from the previous arguments, it is important to analyse results for each one and consider the related errors. The argument presented previously is confirmed. It is detectable how results improve by simply inverting the sensorcraft. For candidate 8, there is slight to almost no difference and that fact is due to loads being applied with a high number of application points and, therefore almost simulating a distributed load case, minimizing the discrepancy between both settings. Resuming, good results are achieved for both candidates when turned upside down. Hence, and taking into account that the difference between the two is short, especially for displacement and twist angle, candidate 2 shall be the chosen one to be used in the GSLT. This solution can provide a considerably good approximation to the 2.25 g maneuver’s deformation with a significant simplicity.

Detailed results are presented in figures 3.12 and 3.13 which illustrates the case where aircraft is attached upside up to the table, and in figures 3.14 and 3.15 which represents the scenario where the sensorcraft is clamped upside down to the ceiling. In the plots it is possible to analyse aft wing’s $\Delta z$ and the respective RE along its span. Regarding the upside up case, the difference between candidate 2 and candidate 8 is considerable, whereas in the upside down case plots are similar when comparing candidate 2 against candidate 8. The fourth set of plots represents $\theta$. Here, candidate 2 has a slightly larger error for the pseudo equivalent load case comparing to candidate 8, where results match easier, specially in the upside down case. Lastly, final plots represent aft wing’s bending strain bridge, where ribs and measurement points are represented. Knowing from figure 2.2 how aft spar’s cross section is,
strain gages will be placed in pairs, on the leading edge protrusion, one on top surface and the other on bottom one. Thus, we shall have a value for bending strain at those locations. Hence, between each rib (represented by vertical black lines) there are two pairs of strain gages, evenly spaced between each other, thus ten bending strain values along the spar are retrieved. **This means that from now on, strain** \( \varepsilon_{xx} \) **will always be referred as bending strain.** Therefore a strain bridge plot was formed with FE data, with strains being taken on the exact same place where they will be measured on the test, showing what layout should be expected for the experimental data. Regarding plots, legend is presented for first ones, but it is also valid for the ones below. It is almost irrelevant the position in which candidate 8 is tested, given its high number of concentrated loads. The same is not true for candidate 2 where results become significantly better when it is turned upside down, clamped to the ceiling.

Loads to be applied on GSLT may be retrieved from FE analysis and it is recognizable that the highest forces in magnitude are the ones on the inner application points, which are handled with masses. The outer loads are applied upwards with turnbuckles, however their magnitude is extremely small and almost impossible to determine their exact value using the load cells available. This means that these loads are negligible when comparing to the ones applied at the wings junctions. There is, then, the possibility of simplifying even more the forces application by reducing the applied loads to only one at each wing. As a result, and analysing this hypothesis, errors are found to be identical to the ones belonging to candidate 2, not being more than 1% greater. **Recapitulating, ground static load test will be carried out using candidate 2 with no loads applied at the wing tips, because its simplicity outweighs results obtained which are indeed, acceptable.** Also, final loads were calculated and they will be 5.659 kg on each wing.
Figure 3.12: Candidates results I - upside up configuration.
Figure 3.13: Candidates results II - upside up configuration.
Figure 3.14: Candidates results I - upside down configuration.
Figure 3.15: Candidates results II - upside down configuration.
3.5 Static Rig Design

3.5.1 Sensorcraft Fixture

The initial idea was to have the sensorcraft hanged on the ceiling by means of some device that could support it by the hole used for the launcher. The gear to implement this had to be designed from scratch. In figure 3.16 it is possible to see the first idea applicable for the support rig. There are two different parts that attach to each other. In white, the hook that goes into the hole used for a similar hook in order to launch the aircraft into flight. In grey, the support in order to hang the set to the ceiling. This was made with the aim of supporting the sensorcraft exactly where it is being constrained on the computational model, and therefore having similar BC as explained in section 2.1.

![Figure 3.16: Hook and support of initially designed support rig (SolidWorks render).](image)

Although it seemed initially a good idea, it eventually turned out to be inadequate. Being the sensorcraft considerably heavy and highly expensive, it might be unsafe to simply hold it by one single point. This point is located in the main structural rib of the fuselage and despite its stiffness, it is still a risk not worth being taken. Besides, giving the fact that the CG is behind this point of interest, there will be weight exerting stresses on the area where there is only a thin composite skin. Also, there is considerable weight exerting a demanding moment at the point where the sensorcraft was supposed to be held, originating additional stresses in the hook and the fuselage rib. Therefore, this idea was put aside and other kind of support rig was chosen. More conservative in terms of safety, being the biggest problem the fact that in terms of BC it might not be the best solution.

The new rig shall consist of a support that holds the aircraft in four points. Two of these points being in the interface between fuselage and the right wing, with one of them ahead and the other further back. The two other points follow the same reasoning but instead, for the left wing, as represented in figure 3.17. This being said, besides the changes in the framework itself for the new set of results, there will be additional changes to boundary conditions, so that they can match to what will happen in the test, which are represented in figure 3.19. Taking this fact into account, further investigation must be carried out, on how this fact may or may not influence results, originating new methods and approaches.

In figure 3.18 it is possible to see the difference between the several different boundary conditions. In this particular case, the error refers to the quantity for the pseudo simplified equivalent case with
the shown BC, relatively to the same quantity for the distributed load case with the original boundary conditions, which is the sensorcraft being fixed on the hook’s hole. Thereupon, the new BC should be equivalent to what was considered previously, as shown in figure 3.18.

In short, we shall now have fixed constraint conditions in the beginning of each wing, for both the test and FE analyses, being the biggest difference regarding flight. To make this assumption we must be sure that there is slight to almost no difference between both conditions. The fuselage is considered to be almost perfectly rigid, which means that having a fixed constraint where wings start should be identical to flight conditions. It is important to state that the interface between fuselage and wings (where new BC are being applied) is still far from the point where wing starts to be flexible. This can be seen in figure 1.3 and 3.19, where green dots represent the fixed conditions equivalent to reality.
3.5.2 Load Fixtures

Forces need to be applied to the sensorcraft and a non intrusive method is desirable. This might be achieved using a system that can be attached to the surface where forces should be applied, and easily removed without leaving trace behind or damage to the structure. Holes on structure or surfaces are not wanted, therefore the fixture shall be based on some kind of tape, which is either easily attached and removed. Therefore, loads have to be applied with some device that can be attached to tape’s surface easily. After some thoughtful consideration, the ideal situation would be to have a small sample from an ‘L’ aluminium extrusion available, cut into $\frac{1}{2}$ inch long pieces. Bottom surface of the sample is glued to the aluminium tape with epoxy. In order to successfully test this set-up and to decide their ideal sizes, to safely apply the desired forces, specimens of a sandwich carbon fibre panel were used. These specimen’s surface is similar to the one of the aircraft’s body. Hence, by applying the same paint finish used for the sensorcraft, the final surface of specimen shall be the same as the one on the aircraft. Conditions will then, be the same, and results trustworthy. Materials and used processes are shown in figures 3.20 and 3.21, respectively.

Knowing from preliminary analyses done in chapter 3.4 that the highest load magnitude will be between 5 and 6 kg tests must be carried out to take conclusions, or to make possible improvements, if load is not being supported with a minimum safety factor margin. The tested device must support at least 10 kg so that a minimum safety factor of almost 2 may be achieved. Specimen will be clamped to optical table and load will be applied vertically using a turnbuckle with a load cell as illustrated in figure 3.21. Load is applied with steps of 1 kg and between each step, one minute is taken to wait and verify if load value presented on load cell holds steady, which means that it is supporting the load, without starting to fail. Results and conclusions for the several tests carried out are presented as follows:

Test #1: anchor and dry epoxy detached from aluminium tape when the load applied was 4 kg. Analysing the bottom surface of the anchor, it is apparent that the epoxy glued well to the anchor but not as good to the aluminium tape’s surface. It kept a smooth surface leaving no epoxy residue on the tape. After some consideration, for the following test, it was decided to sand tape’s surface so that the contact area could increase, having then a stronger connection between epoxy and aluminium tape.

Test #2: aluminium tape started to detach from specimen’s surface, in one specific side, with 6 kg
(a) To the specimen is applied paint

(b) To the painted specimen is applied aluminium tape

(c) Epoxy is administered to painted and taped specimen

(d) Final test specimen

(e) Test set-up

Figure 3.21: Process used to test the loads fixture.

being applied. This means that the previous improvement (sanding tape’s surface) yielded useful and effective results. A larger force is now sustained but a problem arose elsewhere. Tape is not detaching evenly, being clear that it is unsticking earlier from one side. The reason behind this is the fact that the anchor has an ‘L’ shape and, consequently, the load is being applied unevenly to the tape’s surface and not exactly in the centre, as it should be, to achieve an ideal distribution of force per area. Accordingly, there will be a pressure exerted in one side of the tape’s surface that reaches the edge sooner, resulting in clear detachment, on that specific side where vertical surface of the anchor is. For better understanding, failure can be seen in figure 3.22(a). Consequently, in order to avoid the problem observed, the next solution is to use an anchor with a ‘T’ shape instead. This solution will be tested as follows.

(a) Test #2 failure.

(b) Test #3 failure.

Figure 3.22: Anchor test failures.

Test #3: There was an enormous improvement because the load started to drop when 12 kg were
being applied. The tape started to detach homogeneously in the area underneath the anchor, and it started to tear itself apart. At the same time, the anchor started to detach from the tape. This means the whole set, for this configuration, does not support loads higher than mentioned above, for the sizes used.

Hence, it is known that the suggested idea supports loads up to $12\, kg$ but for short periods of time. For the GSLT, loads will be applied for a considerable amount of time, enough so that all quantities may be measured and pictures taken. Given the fact that the highest magnitude of load shall be between $5\, kg$ and $6\, kg$, there is need to improve the strength of the fixture, therefore the best solution is to increase anchor’s and tape’s sizes. This will allow a higher magnitude of force to be applied, knowing that the problem in question is the pressure that the set withstands and, increasing the area should allow an equal increase in the exerted force without compromising the fixture. Even though doubling the area might be enough, when loads are applied, the time required to take measurements might be considerable. Thus, the final decision was to have ‘T’ extrusion pieces with $1\times2$ inch bottom surface, with a $2\times4$ inch tape underneath. The final fixture is shown in figure 3.23.

![Figure 3.23: Load fixture.](image)

(a) Zoom out  
(b) Zoom in

3.6 Data Acquisition System

As part of the test planning, data acquisition system is a critical subject to be discussed. The same quantities referred in previous preliminary analyses are being measured in the GSLT i.e. $\Delta z$, $\varepsilon_{xx}$ and $\theta$. For the experimental results a MATLAB script, that parses all the data and compares it with FE analysis data, was developed.

Regarding displacements, they were measured by determining the coordinates of certain a priori chosen points, using a coordinate measuring machine (CMM). From what was previously done by other researchers and as referred in section 1.4 the maximum precision achieved was around $0.04\, mm$ and the minimum around $0.25\, mm$. With the chosen CMM model, the precision annunciated by the manufacturer is of $0.1\, mm$, which is somewhere between both precisions already achieved with photogrammetry work. The model to be used is Creaform’s GO! Scan 3D. The process behind the use of this machine is the
following:

1. Dots to be measured are highlighted on sensorcraft’s surface, with fiducial markers.

2. Structure is scanned at the desired position. Hand scanner is moved around the structure so that the whole surface gets recognized into the system, being registered as the scan goes, and saved in the computer, as represented in figure 3.24(d). This machine is manually controlled, with an easily portable hand-held 3D scanner, which captures geometry, colours and targets relatively to a predefined coordinate reference system, using the emission of white light (LED).

3. Registered surface is saved and fiducial markers are verified within the acquired data.

4. Data is converted into desired format, which is a `.unv` file with all the desired nodes and their respective \( xyz \) coordinates with respect to the origin. The origin is the centre of the coordinate system represented in figure 3.25.

![Figure 3.24: Coordinate measuring machine.](image)

Using a simple tool, a yet acceptable precision is achieved for displacements measurements (each measurement of a complete structure takes about 10 minutes to be successfully carried out). The auxiliary coordinate reference system was designed and 3D printed to be mounted into the sensorcraft’s nose where engine was removed, allowing therefore the measured points to have the same coordinates
as the FE model, avoiding thereby additional time expenditure by making the coordinates agreeable between each other. The scanner scans the coordinate system, and the target dots are given relatively to the centre of the reference coordinate system, which might be seen in figure 3.25. It was modelled so that it could be accurately attached to the engine mount, without being moved. Any type of movement or rotation in the coordinate system might mean errors introduced in the coordinates being measured. This is the reason why it was modelled this way and mounted into the engine fixture. Measurements are being taken in certain points that where chosen based on having a direct equivalent node in the FE structural framework, as shall be presented in figure 3.28.

Figure 3.25: 3D printed coordinate system.

In this specific case, CMM will identify predefined targets which will be highlighted with fiducial markers on top of the sensorcraft’s surface. The dots location was chosen based on the criteria that they should be well distributed, along the sensorcraft, that might be then sufficiently represented if the dots are plotted in a 3D graph. Also, they must have a simple correlation with FE model’s nodes. Therefore on the essential parts, as the flexible parts of the wings (spars), dots will be placed over every single rib whose location is easy to know both in the sensorcraft and the FE model. Auxiliary further back dots were used, as will be explained afterwards, to measure twist angle. Also, other points were placed on rigid parts like the wing tip, boom and fuselage in order to have a global shape of the aircraft. The placement of the dots along the sensorcraft was a demanding task given the precision with which they should be applied. A short description, with pictures may be seen in figure 3.26. Also, dots’ distribution and their numeration is presented in figure 3.28.

Strains were measured using conventional foil strain gages, made from a proprietary alloy encapsulated in kapton. They were placed on top and bottom surfaces of each spar to collect a bending strain, as explained previously in section 3.4. The aft spar has 10 measurement points on the leading edge and 2 on the trailing edge (which are not being used) and the forward spar has 3, on the centre. Strain gages’ location may be seen in figure 3.27. On aft spar’s leading edge there are 2 evenly spaced gages between every rib. Knowing that there are 6 ribs and strains are being measured only between ribs, there are then 10 useful strain gages. Regarding forward wing, their location is represented in figure 3.27(b).
(a) Materials used: masking tape, scissors, calliper, ruler, pen, hole punch and vinyl tape

(b) The spar is highlighted with masking tape which has a centreline drawn on it

(c) The spar tape is outlined with a thinner green masking tape. Also the trailing edge is highlighted with thin masking tape

(d) Blue tape is removed

(e) Ribs are marked on each panel. Blue tape is cut into small pieces and placed between the two green lines close to a rib location. Orange vinyl dots are placed in the intersection between spar centreline and rib line

(f) Auxiliary tape is removed and main points to be measured are successfully marked

(g) Tape is placed right behind main dots according to wing span. A sheet of paper is cut with the same angle as the trailing edge has relatively to streamline (51.5°). Streamline is then drawn on the tape

(h) Twist measurement dots are placed using thinner tape pieces

(i) Application of points is finished and the process is repeated for the remaining wings and other parts

Figure 3.26: Application of displacement measurement dots.

Twist angle was measured based on displacement measurements. The main points along the wings whose displacements are being measured have further back, auxiliary dots (on the same streamline)
Figure 3.27: Strain gages location and implementation.

so that twist angle on those main points might be calculated. Knowing the coordinates for both points
(the main one and the auxiliary one further back), $\theta$ is automatically calculated in the MATLAB script.
Figure 3.28 demonstrates the implementation of those auxiliary dots. Twist was measured on 3 points
along each forward wing, which are the 3 central ribs. As far as the aft wing is concerned, twist is
being measured on every rib, so 6 points on each wing. As an alternative approach, for future work to
be developed, an optical device, which can be seen in figure 3.29, was designed and 3D printed and
then mounted on the bottom surface of each aft wing. The twist might then be measured using the
video recorded during the GSLT, by the cameras shown in figure 3.30. Two cameras are used to record
images for each wing, one from the top surface of the fuselage and the other one from the bottom.

Figure 3.28: Displacement measurement dots.

Figure 3.29: Optical devices used for future alternative $\theta$ measurements.
3.7 Linear Benchmark

As explained previously, the main objective is to update the FE model based on linear deformations and using a linear solution method. The biggest reason behind that is the fact that a linear update is quite less time consuming than doing it by means of a non-linear solver and deformations. Therefore, there is need to set a benchmark such that the FE model can be updated to represent the as-built aircraft. In this section the load factors, at which linear and non-linear analysis roughly match, are found. Plots for the twist angle and displacement dependant on the load factor applied were created, where ‘1’ represents the maximum load case which is equivalent to the 2.25 g PUM. The angle is being measured roughly half way through aft wing’s span, for both quantities being measured.

By the previous plot, the first idea is that up to a load factor of 10% of the maximum load, the trends for displacement and twist are similar between linear and non-linear solutions. However this is still not clear enough because these quantities are being measured midway through aft wing’s span. To get a full idea of the linearity of the structure under these conditions the displacement on the wing tip must be observed and compared to the reference wing span. Since the sensorcraft is still not available at this time, the estimations have to be done based on FE simulations. The Linear Benchmark Factor (LBF) is calculated for the following load cases, knowing that half-wing span is 1.5 m and that the displacement
is measured at the wing tip:

\[ LBF = \frac{\Delta z_{wt}}{1.5} \]  

(3.4)

- **Sensorcraft undergoing 10% of the maximum load.** From the FE analysis, it is known that at the wing tip \( \Delta z \approx 0.097 \) m, therefore \( LBF = \frac{0.097}{1.5} \approx 6.4\%. \)

- **Sensorcraft undergoing 5% of the maximum load.** From the FE analysis, it is known that at the wing tip \( \Delta z \approx 0.071 \) m, therefore \( LBF = \frac{0.071}{1.5} \approx 4.7\%. \)

As calculated, for the 10% load factor to be applied to the sensorcraft, the validity of linearity is not safe to be assumed as such. Hence, **sensorcraft’s deformation when undergoing 5% load factor might still be considered linear**, considering that the displacement up to 5% of the reference wing span (LBF previously calculated) may be considered negligible. Thus, a linear solver may then be used to analyse the FE model, in those circumstances, and also to update it based on smaller deflections originated from small load factors.

### 3.8 Test Procedure

Given the highly non-linear response of the structure, the fact that it will be updated based on linear deformations, and also the troublesome model updating procedure, the GSLT and consequent model update will be divided into easier and simpler steps, which are being explained as follows. As will be explained later, the update was plainly based on changing the EM of elements, which were retrieved by "Inverse Problem". Also, before Young modulus, masses and inertias were updated.

1. Spars are individually statically tested, like a cantilever beam, with a wide range of loads.

2. \( \Delta z \) from one specific load case (within linear range) are measured and introduced in the FE model.
3. FE model is updated based on measured $\Delta z$ and new EM is retrieved. Forward spar new Young’s modulus is introduced in the tailless and complete sensorcraft models. Rear spar new Young’s modulus is introduced in the complete sensorcraft FE model.

4. Tailless sensorcraft model is statically tested, with a wide range of loads.

5. $\Delta z$ from one specific linear load case are measured and introduced in the FE model.

6. FE model is updated based on measured $\Delta z$ and new EM are retrieved. They are also introduced in the sensorcraft FE model.

7. Sensorcraft model is statically tested, with fractional loads, being the 5,659 kg equivalent to the 2.25 g PUM.

8. $\Delta z$ from one specific linear load case are measured and introduced in the FE model.

9. FE model is updated based on measured $\Delta z$ and final EM are retrieved.
The most efficient process to define important features and characteristics of structures is experimental testing. The drawback is the fact that it still is, even nowadays, a costly and time-consuming method. However it is still a necessity, if reliable high-end structures are to be built and its computational models to be updated and validated. Experimental testing can be done in several different ways, delivering diverse results, namely measured quantities. The GSLT to be carried out required a considerable amount of preparation, where the load case was decided and the instrumentation prepared.

In this chapter, the experimental static ground tests are presented. The spar test, the tailless model test and the complete sensorcraft test are described.
4.1 Spars Static Load Testing

First, the cantilevered spar tests were carried out to better characterize the response of the spars themselves, and to confirm that the sensorcraft is able withstand the operational 2.25 g pull-up maneuver. This test also allows to determine the modulus of elasticity for the computational model and to use it in the full sensorcraft finite element model. Knowing that the flexible spars exhibit large deflections, they will exhibit a non-linear behaviour. Therefore it was desirable to have a first insight on the non-linear solver chosen for this purpose and for the whole project. FE models were designed for both aft and forward spars, also to get an approximation for the yield strength. Both computational models are created with 1D CBEAM elements. Linear and non-linear analyses were carried out for both cases so that both can be compared. Regarding boundary conditions, spars behave like cantilever beams so, ideally, in FE model, they are clamped in one end, \textit{i.e.} fixed both in displacement and rotation at one of the tips. Nevertheless, this constraint in a computational model is always utmost, \textit{i.e.} in real life situations, the clamping system is never totally rigid. This being said, there is the awareness that boundary conditions will be one possible target where model can be updated, as well as for the elasticity modulus, so that it matches the same displacements measured experimentally.

Displacements will be the only quantity being measured. Three equally spaced vinyl dots were placed across spar’s span, and their vertical coordinates at each load step were measured.

![Forward wing spar](image1.png) ![Aft wing spar](image2.png)

Figure 4.1: Spars static tests.

Each spar was tested with incremental load steps. The initial idea was to have for both cases, spars clamped to the optical table. However, the available height between table and ceiling is considerably short and for the aft spar exclusively (which is the more flexible one and consequently with bigger deflections) it was impossible to implement those displacements with the available turnbuckles. Therefore, aft spar is being tested from the ceiling, being masses applied, whereas forward spar, on the other hand, is locked to optical table and loads are applied upwards with turnbuckles. Vertical displacements are being measured with a calliper in three different points along spar’s span, which were highlighted on their
surface with orange vinyl markers. The materials and processes used are described as follows:

- **5 1/4 inch screws.** Used to clamp spars to table/ceiling.

- **1 calliper.** Used to take measurements with a reasonable precision, to correctly place the orange vinyl markers along spar’s span. These markers have a correlating node in the FE model. Calliper is also used to measure the desired displacements.

- **1 ruler.** It is used to measure displacement of the aft spar when it starts deforming beyond the maximum allowed for the calliper to measure.

- **3 kg of led small spheres.** Led is used as mass to apply the desired forces. Being it in small spheres becomes helpful when making the several bags with different weights, that are recreating the load steps to be used.

- **1 plastic bag.** Used to put the led inside, making therefore a mass bag, for the aft spar test.

- **1 load cell and turnbuckle.** It is necessary for the forward spar test, as the loads are applied upwards. The load is manually exerted by tightening up the turnbuckle and the load cell gives its magnitude.

- **2 chains.** It covers the distance between the spar and the turnbuckle.

- **2 eye bolts.** The eye bolts are attached to the end of each spar, where load shall be applied.

- **1 scale.** The scale is used to correctly measure the mass bags.

- **1 photographic camera with tripod.** A tripod and photographic camera are placed and not moved throughout the tests. Pictures are taken for every load step, so that an overlay might be created and presented, as might be seen ahead.

Spar’s clamps were designed from scratch in order to behave similarly to the ones used in the sensorcraft and can be seen in figure 4.1, as well as configuration used. Holes were made in the other end in the exact same place where the wing tip is attached. An eye bolt is placed in the hole and loads can be applied safely.

Results are presented in figures 4.2 and 4.3 where it is observable the deformation at each load step applied and a plot for displacement at the farthest lengthwise measurement point for each applied load. As it is clearly observable, experimental data is unquestionably close to the one retrieved from the FE non-linear solver. The predictable non-linearity for the behaviour, given the large deflections, is confirmed in the spars themselves, and the FE NL solver keeps up well with reality. Therefore we can say that the solution method number SOL 400 is a wise choice for geometrically non-linear problems.

The loads at which the spars start to deform plastically were also determined, measuring after each load step, the coordinates of the measurement dots for the unloaded case. The forward spar started to plastically deform when 12 kg where applied. Regarding the rear spar, 3.5 kg where being applied when it started to plastically deform.
4.2 Sensorcraft Static Load Testing

As soon as the sensorcraft became available, there was the need to tweak some minor properties so that the FE model could accurately represent the as-built aircraft, such as inertias and mass locations. The FE model consists in the structural elements of the real aircraft, such as the spars, and mass points that represent the panels and related parts of the outside body. The first step was to re-update the location of those mass points in the FE model. The second step was to carry out a bifilar pendulum (BFP) test, to retrieve inertias and update them in the FE model. Figure 4.4 shows two pictures from the BFP test.

As previously explained, sensorcraft GSLT will be divided into two different stages. Statically load the sensorcraft with no aft wing and update the respective computational model, repeating then the process for the complete configuration instead, having therefore a more systematic and possibly simplified process. Linear range was already determined considering that the model will be updated based on its
linear range of deformation, being it one of the main purposes of this work: assessing the feasibility of updating a highly non-linear structure using linear range of loads and a linear solution method as well.

As calculated in section 3.4, loads are determined and will be applied downwards with mass bags. The maximum and reference deformation will be induced with the previously calculated loads. Apart from this, loads will be applied in load steps as a percentage of the maximum one. Having coordinates for the ‘non deformed structure’ would be ideal, so that references could be established. The issue is that the aircraft is highly flexible and, therefore only gravity is enough to make it deflect considerably. The available option is to proceed with one measurement of coordinates when the sensorcraft is upside down.
on the ceiling (statically undergoing only gravity) and another while upside up on the table, and likewise undergoing only gravity. The mean value for these two results could possibly give us an approximation for a theoretical non deformed structure, although not exactly precise because the sensorcraft is not symmetrical regarding $xy$ plane. Determining this ‘non deformed structure’ turns out to be useful to calculate strains correctly, because they are determined based on a calibration which is done when the sensorcraft is right side up on the table. This being said, every strain calculated is regarding a position which is not a non deformed one. Therefore an average between table and ceiling positions must be obtained so that strain might be corrected. Besides, this is also important for the same reason, to calculate twist angle.

There are three sets of main data that will be taken from the experimental test, as previously referred. These will be strains ($\varepsilon_{xx}$), displacements ($\Delta z$) and twist angles ($\theta$) on both aft and forward wings. Experimental results will then be compared to FE ones, to assess the accuracy of the current computational model. A total of 15 tests will be carried out and they are shown in table 4.1, where tests 1 through 9 are for the complete sensorcraft and tests 11 through 15 are related to the tailless aircraft. Materials used are itemized as follows:

- **6 1/4 inch screws.** Used to clamp sensorcraft’s rig to the ceiling.

- **Sensorcraft clamping rig.** The final support was shown in figure 3.17.
• **CMM and fiducial markers.** The coordinate measuring machine is used to measure structure’s deformation based on highlighted dots with fiducial markers.

• **Photographic camera with tripod.** A tripod and photographic camera are placed and not moved throughout the tests. Pictures are taken for every load step, so that an overlay might be created and presented, as might be seen ahead.

• **Led small spheres and plastic bags.** Led is placed inside bags to recreate the desired masses, for every load step.

• **Mobile work station.** The movable station (which might be seen in figure 4.8) has a computer directly connected to the sensorcraft’s data logger. Video information from cameras and strains are collected in this station.

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</tr>
<tr>
<td>Test 9</td>
<td>Ceiling</td>
<td>Gravity + 100% of Maximum Load</td>
</tr>
<tr>
<td>Test 11</td>
<td>Table</td>
<td>Gravity</td>
</tr>
<tr>
<td>Test 12</td>
<td>Ceiling</td>
<td>Gravity</td>
</tr>
<tr>
<td>Test 13</td>
<td>Ceiling</td>
<td>Gravity + 5% of Maximum Load</td>
</tr>
<tr>
<td>Test 14</td>
<td>Ceiling</td>
<td>Gravity + 10% of Maximum Load</td>
</tr>
<tr>
<td>Test 15</td>
<td>Ceiling</td>
<td>Gravity + 20% of Maximum Load</td>
</tr>
<tr>
<td>Test 16</td>
<td>Ceiling</td>
<td>Gravity + 40% of Maximum Load</td>
</tr>
</tbody>
</table>

Table 4.1: Sensorcraft tests.

Figure 4.6: Complete sensorcraft deformation.
The sensorcraft was successfully tested and all required data was collected. A MATLAB script had to be developed to post-process the data acquired, for both displacements and strain. The script had the goal of reading and parsing, the displacement data outputted by the CMM, and the strain data coming from the strain data reader software. It then organizes and plots it with equivalent data from FE results so that results may be compared. Some pictures are now shown.

![Figure 4.7: Rear cameras' view.](image1)

![Figure 4.8: Sensorcraft and the mobile station used for data acquisition.](image2)

### 4.3 Experimental Results

Data is provided in plots where in the $y$ axis is represented the quantity in question and in the $x$ axis there is the distance, for either the aft or forward wings, along which the desired quantity varies. This distance is non-dimensional. Concerning the forward wing, in the displacements and twist angles plots, '0' represents the root of the spar, *i.e.* the beginning of the flexible part of the wing and '1' means the wing tip. For the strain, '0' represents the root of the spar and '1' portrays the end of the spar. Regarding the rear wing, in the displacements, twist angles and strain plots, '0' represents the root of the spar, *i.e.* the beginning of the flexible part of the wing and '1' symbolizes the end of the aft spar.

Given the conclusions taken on section 3.7, the deformation may still be considered linear up to a load factor of 10% of the maximum load case. Consequently and to safeguard results with a safety factor regarding the prediction of linearity, test 3 and 13 (5% of the maximum load case + gravity) will be the
chosen ones with which the sensorcraft will be updated. As follows, figures 4.9 and 4.10 present the experimental results obtained for the complete sensorcraft, at the same time compared with FE results. The same boundary conditions were applied for both FE and experimental. The light blue line represents the experimental results. The light green line symbolizes the FE analysis’ results, with non-linear solution method, whereas the dark green represents the FE, with linear solver, results. On the right side of each graph there is the right wing and on the left side, the left wing, *i.e.* in a pilot’s perspective. The thinner red line portrays the RE between experimental and FE analysis (with non-linear solver) results, where:

\[
RE = \frac{|X_{FE} - X_{GSLT}|}{|X_{GSLT}|} (4.1)
\]

Where, \(X\) represents the quantity being analysed, which might be either \(\Delta z\), \(\varepsilon_{xx}\) and \(\theta\). \(FE\) stands for the FE non-linear analysis’ results and \(GSLT\) for results measured experimentally from the test.

Relatively to the forward wing, whose plots are represented in figure 4.9 and its displacements, the error plot is unsteady close to the root and surrounding area. There are several peaks and valleys on the right side which represent each of the displacements measurement points. This bumpiness might be explained by the lack of symmetry in the structure and the fact that in the inner part of the wings (in the zone close to ‘0’), displacements are smaller and the variation of error between sequential dots might be larger. It is possible to say that the RE is around 17% for the left side and around 13% on the right side of the sensorcraft. Strain was only measured in one side of the aircraft, the right one. The computational strain follows the trend of the experimentally measured one, growing almost linearly, spanwise. The values obtained from FE analysis are not as high (in magnitude) as the measured ones. The error goes from 26% close to the spar root, reaching 36% near the end. If a look is taken again, back into the displacements plot, there is the chance of logically associating the facts. The FE model is not displacing as much as the as-built aircraft, therefore its strain is also smaller. Regarding \(\theta\), relative error is smaller in the inner part of the wings where it is around 10% for the left side and 15% for the right side, but grows up to 35% for both sides. The portion where \(\theta\) remains steady is the outboard part of the wing, where FE results present roughly 1° less twist.

In the rear wing’s displacements plot, which can be seen in figure 4.10, it is clearly possible to see how unsymmetrical the aircraft is on the rear area. There is a bias, most likely to be there for manufacturing reasons. This fact translates into a more troublesome model update. Given the non-symmetry of the as-built aircraft. Either the update is based in only one side of it or it is done regarding an average between both sides at the same time. This will be further discussed in the next chapter. The left inner side of the aft wing is looser as it deforms more. Eventually, going towards the outer part of the wing, \(\Delta z\) between both sides become similar. There is a big peak of error of around 95% on the left side of the root. Apart from that, the error is smaller in average on the right side comparing to the left one. Once again, the trend is preserved, *i.e.* the FE model is less flexible than the experimental aircraft. Regarding \(\varepsilon_{xx}\), results turn out to be quite similar between experimental and computational data, although there are two main peaks (one in each side), which happen due to results being close to 0, in those locations. Apart from that, error is around 64%. For the twist angle, it is observable how
the left side has larger error comparing to the right one. Also, the trend is similar and there is a slight difference between the linear and non-linear solution methods, specially on the inner part of the wing.

As theoretically assumed previously, it is possible to verify that for the 5% load case, the behaviour might still be considered approximated to linear. It is observable that linear and non-linear solutions behave similarly, specially for displacements and strains, where the linear solution still provides excellent results. Regarding the twist angle, it is slightly different between both solutions, particularly in the aft wing, which is the component more susceptible to non-linearities.

Results for the linear load case were presented and analysed. Now it is important to go through the same procedure, instead for the maximum load case, the 5.659 kg. It is of vital significance to understand how the sensorcraft behaves on its full deformation and how better the non-linear solution method is compared to the linear one. The overall behaviour is similar to the smaller load case, i.e. the as-built aircraft is more flexible than the computational model.

In figure 4.11, displacement plot shows that the RE is quite similar to the one presented previously, being around 15%. Besides, in terms of displacements, the linear solution method proves to be still a good approach. Regarding strain, the error fluctuates between 16.5% and 20% and, once again, the outer part of the wing is where the error is larger. Concerning twist, the error is larger as it goes towards the wing tip and has similar values to what was presented for the previous load case. The linear and non-linear solution methods are still similar. The forward wing behaviour by itself is still roughly well represented by a linear approximation.

Figure 4.12 introduces results for the aft wing. In terms of displacements there is a slight difference between linear and non-linear solvers and, it is possible to see how the latter is the closest to the experimental results. Also, the non symmetry verified in the other scenario is not as much clear with the maximum deformation occurring, although the error is larger in the left side, as predictable. Neglecting the peak happening at the root, the error is around 35% for the left side and 20% for the right one. The trend for the FE analysis non-linear $\varepsilon_{xx}$ is similar to the one observed experimentally, however results are off by around 100%, this is explained by the fact illustrated in figure 2.2. Strain is being taken out of the computational model in the edges of the plate elements. Given the way the aft spar was modelled, this was the only place equivalent to where strains are measured experimentally. Therefore, results are already expected to be considerably off in terms of magnitude, however the trend is similar, which is important. Strain presents well why a linear solution method is not advisable for such a flexible structure. It clearly does not follow the experimental trend. $\theta$ data strengthens even more this fact. Linear and non-linear analysis have totally different results and this is where an appropriate non-linear solution method is important. The twist RE between FE and experimental is around 25%.

A table resuming all the data, quantitatively, will be presented in the next chapter.
Figure 4.9: Results for the GSLT with 5% of the 2.25 g maximum load being applied - forward wing.
Figure 4.10: Results for the GSLT with 5% of the 2.25 g maximum load being applied - aft wing.
Figure 4.11: Results for the GSLT with 100% of the 2.25 g maximum load being applied - forward wing.
Figure 4.12: Results for the GSLT with 100% of the 2.25g maximum load being applied - aft wing.
Chapter 5

Model Updating and Results

The last step in the static test process is to update the computational model. Regardless of the fidelity of the structural model, it needs to be validated and evaluated by experimental testing of the as-built aircraft. The model updating procedure is a time-consuming, not well defined and complex task. Nevertheless, model updating is a necessary step in the design process because the finite element model needs to be a faithful representation of the physical model in order to ensure a safe and reliable flight testing step.

Model updating allows the designer to understand how the system response is influenced by design variables. The model is updated using mass, inertia and stiffness parameters, using data from static testing. The update procedure in terms of elastic moduli has been performed using Dynamic Design Solutions’ FEMtools software. The program is a versatile tool used in experimental static analysis.

In this chapter, the update procedure is discussed and the results are presented. First, the model update of a spar is carried out in order to understand the process and test on a simple structure. Next, the full sensorcraft finite element model is validated and evaluated.
5.1 Model Update Procedure

With reference to the model update procedure, several aspects need to be clarified. The model was updated using the Young's modulus solely. This is due to the fact that the geometries of the aircraft should not be changed, because the finite element structural model has a direct influence on the ASWing aero framework results. Therefore, by updating stiffness parameters only, there is no need to update the aerodynamic model, from a geometric point of view. The target responses are displacements and no other criteria is being considered when updating the model, although other data is collected experimentally and computationally. The only requirement that needs to be fulfilled is the matching of the static deformation of the sensorcraft. This is due to its high flexibility. The FE model needs to accurately represent the deformation experienced by the as-built physical model, so that aeroelastic computational analysis can properly estimate the flight loads and deflections.

Every element of the full sensorcraft model had an initial Young's modulus of \( E = 7.17 \times 10^{10} \) Pa, with the exception of the re-updated spar's elements. Those numbers will be updated for every different section based on experimental results.

The update of a computational model is a time-consuming task and it is based on trial and error, in the sense that there is a decision that needs to be made in relation to the elements to update and the most efficient way to do it. Sometimes, it is better to update elements by region and at different steps. Also, it is possible to update all elements at once in a single step. The steps for a generic model update procedure are as follows:

1. Data is imported into FEMtools interface namely the deformed experimental data (measured dots, which \( a \ priori \) need to go through a previously developed MATLAB script to convert it into an openable file by the program) and the FE model framework.
2. An analysis is carried out for the FE model with the desired boundary conditions (5% of the maximum load case + gravity) using the linear Nastran solver.
3. Two correlations are done. The first one is a node-point pair correlation, where the nodes from the FE model are associated with a respective point from the experimental results, so that later on, \( \Delta z \) might be matched between previously correlated nodes and points. The second one is the displacement shape correlation where both deformed shapes are correlated and the difference between displacements is recorded.
4. Responses are selected. Here, \( \Delta z \) are chosen as target responses, however not every measured point is used.
5. Parameters to be updated are defined. In this case, the modulus of elasticity for the desired elements are chosen as updating parameters. Only selected elements are updated.
6. Automated sensitivity analysis is carried out to assess how sensitive is the structure when deforming, with different elastic moduli for the several elements or groups. The goal is to better under-
stand how the structural responses and behaviour of the computational model are influenced by changing its properties.

7. Model is updated. FEMtools reads the computational $\Delta z$ and iteratively changes EM of the elements previously chosen as target parameters, to match the desired experimental $\Delta z$.

The updating parameters work like the unknowns in an equation and the target responses are the equivalent to the constraints/boundary conditions applied to it. As referred previously, if there are too many update parameters, FEMtools diverges and does not solve the problem. If too many target responses are chosen, the system is then over constrained and once again FEMtools incompletely solves the problem, leaving parameters not updated or with unreasonable values. Therefore, sometimes the update process previously explained, has to be carried out separately for different groups of elements being updated. There are two distinct ways of updating elements’ properties, which are represented as follows:

- **Local update.** With this type of update, elements’ properties chosen as target parameters are updated individually, *i.e.* each element has its own property. Local parameters are physical properties of a single node/element [24].

- **Global update.** As represented in figure 5.1 there are different groups of elements within the FE model that intend to represent the several logical different sections of the real aircraft. The global update procedure is based on changing the property of those groups in order to achieve the desired displacements. Every element within a group has the same properties. Therefore, a global parameter represents a coincidental change of physical properties of a set of nodes/elements. A global parameter is a linked group of local parameters [24].

![figure 5.1](image)

(a) As-built aircraft  
(b) FE model

Figure 5.1: Different sections used for model updating.

### 5.2 Spars Update

Spars were updated based on a still considered linear range of loads. Displacement was collected in three different points along its span and they were used as reference displacements to be matched to the FE model within FEMtools (tool used for the update of computational models). As follows:
• **Forward spar.** It was updated based on the deformation imposed by a load of 4 kg.

• **Rear spar.** It was updated based on the deformation imposed by a load of 0.4 kg.

![Image](image1.png)

(a) Forward spar was updated and a final Young's modulus value was retrieved as being $E = 7.368 \times 10^{10}$ Pa.

![Image](image2.png)

(b) Rear spar was updated and a final Young's modulus value was retrieved as being $E = 6.454 \times 10^{10}$ Pa.

Figure 5.2: Spars' update based on displacements.

Once the updated Young’s moduli are retrieved, as explained in section 3.8, those numbers are introduced in the tailless and complete sensorcraft models. Table 5.1 summarizes the information regarding the new EM, along with figure 5.2.

<table>
<thead>
<tr>
<th>Spar</th>
<th>Initial Elasticity Modulus [Pa]</th>
<th>Updated Elasticity Modulus [Pa]</th>
</tr>
</thead>
<tbody>
<tr>
<td>Aft</td>
<td>$7.17 \times 10^{10}$</td>
<td>$6.454 \times 10^{10}$</td>
</tr>
<tr>
<td>Forward</td>
<td>$7.17 \times 10^{10}$</td>
<td>$7.368 \times 10^{10}$</td>
</tr>
</tbody>
</table>

Table 5.1: Spar’s updated properties.

### 5.3 Sensorcraft Model Update

#### 5.3.1 Tailless Model Update

Once the spars are updated, the new stiffnesses are introduced into the tailless sensorcraft model. Afterwards, the model is updated. It is important to highlight that regarding the spars, the material and geometry are physically well known. This means that within the update process, these specific elements are being re-updated, although inside a restricted window for the EM. As the material and physical properties are known, it makes no sense to give total freedom to these groups of elements (which on some initial attempts occurred). Otherwise these elements would have EM with physically unreasonable values, which is not desirable. Figure 5.3 shows FEMtools interface where experimental measured dots are overlaid onto FE model. Figure 5.1 shows the different groups to be updated and, table 5.2 refers to those groups.
As will be explained in the next section for the complete sensorcraft model, the fuselage is stiff enough to be considered rigid. Therefore, and as a result of data collected in figure 5.4, the stiffness is high enough so that these two groups are not updated. For the light green group, the value changed quite insignificantly. For the yellow group, the stiffness was reduced, making the spar more flexible. The red group also had its stiffness reduced.

### 5.3.2 Complete Model Update

In figure 5.1 the group represented by the white colour is the one that portrays the fuselage. From the white part onwards it is already considered wing. However, as previously explained, there are two different crucial parts of the wings: rigid and flexible. The black and light green parts are the initial rigid part of the forward wing. It is crucial to understand how rigid these three different section are.

First, the fuselage (white part) is theoretically stiff enough to be considered totally rigid. The elements which make part of this section have a big cross sectional area, therefore even with a big change in elasticity modulus, the behaviour of the structure remains similar. Also, as referred previously, geometries were not changed and so, cross sectional area had to remain the same. Taking this into consideration, there is no reason to update the value for this group of elements, because in the FE model it is already behaving as a rigid part. The previous reasoning is also valid for the first part of the wing (black part). Hence, a graph was created to demonstrate how point 56 (highlighted) moves as loads are applied, in figure 5.4. As might be seen, there are slight oscillations which are due to the imprecision of the measurement system, namely, rotations on the coordinate system between loads that make coordinates slightly change. Besides this, it is observable that point 56 does not move and has roughly the same
coordinate on the lowest and highest loads. Thus, we can consider this part of the wing to be rigid and an update to these elements is unnecessary. Regarding the light green area, it should still be considerably rigid, however, because it contains the fixture for the forward spar, some flexibility might be induced. Thus, freedom for this group for the update shall be given. For every other group represented in figure 5.1, freedom for update shall be given.

Figure 5.4: $z$ axis coordinate vs. load at point 56.

Knowing that the update is being carried out in a flexible aircraft, specially being it a joined wing, one of the most critical and crucial areas are the existent joints. Taking this into consideration, the first step is to try to update individual elements which represent important joints. These are all the beam elements which account for forward spar’s inner and outer fixtures, aft spar’s inner and outer fixtures and junction between rear and front wing.

The update process was carried with several attempts which are described as follows:

- **1st attempt.** Results are acceptable but there is room for improvement.
  1. Update joints locally, *i.e.* updating its elements’ EM individually (interface between yellow and light green areas, interface between pink, yellow and red, and interface between blue and orange areas). These properties are blocked after being updated and the next step is done independently.
  2. Update spars and every other group, globally, *i.e.* each group has its own elasticity modulus but all elements within a group have the same EM.

- **2nd attempt.** Although results seemed reasonable they never become as good as they are for the chosen final procedure. Updating joints is a difficult task because it easily changes aircraft’s shape.
  1. Update joints locally, *i.e.* updating its elements’ EM individually and update spars and all the
other allowed groups referred previously, globally. These tasks were done simultaneously, opposing to what was done previously.

• 3rd attempt. Results were acceptable for displacements, yet regarding strain and twist angle, data becomes irregular because the spar is divided into three different groups. When updating the EM, in order to match displacements, the three different groups within the spars end up having considerably distinct updated values. This makes $\theta$ and $\varepsilon_{xx}$ inconsistent, spanwise. Thus, it is not a good approach.

1. Spars (yellow and orange groups) are updated sub-globally i.e., they are divided into three smaller groups of elements. These spars’ sub-elements are locked after being updated and the following step is done independently.

2. Remaining aircraft groups are updated globally.

• 4th attempt. Results are similar to the ones achieved previously. The same inconsistencies are found on strain and twist data due to the sub-global update on the spars.

1. Spars are updated sub-globally and every other group is updated globally. Both tasks are done at the same time, being freedom given to all parameters available.

• 5th attempt. It became difficult to exactly match the experimental displacements, with every parameter being updated at the same time. For some parameters, results are unreasonable and are not good enough to match the desired goal for $\Delta z$.

1. Every group is updated globally and at the same time.

• 6th attempt. This proved to be the best approach. There are several different ways of doing the update procedure separately, yet for the ones tried, results were good.

1. The sensorcraft is updated globally, yet groups are updated separately, in several different ways.

The first two attempts were done with the intention of understanding how the joints are important and how focusing on it locally might be a wise approach. Results were acceptable but not good enough, although its importance is known. It might be an approach to deeply explore in the future. Third and fourth attempts had the intent of understanding how good it could be to sub-globally update the spars. Fifth and sixth attempts have proven to be the best ones. The final stiffnesses are presented in table 5.3.

5.4 Updated Data

The same results as presented previously are now presented in the same shape, yet for the updated structure. The framework was updated based on the linear deformation. New results for the 5% load case are being presented. Afterwards, updated results for the maximum load case are also shown. It is
important to understand how the linear update extrapolates over to the non-linear behaviour. Concerning the forward wing, represented in figure 5.6 the achieved improvement is clear. Displacements wise, the experimental and FE non-linear are almost overlaid. The average relative error is reduced to around 4%, where the biggest error occurs in the root surrounding area, due to the lower $\Delta z$ values. For $\varepsilon_{xx}$, the lines get closer and the RE ends up being around 18%. Once again, the error is steadier (around 16%) in the inner part of the wing, where the strain is larger in magnitude, and then grows significantly in the outer section to around 23%. Relatively to $\theta$, the error drops to around 10% and it is larger on the outer part of the wings, thus the FE model is gradually not twisting as much as the as-built aircraft, spanwise.

The rear wing data is represented in figure 5.7. Displacement error is roughly half of the initial one, being now around 16%. The aft wing is the most critical area, being displacements impossible to match along its span, and visually results are considerably better. Given the non symmetry of the sensorcraft, it is impossible to match both left and right sides equally. A small improvement is achieved on the strain data, but the error is still considerable, being around 54%. Strain is the more problematic area, taking into consideration how the rear spar was geometrically modelled, and FE results never come close to experimental ones. The twist drops to around 50%. Left side is closer to the linear FE results and the right side is really similar to the non-linear ones.

Regarding the update for the 5% load case, improvements for all quantities were achieved. This is even more evident for the forward wing where errors are reduced to more than half the initial ones, which is also expected, given the fact that the area where non-linearities are more evident is the aft
wing, for which there is also a reduction in error but not as significantly. Displacements still yield good results. Strain is still not good after the update, although this was expected, given the aft wing modelling, as previously explained. The geometric configuration of the plate elements is highly suspect for strain results. Also, the twist angle is more susceptible to larger errors, because it is calculated from two points whose coordinates are being measured. Given the small distance between those two points, if both have, even a small amount of error, this can cause a considerable accumulation of error for the $\theta$ calculations. Results show that the trend is similar, which is a good sign that the chosen method for angle measurement is a valid choice. Taking into consideration that the sensorcraft is being updated solely on displacements, it is likely that the updated values for $\theta$ might not be as accurate as $\Delta z$. Therefore, and being the twist for the aft wing small (up to a maximum of $2^\circ$, in the 5% load case), even one degree of difference (which is a small magnitude) causes already a considerable amount of error.

Results are now presented for the 100% of the 2.25 g load case. It is important to understand how the update procedure, based on the lower and still linear load case, influences the maximum displacement case. The forward wing is represented in figure 5.8. In terms of displacements, experimental and FE non-linear match almost perfectly. The relative error between both drops to an average of 4%, which is a really good result. The plot for $\Delta z$ along wing’s span shows how the light blue and light green lines are almost overlaid. Regarding strain, lines get closer, and the difference is now smaller when comparing to the updated 5% load case, being now the average RE around 7%. Twist angle is also considerably improved, with a relative error between experimental and FE non-linear of 5%. The forward wing as a whole presents surprising results for the maximum load case. All three measurands improved significantly the respective results, becoming FE non-linear similar to the measured experimental data.

Results for the aft wing are presented in figure 5.9. Displacements are considerably better, being the error now around 18%. However, it does not have the ideal shape, which was impossible to achieve without causing a dramatic change in either twist or strain, being this the best achieved result for the several different options tried. The FE aft wing does not buckle as much as the as-built one. This fact might also be due to the way the spar was geometrically modelled. Concerning $\varepsilon_{xx}$, the shape was corrected and it is now similar to the experimental measured one, although with a considerable difference in magnitude. Once more, the huge error is due to the lack of accuracy of the model as far as strains is concerned, in the aft wing. $\theta$ was also improved, having now an error of around 13%. The FE model is now more flexible, displacing and twisting more, accordingly to the as-built aircraft. Both twist and strain, in this case, show how important it is to approach this problem with an accurate non-linear solver, being the difference between linear and non-linear solutions hugely different.

Being the sensorcraft updated based on measured $\Delta z$, it was expected for this quantity to be the most precise one, which could provide a better insight over the update procedure, i.e. FE model’s Young’s moduli were updated using the measured displacements, and for the sake of the update procedure itself, this is the only quantity that can tell if the update procedure was successful. Although the good improvement achieved for both strain and twist, they are not a major fact when assessing the feasibility of the update procedure. A summarizing table with average errors encountered for the analysed data is next presented.
Figure 5.6: Updated results with 5% of the 2.25 g maximum load being applied - forward wing.
Figure 5.7: Updated results with 5% of the 2.25 g maximum load being applied - aft wing.
Figure 5.8: Updated results with 100% of the 2.25 g maximum load being applied - forward wing.
Figure 5.9: Updated results with 100% of the 2.25 g maximum load being applied - aft wing.
<table>
<thead>
<tr>
<th></th>
<th>5% Load RE [%]</th>
<th>100% Load RE [%]</th>
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<tbody>
<tr>
<td></td>
<td>Initial</td>
<td>Updated</td>
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<td></td>
</tr>
<tr>
<td>$\Delta z$</td>
<td>14.5</td>
<td>4.1</td>
</tr>
<tr>
<td>$\varepsilon_{xx}$</td>
<td>30</td>
<td>18.3</td>
</tr>
<tr>
<td>$\theta$</td>
<td>28.1</td>
<td>10.1</td>
</tr>
<tr>
<td>Aft Wing</td>
<td></td>
<td></td>
</tr>
<tr>
<td>$\Delta z$</td>
<td>32.4</td>
<td>15.9</td>
</tr>
<tr>
<td>$\varepsilon_{xx}$</td>
<td>64.3</td>
<td>53.6</td>
</tr>
<tr>
<td>$\theta$</td>
<td>71.7</td>
<td>50.1</td>
</tr>
</tbody>
</table>

Table 5.4: Final results summary.
Chapter 6

Conclusions

Conclusions over the work done in this kind of project is an important chapter to go through. This is relevant to tell whether the approach and methods used were successful or, even if not, if it is worth trying other processes and ideas.

In this chapter, achievements, conclusions and future work to be done are discussed.
6.1 Achievements

The work carried out for this thesis was completed in a timely manner. All the stated objectives were achieved, yielding valid and useful results. The static load test was well designed and executed, and the experimental data was acquired successfully which also allowed a proper updating of the finite element model.

The experimental data on the static non-linear behaviour of the flexible joined wing aircraft can be used for the validation of the Boeing design framework.

The model updating procedure presented a challenge when selecting the design parameters. The updated framework may also be used to validate the dynamic non-linear aeroelastic behaviour. The goal was to understand the behaviour of the structure when undergoing significant loads, that induced a non-linear response of the structure in terms of the twist angle on the aft wing.

The significant conclusions from the thesis research can be summarized as follows:

- The MSC Nastran’s 400 non-linear solution method provided results that agreed well with the experimental data.
- The aft wing presents clear evidence of non-linear behaviour. If the forward wing is being considered alone, a linear solver might still be accurate, however considering also the existence of the aft wing, a non-linear solver must necessarily be used.
- Overall, a reduction of more than half the original error between experimental and computational results was achieved. For the maximum load case, the average error is around 10% for all measured (excluding strain in the aft wing).
- It was found that the aft wing spar geometric modelling is inadequate, in terms of strain measurement, for this specific case. A new modelling is suggested, either with CQUAD elements in a different layout or using 3D elements instead, as it might be seen ahead.
- Experimental static test procedure proved to be successful and an adequate approach to update the finite element model of the non-linear aerostructure.

Although the update was done based on linear deformations, and using a linear solution method for the inherent computational analysis, its effectiveness extrapolates well over to the non-linear maximum load case. As it might be seen on table 5.4, errors are actually smaller between FE and experimental results, for the maximum load case, which was the one that was not used as reference for the update procedure. Also, in general, computational model represents better forward wing’s real behaviour than aft’s one. This might be due to several facts, namely, aft wing’s modelling and joints between forward wing and aft wing. Besides, experimental results are assumed to be the reference because they represent what is really happening, however they also entail their own share of error. This error might be due to experimental measuring (from CMM and strain measuring rig) and also due to associated defects of the as-built aircraft for manufacturing reasons. In the data shown in previous chapters it is possible to see some inconsistencies between left and right sides of the aircraft (experimental results) which show
the lack of symmetry that made the update procedure even more demanding. These errors are more
evient in $\theta$ measurements, which was the quantity being acquired with the highest likelihood of error
for two main reasons. Firstly, $\theta$ was being calculated based on $\Delta z$ measurements of two points, and if
those two points have an associated error, twist would be influenced by a larger amount of experimental
error. Secondly, the wings had panels (on top of which $\theta$ was being measured) whose surfaces were not
perfectly aligned, given their process of manufacturing (composite hand-built panels).

6.2 Future Work

The next step in this project is to improve the finite element model (geometry or the elements used
when modelling the aft spar), to adequately capture the strain measurements. The initial model was
built with the purpose of being simple and parametric and easily adjusted for development of a design
space, to better understand sensorcraft's behaviour. It was arguably a good approximation, from what
static load test results tell. However, improvements could be carried out as follows:

- Some joints are modelled as rigid elements, not allowing then as much freedom for the update
  procedure as it could. Also, the freedom of different sections against each others becomes com-
  promised. A joint parametrisation for the several interfaces within the structure would be an im-
  provement. These main interfaces are the forward wing with the fuselage, the aft wing with the
  boom and the forward wing with the aft wing.

- Adapting the aft wing to better represent the interfaces where it was decided to measure strain
  from, i.e. to remodel its geometry and constituent elements. Some suggestions are given:

  - The simplest way of remodelling would be to maintain the plate elements, although changing
    their position so that strain could be measured in the middle of the surface and not in the
    edge. This solution might be seen in figure 6.1(b).

  - Aft spar could have a hybrid modelling with 2D shell and 3D solid elements. For instance, the
    same modelling with horizontal shell elements in the middle section could be kept, with the
    two protuberances being changed to 3D solid elements. Hence, strain would be measured in
    the side of a solid element. This can be seen in figure 6.1(c).

  - Lastly, aft spar could be integrally modelled with solid 3D elements, reliably recreating the
    as-built spar, as represented in figure 6.1(d).

Besides, a characterization of the dynamics of the non-linear joined wing airframe should be pursued.
To this end, a ground vibration test should be carried out to fully understand the non-linear behaviour of
the aft wing and unknown properties. Modal analysis is a trending subject. Finally, the last future task
shall be the flight test. It is important to understand how the data gathered on ground correlates with the
data that should be collected during flight.
Figure 6.1: Aft spar modelling - cross section.
Bibliography


[22] MSC Nastran 2014 Non Linear User’s Guide In *MSC Software*
