UNIVERSIDADE DA BEIRA INTERIOR
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Modelling of Standard Components to Test Full-Scale Aeronautical Structures

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Dedicatória

Aos meus Pais, como tributo pela educação que me deram.

À minha Mãe, a quem nunca poderei retribuir a sua inefável dedicação.

Ao meu Pai que sempre me apoio e por todos os esforços possíveis e impossíveis feitos.

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Resumo

Nesta tese é desenvolvido um sistema de testes estruturais simples, recorrendo a componentes já disponíveis no mercado. Esta investigação nasce devido à necessidade de aprofundar e solidificar o estudo de um componente por parte do CEIIA e, para isso, será desenvolvido uma estrutura capaz de testar tal componente.

Com estes testes pretende-se um sistema capaz de testar e validar tal componente, segundo as normas que regem a indústria aeronáutica mundial. Do ponto de vista do projecto de uma aeronave, os testes para certificação são de extrema importância. Estes permitem verificar que não existem erros de projectos, que a estrutura aguenta as condições previstas no projecto e, finalmente, assegurar que não constitui um perigo, devido à importância do componente.

A metodologia utilizada para se desenvolver um trabalho conciso e coeso foi bastante diversificada e adequada à situação. Para isso foi necessário contactar vários fornecedores após o componente ter sido projectado em CATIA e recorrer ao software ABAQUS com o objectivo de obter uma análise simplificada sobre o comportamento estrutural do componente. Uma análise bibliográfica aprofundada sobre vários tipos de testes efectuados e de várias estruturas de ensaio existentes desde 1920 tornou-se de extrema relevância para o desenvolver da dissertação.

Com os resultados obtidos, posso afirmar que o modelo utilizado é capaz de realizar estes testes estruturais mas, no entanto, são necessários mais estudos e, de forma a assegurar que esta solução é mais barata do que as outras que são analisadas, é necessária uma comparação dos preços intensiva.

Abstract

This thesis develops a simple structural test system, using components already available in the market. This research is relevant because there is a need to deepen and solidify the study of a component by the CEIIA and for that will be developed a structure able to test this component.

It is intended a system able to test and validate the component according to the rules governing the global aviation industry. From the point of view of an aircraft project, the tests for certification are paramount. These allow to check that there are no errors in the projects, that the structure withstands the conditions of the project and, finally, ensure that doesn´t constitute a danger due to the importance of the component.

The methodology used to develop a concise and cohesive work was diversified and appropriate to the situation. This required several vendors contact after the component has been designed in CATIA, software ABAQUS in order to obtain a simplified analysis on the structural component. A thorough literature review on various types of test structures existing since 1920 has become extremely important for the development of the dissertation.

With these results, I can say that the model used is able to perform these structural tests, however, more studies are needed. And to ensure that this solution is cheaper than the others that are analyzed, it is required price comparison intensive.

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List of Acronyms

C.G. .... Center of Gravity
CEIIA.. Centro para a Excelência e Inovação na Indústria Automóvel
CFRP .... Carbon Fiber Reinforced Plastic Technology
CS.. Certification Specifications
CTA.. Centro De Tecnologias Aeronáuticas
DIN.. Deutsches Institut für Normung
DSTO.. Defence Science and Technology Organisation
EASA... European Aviation Safety Agency
EMBRAER.. Empresa Brasileira de Aeronáutica
EU.. European Union
FAA... Federal Aviation Administration
HB.. Brinell Hardness
ICAO.. International Civil Aviation Organization
ITS... Infinity Testing Solutions
JAA.. Joint Aviation Authorithies
MIT.. Massachusetts Institute of Technology
NAE.. National Aeronautical Establishment
NIAR.. National Institute for Aviation Research
NWTC.. National Wind Technology Center

1. Introduction

1.1 Objectives and motivation

In 2006, Embraer, an aerospace conglomerate in Brazil, who participates and develops its functions in various phases like design, development, manufacture, sale and after-sales support in the aviation industry, including commercial aviation and business aviation, began studies on a tactical military aircraft, KC-390, intended to become a substitute to C-130. This kind of important project needs help from all kinds of enterprises and *Centro para a Excelência e Inovação na Indústria Automóvel* (CEIIA) has the responsibility to develop engineering solutions of the complete set of elevators considered a primary structural component, the central fuselage, that is one of the main components of the airplane and sponson, which is considered a critical structural component for the KC-390. Both products are made from CFRP (Carbon Fiber Reinforced Plastic Technology) as the primary structural material strategically reinforced with metallic parts. CFRP is the name given to a compound material combining carbon fiber and matrix resin. It is light and strong, and is therefore used in a range of applications, from the aerospace industry through to general industrial parts and sports equipment. One of the tasks assigned to this company and included in my master's thesis is to develop a test facility capable of performing static and fatigue tests on the elevator.

Figure 1 - KC-390 (1)

The primary objective of this Master Thesis is to research a much more expeditious and cheaper way to test an aeronautical component. For that, a test facility capable to acclimate to any kind of situation will be needed. The structure must be able to perform full-scale tests in virtually all kind of aeronautical components and also partial-scale tests. Additionally, those tests must comply with the certification applied to these kinds of situations.

1.2 Component to be tested

To understand and analyze the component that will be tested and studied in this thesis is necessary to understand the motion of fixed-wing aircraft. It is characterized by wings that gain lift by the forward movement of the aircraft in relation to relative wind and it is affected by changing natural conditions such as wind, temperature and time of day. As seen in the Figure 2, there are three axes whose origin corresponds with its centre of gravity (CG), a point where the force of gravity is centered on the aircraft. Its movements are defined by the rotations around these axes. There are three:

- Roll, L, around the x-axis. This movement results from the action of the ailerons, located in the wings;
- Yaw, N, around the z axis. This movement results from the action of the rudder located on the vertical stabilizer;
- Pitch, M, around the y-axis. This movement results from the action of the elevator located on the horizontal stabilizer.

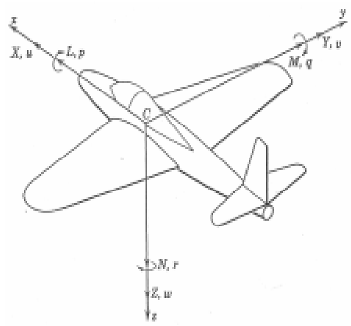


Figure 2 - Standard notation for aerodynamic forces and moments, and linear and rotational velocities in body-axis system (2)

If the pilot wishes to pitch up the aircraft, a flight control system is required to rotate the aircraft around the pitch axis (M) in a nose-up sense to achieve a climb angle. Upon reaching the new desired altitude the aircraft will be rotated in a nose-down sense until the aircraft is once again straight and level.

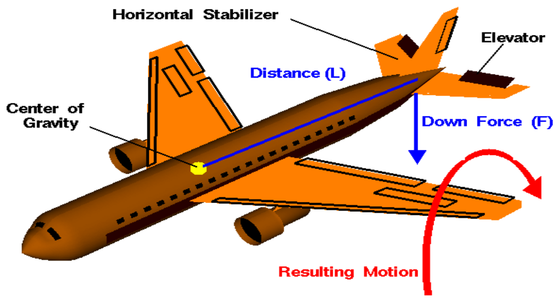


Figure 3 - Illustration of the pitch moment (3)

To control the pitch motion on an aircraft it is necessary to change the amount of lift generated by small wings, located on the trailing edge of the tail plane or horizontal stabilizer, called elevators. The elevators are attached by hinges to the stabilizer, which is a fixed wing section whose job is to provide stability to the aircraft, i.e., it prevents up-and-down motion of the aircraft nose. The elevator is the small moving section located at the rear of the stabilizer. Deflecting the elevator generates a moment about the c.g. which causes the airplane to pitch.

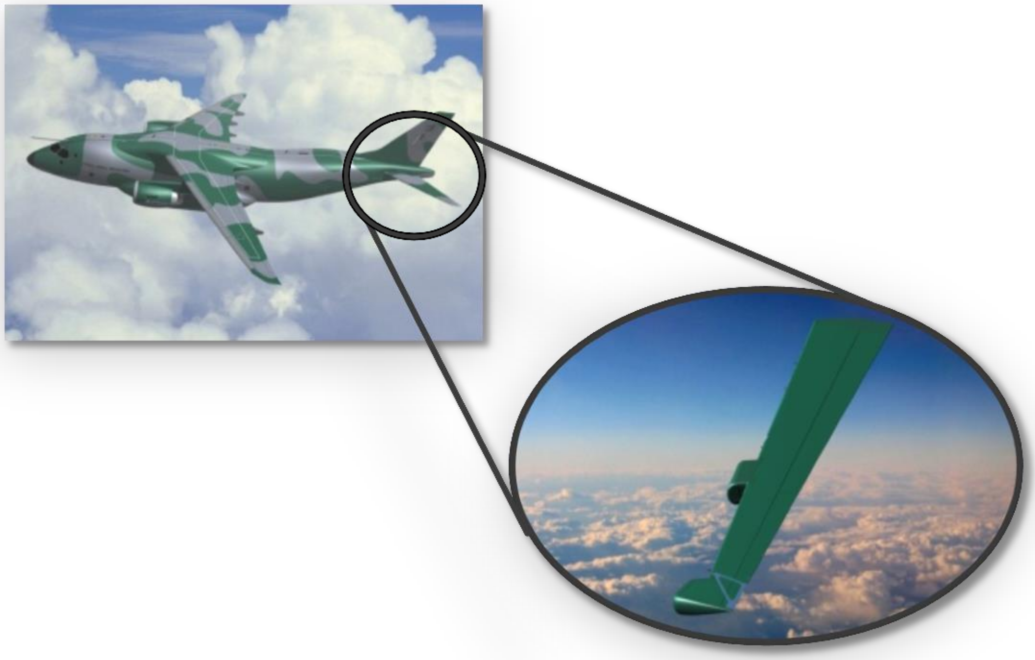


Figure 4 - Position and detail of the Elevator

2. Certification

All aircraft and its components have to be able to fly in safe conditions. The possession of certificates is required to ensure those conditions. The applicable JAA/FAA/EASA airworthiness standards for the certification of aircraft to be internationally recognized are issued in accordance with the ICAO Annexes.

JAA is a European organization, existing since the '70s, responsible for controlling aviation throughout the European Union. This results from the combination of several civil aviation regulatory authorities of several European countries, that have joined together to develop and implement safety standards and procedures to be applied and used in all countries equally.

Later, in 2002, regulations of all countries of the European Union were replaced by the creation of the EC Regulation 1592/2002 (Basic Regulation) of 15 July 2002, that addressed common rules in the field of civil aviation. This situation led to the creation of the European Aviation Safety Agency (EASA, whose rules and regulations imposed are mandatory for each Member State of the European Union. With the publication of the revised EC Regulation 1592/2002 on 19 of March 2008, EASA took over all responsibilities which were previously in the hands of JAA.

In 2008, they decided that the JAA Liaison Office in Cologne would be disbanded by 30 June 2009, but the JAA Training Organization continues to develop their work and their activities in order to contribute and cooperate with EASA.

In order to obtain more information about the regulation of aircraft belonging to countries that are not under jurisdiction of EASA (Europe) or FAA (United States of America), EASA states that:

“An Administrative Arrangement on product certification between EASA and the Brazilian Authorities is in force since February 2004. The Arrangement covers the acceptance of new and used airplanes produced by Embraer. (…)

It is the intention of the Commission to prepare a recommendation to the Council in order to authorize the Commission to open bilateral negotiations with Brazil. EASA will assist the Commission in this task. In the case of Brazil the initial idea is to develop a Bilateral Agreement which establishes the “reciprocal acceptance” of certification findings in the field of civil aviation safety and environmental compatibility, covering the certification of aeronautical products (aircraft, engines and propellers), parts and appliances.” (4)

CS-25

In this thesis there is a need to summarize and present the legislation in force for these types of tests. The document to be followed is the CS-EASA 25, equivalent to FAR PART 25 of the FAA. In annex 1 there is an excerpt of CS-25 corresponding to the part of the regulation that fits better in this situation. For this type of tests it is necessary to make a presentation of the most important rules to be respected.

2.1.1 Loads

Loads are forces, deformations, or accelerations applied to a structure or its components. They cause stresses, deformations, and displacements in structures. Excess load or overloading may cause structural failure, and hence such possibility should be either considered in the design or strictly controlled. Mechanical structures, such as aerospace vehicles, marine craft and material handling machinery have their own particular structural loads and actions. The evaluation of structural loads is based on published regulations, contracts, or specifications.

"Strength requirements are specified in terms of limit loads (the maximum loads to be expected in service) and ultimate loads (limit loads multiplied by prescribed factors of safety). Unless otherwise specified, a factor of safety of 1.5 must be applied to the prescribed limit load. According to the limits, these may not exceed the values at which the structure is proven and the values at which compliance with each applicable flight requirement. These loads must be distributed to conservatively approximate or closely represent actual conditions and the methods used to determine load intensities and distribution must be validated by flight load measurement unless the methods used for determining those loading conditions are shown to be reliable." (5)

2.1.2 Static Tests

The static tests are also important and play a key role in the smooth operation of the aircraft. Thus, it became necessary to investigate more about these and the rules to be respected in relation to them.

"In this kind of test, the structure must be able to support limit loads without detrimental permanent deformation and support ultimate loads without failure for at least 3 seconds. Static tests conducted to ultimate load must include the ultimate deflections and ultimate deformation induced by the loading. " (5)

2.1.3 Damage-tolerance evaluation

Finally, the damage-tolerance evaluation from an airplane other becomes another factor to be considered in the development of this thesis.

“An evaluation of the strength, detail design, and fabrication must show that catastrophic failure due to fatigue, corrosion, or accidental damage, will be avoided throughout the operational life of the airplane. The service history of airplanes of similar structural design must be evaluated having regard of differences in operating conditions and procedures. This evaluation must include a determination of the probable locations and modes of damage due to fatigue, corrosion, or accidental damage. Damage at multiple sites due to prior fatigue exposure must be included where the design is such that this type of damage can be expected to occur. The evaluation must incorporate repeated load and static analyses supported by test evidence.” (5)

3. Aircraft Structures Testing

3.1 Need of tests

To assure that a structure will function as expected there is a need for validation which is supported by an extensive test. It is the process by which we gain confidence that the structure will work as customers expect and demand. A detailed analysis of aircraft operations is therefore of crucial importance in the context of structural testing.

The materials and design methods are, as a rule, a mixture of techniques already in use and new. However, it is important to use the most recents materials and the latest and most developed methods, so we can be constantly monitoring them during the tests. Before the design phase progress to the production phase, it is necessary to ensure that the structure will have the expected performance, so the indicated corrections can be made without affecting the program.

During the design phase, structural tests are necessary to reduce the uncertainties due to structural failure. The various components that need to be tested (sample materials, elements, assemblies, components of the aircraft) can be grouped into a pyramid, shown in Figure 6, comprising three levels: test coupon, elements and certification tests. In accordance with the existing methods, the determination of the number of tests is based on previous experience. Since these types of tests are expensive, there is an incentive to reduce these costs without affecting the safety of aircraft.

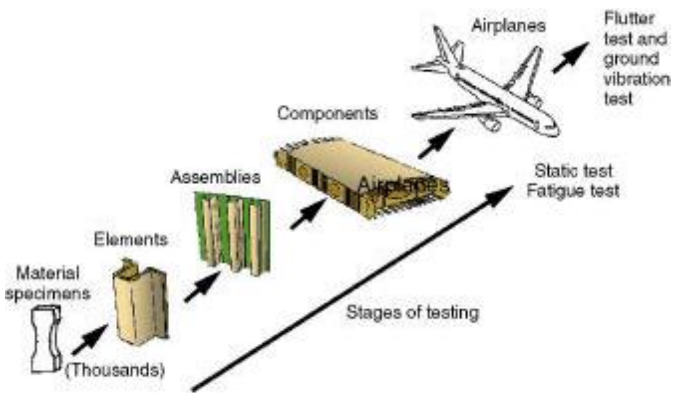


Figure 5 - Levels of structural testing (6)

Coupon are tested in order to obtain information about their proprieties like Young's Modulus (is a measure of the stiffness of an elastic material and is a quantity used to characterize materials. It is defined as the ratio of the stress along an axis over the strain along that axis in the range of stress), Poisson's Ratio (ratio of the relative contraction strain, or transverse strain normal to the applied load, to the relative extension strain, or axial strain in the direction of the applied load), and yield strenght (stress at which a material begins to deform plastically). Increasing the amount of tests to be carried out, more information will be aquired about these properties of the material, but more expensive becomes this process, so there must be a consensus on the number of specimens to be tested.

It is important to know if coupons, when assembled, continue to be safe and for that structural element tests are needed. Finally, in a third level the aircraft is tested as a final product. These kind of tests also pretend to reduce failures due to errors in structural element tests and coupon tests. The main objective of this type of testing is to reduce the errors relating to failures that could happen.

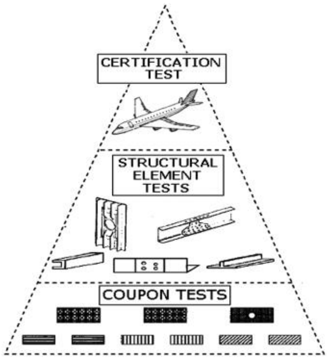


Figure 6 - Simplified Three-Level Tests (7)

3.2 Types of tests

Tests on aircraft components can be divided into three main types: static tests, dynamic tests and vibrations tests. Since the first tests in the beginning of the 20th century, the basic principles have not changed, but the way to run them suffered many changes. The most important example of this evolution is recorded in the data acquisition. The first tests were done manually with the people responsible for recording the deflection at certain points. Electronic devices nowadays are used to record position changes on the order of 0.0025 cm.

In addition to the tests mentioned above, which suffered its evolution over time, and adequate monitoring developments in aeronautics, also the utensils used in the load application have changed and there was a gradual evolution, consistent and, therefore, important to in the form as they were applied. We can make a comparison between the beginning and the best gift for observing these developments. Initially, the forces applied were simulated by using sandbags. With the advancement and development of technology, hydraulic pumps, driven manually by some officials are currently controlled by machines evolved and tested properly for your order, thanks to which it is able to handle more than 150 hydraulic servo-hydraulic systems

3.2.1 Test Preparation

Before any kind of tests, it is always necessary to describe all the loads that will be submitted to the component. Information about these loads can come from both the company that designs based on other existing components static, as well as regulatory authorities. The methods for prediction of loads have evolved in conjunction with the increase of knowledge in aero elasticity, finite element methods, computational fluid dynamics as well as the increase of knowledge about computational tools. It is also necessary to establish the properties of the materials and methods of drawing using existing data.

3.2.2 Static Tests

One of the main objectives of this type of testing is to ensure that the material in question keeps on elastic system when subjected to loads below or equal to the load limit calculated at the design stage. These tests are made to bring the structure to its physical limit in order to ensure that this will support at least 150% (ultimate load) of the efforts that will be subject during his lifetime (limit load). Bellow in Table 1 are shown some results of static test of Boeing's aircraft. We can notice that almost all tests were successful since the components reached the ultimate design load.

Airplane Model	707	727	737	747	767	757	777
% wing ultimate design load @ failure	110	110	106	115	99.4	111	103
Failure location	Lower Panel	Upper Panel	Upper Panel	Upper Panel	No Wing Failure	Upper Fail-safe Chord	Upper Panel

Table 1 - Full scale airplane static test results (6)

3.2.3 Fatigue Tests

Fatigue tests examines how the aircraft structure responds to stress over a long period of time and during different stages of its operations, such as taxiing on the runway, take-off, cruising and landing. In these stages there are some efforts like bending the wing due to the lift, cabin pressurization or the efforts that the landing gear is subject to each landing/take-off.

It is also necessary to test ability to sustain defects safely until repair is done. The approach to account for damage tolerance is based on the assumption that flaws can exist in any structure and such flaws propagate with usage. This approach is commonly used to manage the extension of cracks in structure through the application of the principles of fracture mechanics.

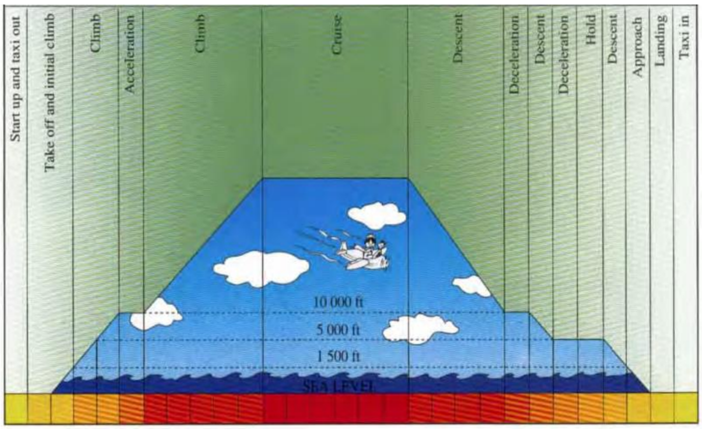


Figure 7 - Typical fatigue mission (8)

4. History of Aircraft Structures Testing

4.1 Brief Introduction

At the beginning, when it was necessary to test a wing there were not many choices. One of the first methods to be used in this type of test consisted on loading the wing with sand, loose or in bags, to check the maximum weight that this withstands. This method did not involve a lot of costs, since the necessary materials were simple and low cost. Sandbags were placed to simulate at each point of the wing the load caused during a flight.

Figure 8 - Test of a wing in 1911 (9)

Only in 1920, when Donald Douglas set up his company, new methods of testing of wings began to be developed. One of the first Aeronautical Engineers trained at the MIT, Douglas, brought knowledge and mathematical tools for this process.

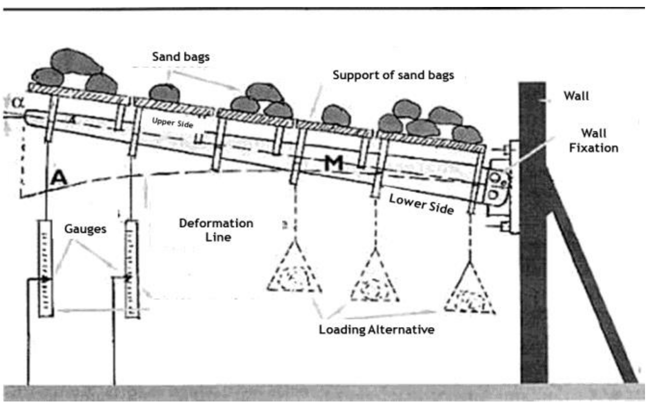
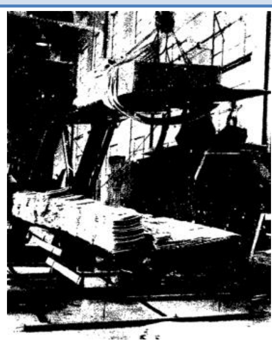


Figure 9 - Wing testing with sand bags

4.2 Testing chronology

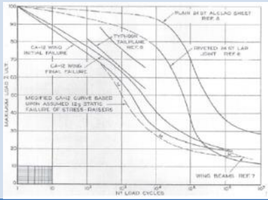
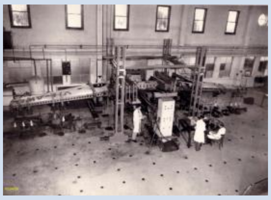
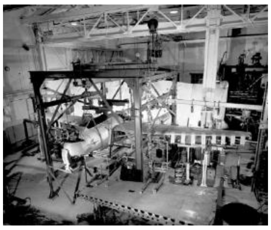
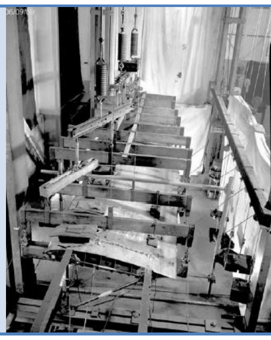
Two of the first laboratories devoted to tests of aircraft structures are DSTO (10) and NAE (11). The first belongs to the Australian Government and is called Defense Science and Technology Organization while the other, belonging to the Government of Canada, is called by National Aeronautical Establishment. And for both of them I have found some records that are presented below.

In the follow three tables, tests performed are presented in chronological order. As the most important evolution that there was in this area was the way in which the components were loaded, the tests will be divided into two subgroups.

Aircraft	Notes	Test Structure
Avro 652  **1941**	One of the first tests carried out on this Laboratory was a static test fuselage of Anson. The loading was simulated by applying sandbags with the help of an operator. Deflections were registered by reading scales. The structure was loaded up to 75% of the ultimate load in both side and down and permanent deflection not seen.	
De Havilland DH-82  **1942**	This essay had as objective to test a defect in the construction of the wings and was tested for 400, 800 and 1000 hours of flight. The test was also done using sandbags supported by some actuators. Wings were charged up to 120% of the ultimate load without damage.	
Beechcraft T-6 Texan  **1942**	One of the first tests carried out in completely metallic structures was in 1942. The objective was to verify the conditions under which the wings showed evidence of a potential fracture. With the wing root bolted to a face plate, it was charged up to 90% of the ultimate load with sandbags. Deflections were measured with strain gauges.	

De Havilland DH 98  **1948**	Followed to a fault detected in a Mosquito wing during the first flights, static tests were carried out to determine such failure. The structure was developed to apply a load distribution on the wing using a set of high-precision actuators. The load distributed across the wing was applied using a total of 16 jacks supplied with hydraulic fluid from four different pressure channels. The objective of this test was to measure the force required to achieve the ultimate load. During the test were placed microphones inside the wing structure for a better perception of moments when structural damage occurs. These occurred in 95% of the ultimate load.	

Table 2 - Tests performed in NAE

Aircraft	Notes	Test Structure
Commonwealth CA-13  **1949**	An improved version of test structure allowed simulation of a life time of 50000 cycles. These tests were part of an investigation that ended in 1949 and resulted for the first time on a fatigue diagram (S-N) for a full size structure.	 
B North American P-51  **1950 - 1960**	To get more information on structures manufactured using aluminum alloys were investigated, after the Second World War, 20 wings of the Mustang P-51.	
De Havilland DH.104  **1950 - 1960**	In order to ensure that this aircraft had a life cycle of more than 100 000 hours, tests were made using the same structure used for the Mustang adding certain differences for best results in vibration testing.	
De Havilland DH.100  **1960 - 1975**	In the case of this aircraft was needed a structure able to test the wings built in the fuselage.	
Cessna 180  **1960 - 1975**	In that aircraft, fatigue tests were performed due to previous fatigue failures, and also due to a catastrophic accident caused by fatigue failures.	
Dassault Mirage III  **1960 - 1975**	Since this aircraft is supersonic, it was decided that before structural tests would be collected data during the various stages of flight.	

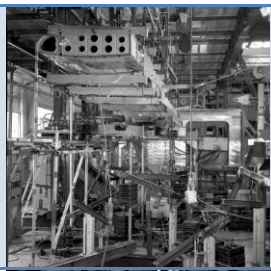
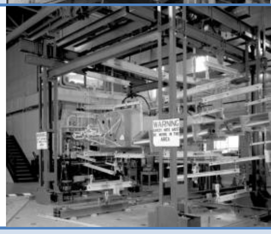
GAF N-22  **1976**	Fatigue tests were performed in the GAF's wings and fuselage. The main wing survived a simulation of 300 000 hours without failure.	
PAC CT/4  **1972**	This test differentiates the previous once the component to be tested is suspended in a structure where are also mounted the actuators and associated mechanisms.	
Pilatus PC-9  **1986**	In that aircraft was necessary to test the main support of the landing gear, empennage, fuselage and wings. The testing framework is designed to be easy access to the central part of the wing. This allowed workers to inspect this component during the test.	

Table 3 - Tests performed in DSTO

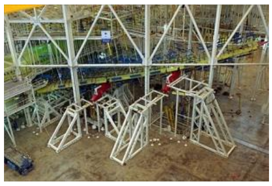
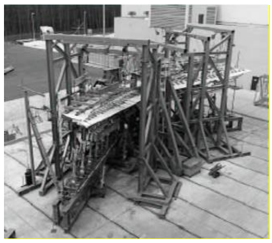
Aircraft	Notes	Test Structure
Caiga HO300  **2011**	The ultimate load was applied from 10% to 100%, holding for 3 seconds and then released. It has conducted up to 67% of the ultimate static load strength test. The test focuses on the strength of the left and the right wing and the joint area on them. (12)	
Airbus A380  **2006**	After completing limit load tests, progressively greater loads have been applied to the specimen towards the required 1.5 times the limit load. The failure occurred between 1.45 and 1.5 times the limit load at a point between the inboard and outboard engines. The ultimate load trial is an extremely severe test during which a wing deflection of 7.4m was recorded. (13)	
Boeing 787 Dreamliner  **2010**	Started by testing small coupons and elements to characterize the material system, testing small and medium size articles of configured structure and then proceed to full-scale testing. (14)	
Boeing 777  **1994**	Boeing tested the 777 horizontal stabilizer and elevators separately. Engineers computed test loads for each static load condition to match the required shear, moment, and torsion values. These test loads were applied to the stabilizer using hydraulic actuators connected to attachment fittings, mounted on the stabilizer structure. (15)	
Airbus A320  **1990**	In the case of the A320-200 in the fatigue test 120000 cycles were made and half of these had as objective the study of cracks. (8)	

Table 4 - Other Examples

5. Development of a Testing Facility

5.1 Examples of test facilities

5.1.1 Test Facility at NAE, Canada

In the year 1947, a larger and more sophisticated testing facility was built in the NAE. The test structure consisted on a series of up to 12 frames made up from steel in which holes were drilled at a standard gauge, pitch and assembled by bolting in the required arrangement.

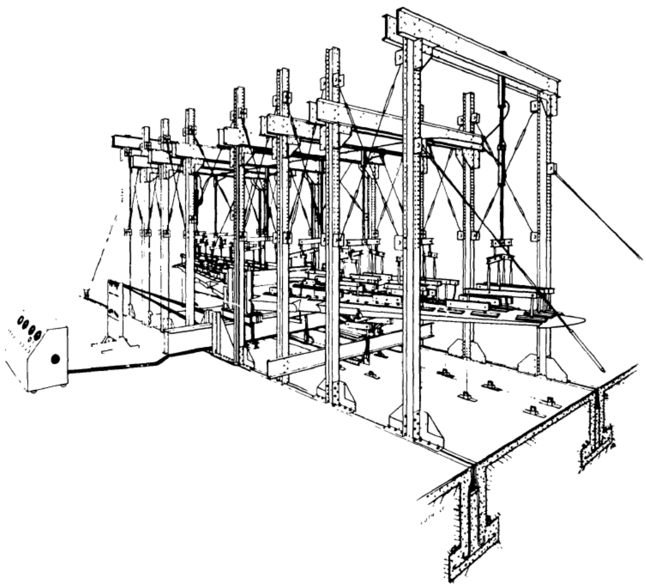


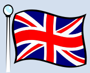
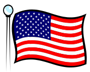
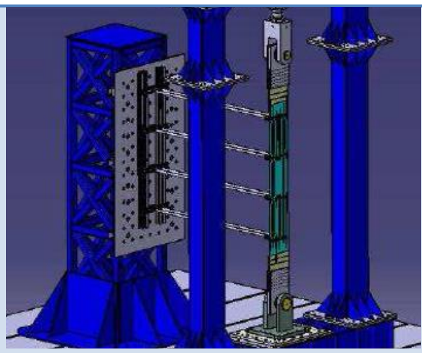
Figure 10 - Static Test Facility in NAE (11)

The structure was anchored using four inverted T-section reinforced concrete beams under the floor with two 15 inch steel channels. The loads were applied by hydraulic reaction jacks suspended. Two hydraulic consoles were available to supply the hydraulic fluid at four different pressures simultaneously. Loads were transmitted to the test article by means of adhesive tension patches.

Deflection was measured using deflection boards. These were simply large sheets of plywood mounted in a vertical plane, with small pulleys, mounted on the top. A small weight was attached to different wires to provide tension and then any vertical displacement of the point of attachment was directly indicated by a similar displacement of the weight.

5.1.2 Other examples

In Table 5 we can see static tests performed on wings in MARSHALL, NIAR, CTA, IABG and NWTC. In all the test facilities there are beams used to apply force across the span and also it is found that in all cases there is a floor plate with rails to fit various types of tests.

Enterprise	Pictures	
Marshall (16) 		
NIAR (17) 		
CTA (18) 		

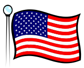
IABG (19) 		
ITS (20) 		
NWTC (21) 		

Table 5 - Other examples of test facilities

5.2 Presentation of the test

After an analysis of other tests structures it is necessary to design a new one with the requirements that have been provided. A structure capable of carrying out static and dynamic tests to an elevator is needed. There are several requirements (22) :

- A bridge crane with an operating range of 12 meters;
- Floor with rails positioned lengthwise with a gauge of 1 meter between them. Must withstand a traction load of 20000 kg/m. Equipped with an appropriate drainage channel;
- Actuators fixed in the floor acting vertically;
- Easy to adapt to other tests;
- Easy to assembly and handling;

The test facility that stood is in Spain, at CTA. This structure were usedfor structural tests on the elevator of the Airbus A320, and the component to be tested is the same as in CEIIA and the actuators behave the same way, that is, placed on the ground acting vertically.

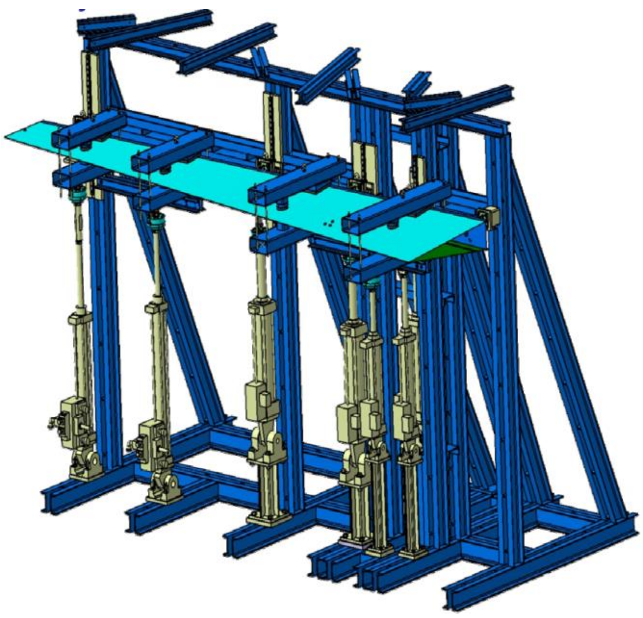


Figure 11 - A320 Elevator Structural Test (23)

This structure provides various supports aimed at transferring to the elevator the forces exerted on the actuators. This device will not be necessary, since it was previously established that this transfer of efforts would be made through two structures, one placed at the root and the other on the tip. Another requirement was to reduce the size of the elevator by cutting the tip and the root following the dimensions given by CEIIA. The elevator with approximately 6 meters wingspan have been transformed into another of approximately 2 m conserving central part as shown in the Figure 12.

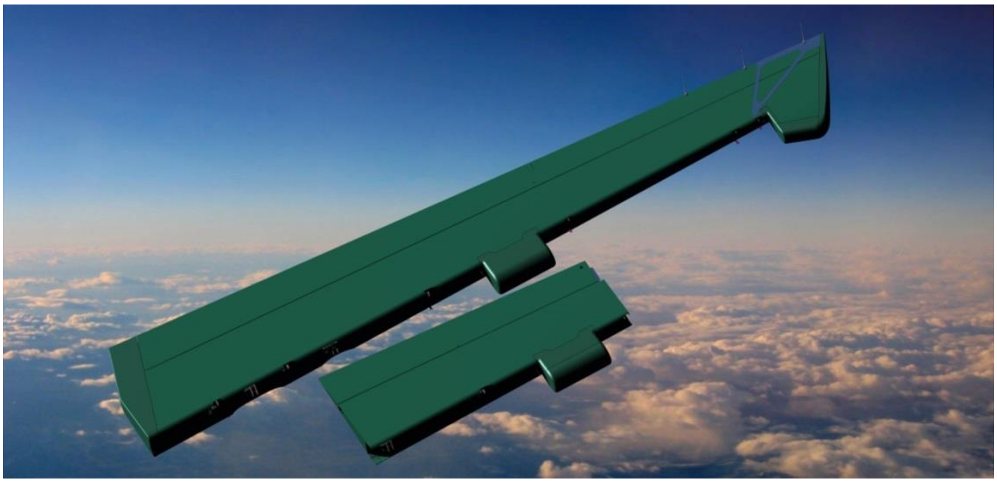


Figure 12 - View of the cut of the elevator

5.3 Ground Anchor

The ground anchor will be made using a steel-plate floor. Those plates are made from a cast iron/steel alloy and offer flexible universal clamping possibilities. The plate chosen is from Stolle, located in Germany, and one of the most recognized manufacturer of these kind of parts and presents a plate with Hardness Brinnel of approximately 180 – 200 N/mm^2 and tensile strength between 260 and 300 N/mm^2. Compared with other solutions this is the one with more ease of clamping.

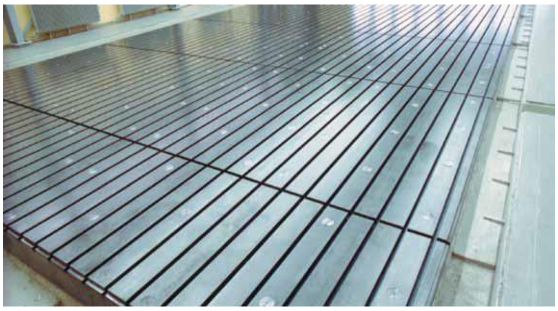


Figure 13 - Floor plate measuring 21 x 11.5 m with individual parts measuring 7 x 2 m and 7 x 1.5 m (24)

Another solution presented by Stolle consist in rails implemented in concrete. This type of structure is cheaper since it is only necessary to build on an existing floor, and another advantage is the ease of choice of the gauge between the rails. This solution is worse than the plate floor, because it has the risk of outside interference tests.

Figure 14 - Floor with clamping rails (24)

So, the most viable solution is a type of ground that is independent of the rest of the floor. For that, plates with t-slots were chosen.

Knowing that the width of the component to be tested measures about 5 meters given the structure that will suppor, this dimension of the plate shall be 7 meters so a gap exists where supports of the structure will be and will also allow future tests of other different components. As for the length, since the component measures about 1 meter, we will need a Board with this dimension equal to 4 meters.

Figure 15 - Surface Plate installed at Hyundai Technical Center (25)

So we will use two plates whose reference is 74200 and with a length of 7 meters and width of 2 metres, performing a total size of 7 x 4 meters. Given its specification (24), the density of the material is approximately 2514 kg/m^3 since I use a height of 0.350 m on the plates. So the weight of each plate is 11760 kg. On our plates, the distance between T-slots is 250 mm. When individual plates are combined into a plate field, the T-slot layout is chosen so that the clearance to the edge of the plate is exactly half the slot distance. This provides then a uniform T-slot layout over the entire surface.

A nominal size of 28 mm for M24 bolts was chosen, so with specifications Table 6 we now have plate designed. In order to fix other elements to the ground will be necessary at first T-slot nuts which are elements that are embedded in the plate's soil. For each set screw nut will need one of these. Since the slots correspond to the chosen hole M24, they have a diameter of 24 mm as well as the screws.

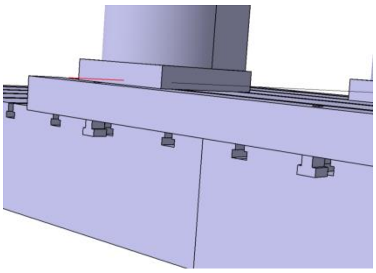


Figure 16 - Detail of clamping on the floor using nuts

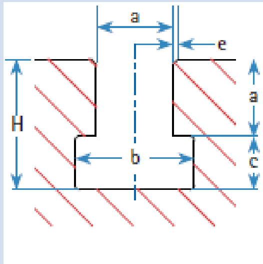
Nominal size: a [mm]	For bolts	b [mm]	c [mm]	H [mm]	e [mm]	Dimensions
10	M8	17,5 - 18	8	18	1,0	
12	M10	20,5 - 21	9	21	1,0	
14	M12	23,5 - 24	10	24	1,0	
16	M14	26,5 - 27	11	27	1,0	
18	M16	29,5 - 30	12	30	1,5	
20	M18	33,5 - 34	14	34	1,5	
22	M20	37,5 - 38	16	38	1,5	
24	M22	41 - 42	18	42	1,5	
28	M24	47 - 48	20	48	1,5	
32	M27	54	22	54	1,5	
36	M30	60	25	61	2,0	
42	M36	70	29	74	2,0	

Table 6 - T-Slot specifications (24)

5.4 Bridge Crane

Once we have a structure comprised of modular components, it is important that we have a lifting mechanism capable of supporting and assemble them and also the component to be tested. Overhead cranes, sometimes also called bridge cranes, are cranes with a hoist traveling along the bridge between parallel runways. The solution adopted, an overhead crane, consists on two beams laid out horizontally with a bridge spanning the gap. In this bridge there is a hoist, making possible lateral and longitudinal movements.

Figure 17 - Example of a Bridge Crane

To design this type of structure a crane is required with a runway length with 15 meters, wide with 13 meters and the value of the height will be between 5 and 15 meters. The crane also needs to support more than 3 tons given the weight of some beams.

5.5 Actuators

Through further studies in this area done in parallel concluded that in the case where the elevator would reach the higher bending strength the resulting force was 107.8 N. For that force, the displacement component was 142.8 mm (26).

Given the safety factor above explained, the amount of force required for the actuators is equal to the resultant force by this factor of 1.5, resulting in 161.7 N. Soon the actuators to be used are hydraulic and need to be able to exert a force greater than or equal to that specified.

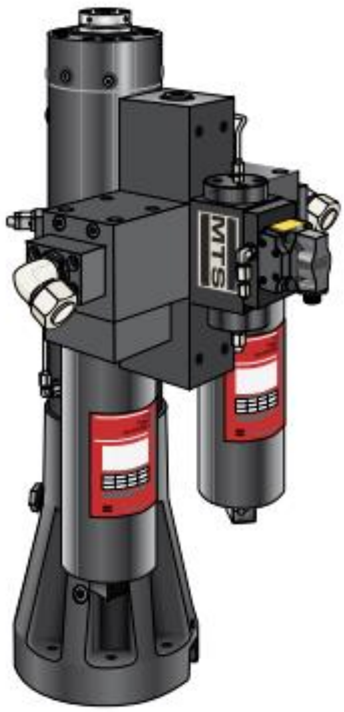


Figure 18 - Example of Actuator from MTS (27)

After the actuator is chosen, it is necessary to select the component which will connect the actuator to the surface to be tested. For this, there are surfaces with holes to be secured to any component by use of screws as can be seen below.

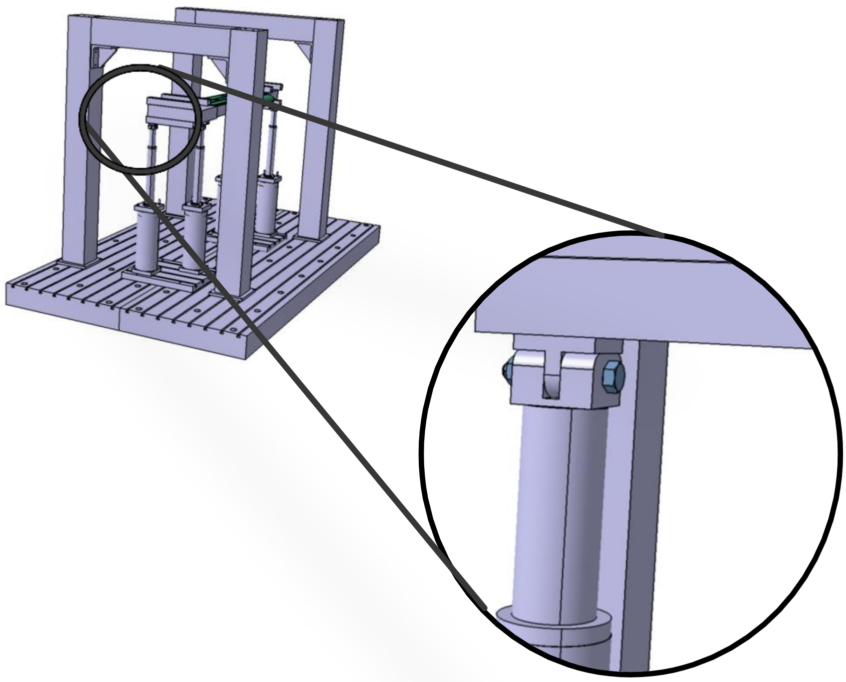


Figure 19 - Elements of connection between actuators and the elevator

5.6 Load Frames

Apart from servo-hydraulic components like actuators we also need appropriate devices to mount actuators and components simply and safely. In order to keep down investment cost, modular solutions are especially useful when test setups are frequently modified. The first components to be installed should be plates that will withstand the actuators in a vertical position (Small Size T-slot plate). These plates have the purpose of facilitating the change of position during and after these tests since they will be mounted on top of the plate with the respective rails to make an angle of 90 ° with the plate.

Then the frames that will be installed, they support the weight of the wing, along with its respective base. Given their positions and weights, it is important they be handled with care and use of crane. This structure is composed by four vertical beams, two horizontal beams connecting the vertical beams using 4 Stiffener supports.

The following table describes and shows in detail the characteristics of the load frames with a figure to make possible a better visualization of the same.

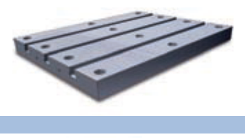
Description	Sizes	Weight [kg]	Reference	Picture
Small Size T-slot plate	900x700x90	405	001 565	
Bottom flange plate	550x550x40	95	001 644.100	
Vertical Beam	3000x350x350	400	001 554	
Horizontal Beam	2000x350x350	300	002 278	
Stiffener support	350x350x370	85	001 564	

Table 7 - Description of the Load Frames (28)

5.7 Connection Elevator-Test Structure

To make the test of component, it is necessary to know where forces are applied. Knowing that the actuators will exercise the necessary force vertically, the most viable solution would be to place two blocks located in the root and the tip of the wing as seen in Figure 20.

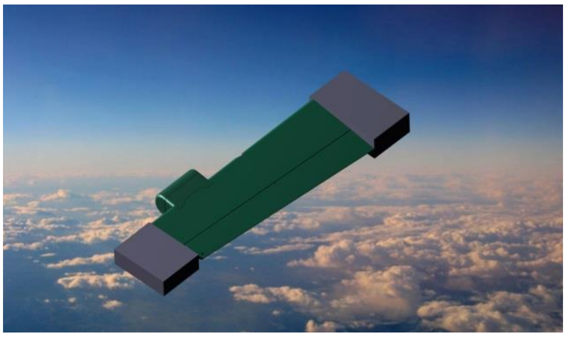


Figure 20 – Elevator with blocks located in the root and in the tip

The actuators will apply the forces in those bocks in order to transfer them to the elevator. Below are two solutions to fix the wing. At first the objective would be to connect the two blocks through two tubes that would pass through the holes of the ribs. This solution has been set aside to the extent that the tubes would lead efforts on ribs that in normal flight conditions would not be present. It could also damage the rib and so, change the properties of the component. Another solution was proposed in which the blocks would fit into the elevator, so efforts to this box and thus avoiding damages that would be present in the previous solution. At the time of delivery of this academic project, the type of fixation was still under study, and therefore it was not possible to present more accurate results or more detailed specifications.

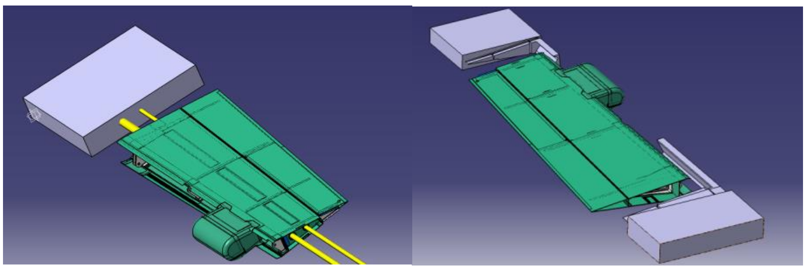


Figure 21 - Solutions proposed for elevator - structure connection

5.8 Measurements

Measurements in structural tests will be carried out using strain gauges properly calibrated to measure tension, torsion and bending moment at different strategic points.

In each structure or piece to be tested are placed several strain gauges. These are small sensors glued to the surface of the material. Small variations in the dimensions of the structure, caused by the application of loads, are then mechanically transmitted to the gauges, which transforms these variations in variations of their equivalent electrical resistance. This variation of electrical resistance generates a signal which is then sent through a wire and then computers decode and treat the information through specific software. Based on this information, it is possible to know if the test material resists mechanical stress specified by project and if you are able to receive the qualification. These strain gauges are still used today, for example, in the case of the Boeing 787 Dreamliner, 9000 actuators were used and 2000 are located in each wing.

Basically a strain gauge is a device which measures strain in a single direction at the surface of a component. Though in some applications this strain may be the primary quantity to be determined, in most cases strain measurements are used to obtain information about the stresses that occur in the components to which the strain gauges are bonded or about the forces which act on such components.

Figure 22 - Strain Gauge on Specimen (29)

As stresses cannot be measured directly, they must be derived by calculation from strain measurements. An introduction to the fundamental physical and mechanical laws is essential. Strain gauge behavior can therefore only be understood on the basis of some fundamental knowledge of these physical and mechanical laws. The characteristic feature of an electrical strain gauge is the change in its electrical resistance with the change in strain. It is therefore possible to determine the strain change by measuring this resistance change.

Each metallic body shows a specific behavior under the influence of external loads. Up to a certain load elastic deformation takes place, i.e., the deformation disappears after unloading. However, if the load is increased beyond this point, first plastic deformation and then fracture occurs.

First, however, the basic characteristics of a strain gauge measurement system will be described. Knowing the gauge factor k, the relation between the resistance change ΔR of the strain gauge and the strain is expressed by:

$$\frac{\Delta R}{R} = k\frac{\Delta l}{l} = k.\varepsilon \;.$$

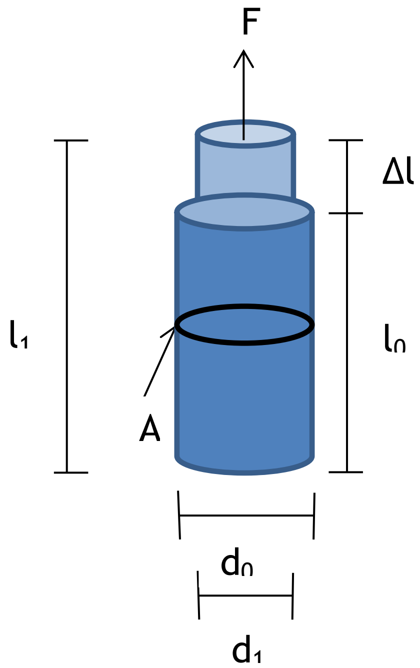


Figure 23 - Circular bar under tension

If we consider a simple circular bar under tension having a cross-sectional area A and loaded with longitudinal force F, as in Figure 23, assuming a uniform force distribution over the entire cross section , the force divided by the cross-sectional area is defined as the stress: $\sigma = \frac{F}{A}$.

Under load F the length l of the bar changes by Δl from l_0 to l_1, $l_1 = l_0 + \Delta l$. The change in length divided by the length l is defined as the longitudinal strain:

$$\varepsilon_l = \frac{\Delta l}{l_0} = \frac{l_1 - l_0}{l_0} \;.$$

Under load F the diameter of the bar changes by Δd from d_0 to d_1, $d_1 = d_0 - \Delta d$. The change in diameter divided by the diameter d is defined as the transverse strain or transverse contraction:

$$\varepsilon_t = \frac{-\Delta d}{d_0} = \frac{d_1 - d_0}{d_0}.$$

The relation between transverse strain ε_t and longitudinal strain ε_l is a material constant known as Poisson's ratio. If the bar is successively submitted to increasing loads and the values of σ are plotted against the values of ε, the result is a stress - strain diagram as seen in Figure 24.

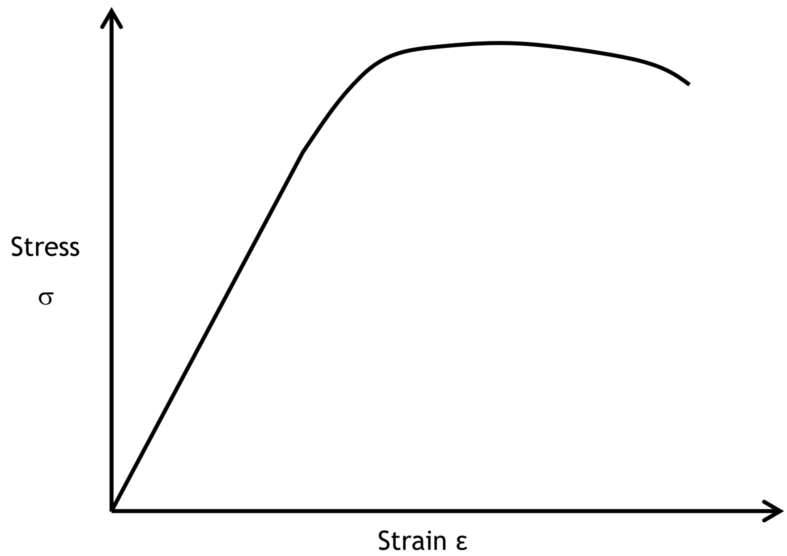


Figure 24 - Typical Stress - strain diagram

Once the strain has been exactly transferred from the surface of the specimen to the grid of the strain gauge, the quality of the measurement is determined by the accuracy of recording the resistance change.

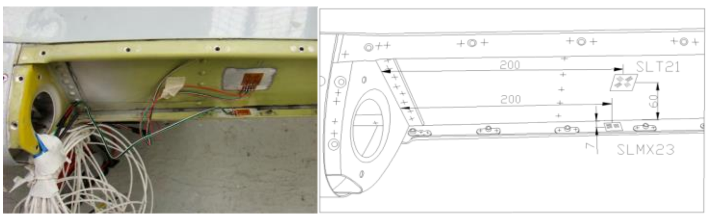


Figure 25 - Strain gauges installed in the middle wing section of the left wing on the front spar. Left part show the actual strain gauges photo and the right one is a part of post installation documentation (30)

6. Conclusion

Structural testing is an important procedure for manufacturer companies all over the world. In order to enhance the methods to perform structural testing capable of certify and approve have been presented. With these methods we can certify aeronautical components in a faster way, at the same time all the process becomes more reliable, which can ultimately lead to lower costs and more productivity.

The design of the test facility in CATIA® developed throughout a three month curricular internship at CEIIA provides important information to build a facility capable to perform all kinds of static and dynamics tests. In order to perform a real test facility, an analysis of the elevator was made to determine displacements results. That information was really important to choose what kind of actuator were needed.

Reflecting on the methodology used and its importance for the development of this thesis and tests, I can say that it has become quite useful and appropriate to the situation under study and the literature review and references were quite diverse, which enabled me to have an enlarging view of the panorama aeronautical world from its beginnings to the present. The main objective of this thesis, project a test facility, was achieved, however, more studies are needed to ensure that the solution is cheaper than other existing in aeronautical industry. To obtain a significant reduction in costs it would be important to contact a larger number of companies in order to obtain information on various types of components as well as their costs.

Primarily, I intended to perform a thorough structural analysis, analyzing all its structures and its components down to the smallest detail, however this structure is still at an relatively early stage of project, which is still not possible to obtain a detailed analysis such as desired. Thus, one of the major difficulties encountered in this analysis was that the actual connection elements of the structure itself are not present and thus inappropriate to the situation under study.

In a short time it became difficult to develop a kind of sophisticated structure like this. Several aspects must be studied such as the choice of actuators and position well to the type of fixation of the elevator to the test facility. However, at long term, using the data in this thesis and the project CATIA will become easier to make a viable structural analysis using detailed information of the component in question.

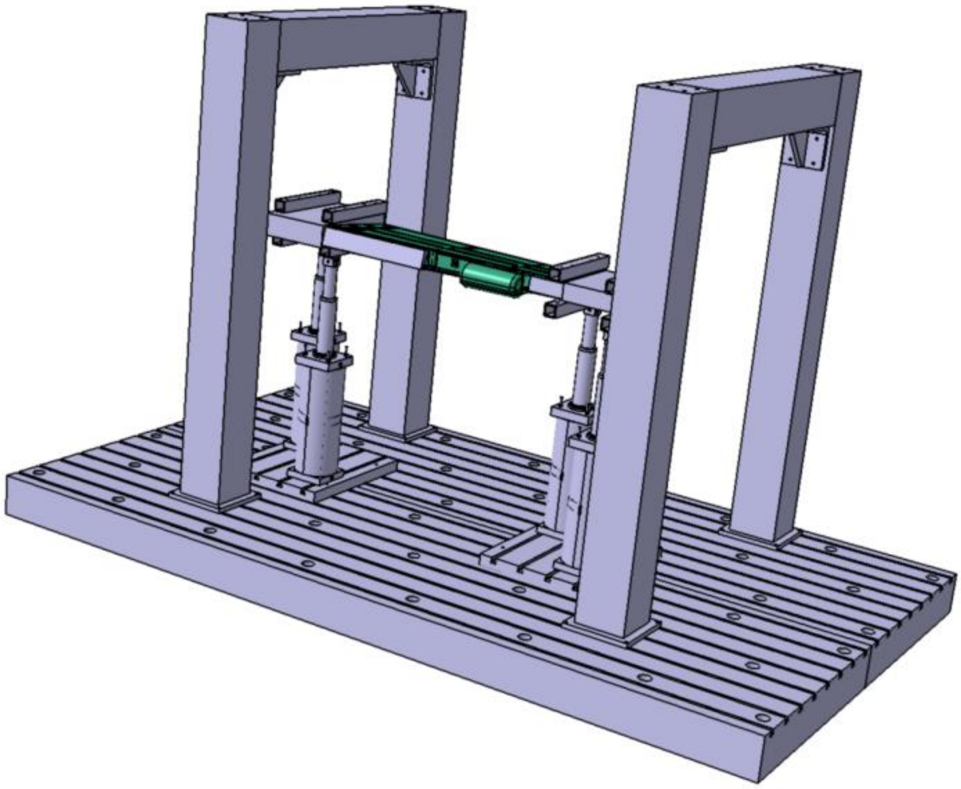


Figure 26 - Final assembly of the test structure

7. Annexes

7.1 Annex I - Certification Specifications and Acceptable Means of Compliance for Large Airplanes

The EASA document that governs large aircraft such as the KC-390 is the CS-25. Here will be presented excerpts of the most important chapters for the case study:

7.1.1 Subpart B

CS 25.21 Proof of Compliance

(a) Each requirement of this Subpart must be met at each appropriate combination of weight and center of gravity within the range of loading conditions for which certification is requested. This must be shown –

(1) By tests upon an airplane of the type for which certification is requested, or by calculations based on, and equal in accuracy to, the results of testing; and

(2) By systematic investigation of each probable combination of weight and center of gravity, if compliance cannot be reasonably inferred from combinations investigated.

(d) Parameters critical for the test being conducted, such as weight, loading (center of gravity and inertia), airspeed, power, and wind, must be maintained within acceptable tolerances of the critical values during flight testing.

(f) In meeting the requirements of CS 25.105(d), 25.125, 25.233 and 25.237, the wind velocity must be measured at a height of 10 meters above the surface, or corrected for the difference between the height at which the wind velocity is measured and the 10-metre height.

CS 25.23 Load distribution limits

(a) Ranges of weights and centers of gravity within which the airplane may be safely operated must be established. If a weight and center of gravity combination is allowable only within certain load distribution limits (such as span wise) that could be inadvertently exceeded, these limits and the corresponding weight and center of gravity combinations must be established.

(b) The load distribution limits may not exceed –

(1) The selected limits;

(2) The limits at which the structure is proven; or

(3) The limits at which compliance with each applicable flight requirement of this Subpart is shown.

CS 25.25 Weight Limits

(a) Maximum weights. Maximum weights corresponding to the airplane operating conditions (such as ramp, ground taxi, take-off, en-route and landing) environmental conditions (such as altitude and temperature), and loading conditions (such as zero fuel weight, center of gravity position and weight distribution) must be established so that they are not more than –

(1) The highest weight selected by the applicant for the particular conditions; or

(2) The highest weight at which compliance with each applicable structural loading and flight requirement is shown. The highest weight at which compliance is shown with the noise certification requirements.

- *Minimum weight*. The minimum weight (the lowest weight at which compliance with each applicable requirement of this CS–25 is shown) must be established so that it is not less than –

(1) The lowest weight selected by the applicant;

(2) The design minimum weight (the lowest weight at which compliance with each structural loading condition of this CS–25 is shown); or

(3) The lowest weight at which compliance with each applicable flight requirement is shown.

7.1.2 Subpart C

Cs 25.301 Loads

(a) Strength requirements are specified in terms of limit loads (the maximum loads to be expected in service) and ultimate loads (limit loads multiplied by prescribed factors of safety). Unless otherwise provided, prescribed loads are limit loads.

(b) Unless otherwise provided the specified air, ground, and water loads must be placed in equilibrium with inertia forces, considering each item of mass in the airplane. These loads must be distributed to conservatively approximate or closely represent actual conditions. Methods used to determine load intensities and distribution must be validated by flight load measurement unless the methods used for determining those loading conditions are shown to be reliable.

(c) If deflections under load would significantly change the distribution of external or internal loads, this redistribution must be taken into account.

CS 25.303 Factor of safety

Unless otherwise specified, a factor of safety of 1·5 must be applied to the prescribed limit load which is considered external loads on the structure. When loading condition is prescribed in terms of ultimate loads, a factor of safety need not be applied unless otherwise specified.

CS 25.305 Strength and deformation

(a) The structure must be able to support limit loads without detrimental permanent deformation. At any load up to limit loads, the deformation may not interfere with safe operation.

(b) The structure must be able to support ultimate loads without failure for at least 3 seconds. However, when proof of strength is shown by dynamic tests simulating actual load conditions, the 3 second limit does not apply. Static tests conducted to ultimate load must include the ultimate deflections and ultimate deformation induced by the loading. When analytical methods are used to show compliance with the ultimate load strength requirements, it must be shown that –

(1) The effects of deformation are not significant;

(2) The deformations involved are fully accounted for in the analysis; or

(3) The methods and assumptions used are sufficient to cover the effects of these deformations.

(f) Unless shown to be extremely improbable, the airplane must be designed to withstand any forced structural vibration resulting from any failure, malfunction or adverse condition in the flight control system. These loads must be treated in accordance with the requirements of CS 25.302.

CS 25.307 Proof of structure

(a) Compliance with the strength and deformation requirements of this Subpart must be shown for each critical loading condition. Structural analysis may be used only if the structure conforms to that for which experience has shown this method to be reliable. In other cases, substantiating tests must be made to load levels that are sufficient to verify structural behavior up to loads specified in CS 25.305.

(d) When static or dynamic tests are used to show compliance with the requirements of CS 25.305 (b) for flight structures, appropriate material correction factors must be applied to the test results, unless the structure, or part thereof, being tested has features such that a number of elements contribute to the total strength of the structure and the failure of one element results in the redistribution of the load through alternate load paths.

CS 25.321 General

(a) Flight load factors represent the ratio of the aerodynamic force component (acting normal to the assumed longitudinal axis of the airplane) to the weight of the airplane. A positive load factor is one in which the aerodynamic force acts upward with respect to the airplane.

(b) Considering compressibility effects at each speed, compliance with the flight load requirements of this Subpart must be shown –

(1) At each critical altitude within the range of altitudes selected by the applicant;

(2) At each weight from the design minimum weight to the design maximum weight appropriate to each particular flight load condition; and

(i) For each required altitude and weight, for any practicable distribution of disposable load within the operating limitations recorded in the Airplane Flight Manual.

CS 25.571 Damage-tolerance and fatigue evaluation of structure

(a) *General.* An evaluation of the strength, detail design, and fabrication must show that catastrophic failure due to fatigue, corrosion, or accidental damage, will be avoided throughout the operational life of the airplane. This evaluation must be conducted in accordance with the provisions of subparagraphs (b) and (e) of this paragraph, except as specified in subparagraph (c) of this paragraph, for each part of the structure which could contribute to a catastrophic failure (such as wing, empennage, control surfaces and their systems, the fuselage, engine mounting, landing gear, and their related primary attachments). For turbine engine powered airplanes, those parts which could contribute to a catastrophic failure must also be evaluated under subparagraph (d) of this paragraph. In addition, the following apply:

(1) Each evaluation required by this paragraph must include –

(i) The typical loading spectra, temperatures, and humidity expected in service;

(ii) The identification of principal structural elements and detail design points, the failure of which could cause catastrophic failure of the airplane; and

(iii) An analysis supported by test evidence, of the principal structural elements and detail design points identified in subparagraph (a) (1) (ii) of this paragraph.

(2) The service history of airplanes of similar structural design, taking due account of differences in operating conditions and procedures, may be used in the evaluations required by this paragraph.

(3) Based on the evaluations required by this paragraph, inspections or other procedures must be established as necessary to prevent catastrophic failure, and must be included in the Airworthiness Limitations Section of the Instructions for Continued Airworthiness required by CS 25.1529.

(b) Damage-tolerance (fail-safe) evaluation. The evaluation must include a determination of the probable locations and modes of damage due to fatigue, corrosion, or accidental damage. The determination must be by analysis supported by test evidence and (if available) service experience. Damage at multiple sites due to prior fatigue exposure must be included where the design is such that this type of damage can be expected to occur. The evaluation must incorporate repeated load and static analyses supported by test evidence. The extent of damage for residual strength evaluation at any time within the operational life must be consistent with the initial detectability and subsequent growth under repeated loads.

(c) Fatigue (safe life) evaluation. Compliance with the damage-tolerance requirements of subparagraph (b) of this paragraph is not required if the applicant establishes that their application for particular structure is impractical. This structure must be shown by analysis, supported by test evidence, to be able to withstand the repeated loads of variable magnitude expected during its service life without detectable cracks. Appropriate safe-life scatter factors must be applied.

<u>CS 25.603 Materials</u>

The suitability and durability of materials used for parts, the failure of which could adversely affect safety, must –

(a) Be established on the basis of experience or tests;

(b) Conform to approved specifications, that ensure their having the strength and other properties assumed in the design data and

(c) Take into account the effects of environmental conditions, such as temperature and humidity, expected in service.

CS 25.613 Material strength properties and Material Design Values

(a) Material strength properties must be based on enough tests of material meeting approved specifications to establish design values on a statistical basis.

(b) Material design values must be chosen to minimize the probability of structural failures due to material variability. Except as provided in subparagraphs (e) and (f) of this paragraph, compliance must be shown by selecting material design values which assure material strength with the following probability:

(1) Where applied loads are eventually distributed through a single member within an assembly, the failure of which would result in loss of structural integrity of the component, 99% probability with 95% confidence.

(2) For redundant structure, in which the failure of individual elements would result in applied loads being safely distributed to other load carrying members, 90% probability with 95% confidence.

CS 25.651 Proof of strength

(a) Limit load tests of control surfaces are required. These tests must include the horn or fitting to which the control system is attached.

(b) Compliance with the special factors requirements of CS 25.619 to 25.625 and 25.657 for control surface hinges must be shown by analysis or individual load tests.

7.1.3 AMC – Subpart C

AMC No. 2 to CS 25.301(b)

3. BACKGROUND

(c) CS-25 stipulates a number of load conditions, such as flight loads, ground loads, pressurization loads, inertia loads and engine/APU loads. CS 25.301 requires methods used to determine load intensities and distributions to be validated by flight load measurements unless the methods used for determining those loading conditions are shown to be reliable. Although this applies to all load conditions of CS-25, the scope of this AMC is limited to flight loads.

(d) The sizing of the structure of the aircraft generally involves a number of steps and requires detailed knowledge of air loads, mass, stiffness, damping, flight control system characteristics, etc. Each of these steps and items may involve its own validation. The scope of this AMC however is limited to validation of methods used for determination of loads intensities and distributions by flight load measurements.

(e) By reference to validation of "methods", CS 25.301(b) and this AMC are intended to convey a validation of the complete package of elements involved in the accurate representation of loads, including input data and analytical process. The aim is to demonstrate that the complete package delivers reliable or conservative calculated loads for scenarios relevant to CS-25 flight loads requirements.

(f) Some measurements may complement (or sometimes even replace) the results from theoretical methods and models. Some flight loads development methods such as those used to develop buffeting loads have very little theoretical foundation, or are methods based directly on flight loads measurements extrapolated to represent limit conditions.

4. NEED FOR AND EXTENT OF FLIGHT LOAD MEASUREMENTS

4.1. General

(g) The need for and extent of the flight load measurements has to be discussed and agreed between the Agency and Applicant on a case by case basis. Such an assessment should be based on:

a comparison of the design features of the airplane under investigation with previously developed (by the Applicant) and approved airplanes. New or significantly different design features should be identified and assessed.

the Applicant's previous experience in validating load intensities and distributions derived from analytical methods and/or wind tunnel tests. This experience should have been accumulated on previously developed (by the Applicant) and approved types and models of airplanes. The validation should have been by a flight load measurement program that was conducted by the Applicant and found acceptable to the Agency for showing compliance.

the sensitivity to parametric variation and continued applicability of the analytical methods and/or wind tunnel test data.

(h) Products requiring a new type certificate will in general require flight-test validation of flight loads methods unless the Applicant can demonstrate to the Agency that this is unnecessary. If the configuration under investigation is a similar configuration and size as a previously developed and approved design, the use of analytical methods, such as computational fluid dynamics validated on wind tunnel test results and supported by previous load validation flight test experience, may be sufficient to determine flight loads without further flight test validation.

(i) Applicants who are making a change to a Type Certificated airplane, but who do not have access to the Type certification flight loads substantiation for that airplane, will be required to develop flight loads analyses, as necessary, to substantiate the change. In general, the loads analyses will require validation and may require flight test loads measurements, as specified in this AMC.

(j) The Applicant is encouraged to submit supporting data or test plans for demonstrating the reliability of the flight loads methods early in the certification planning process.

4.3. Other considerations

(k) Notwithstanding the similarity of the airplane or previous load validation flight test experience of the Applicant, the local loads on the following elements are typically unreliably predicted and may require a measurement during flight tests:

Loads on high lift devices;

Hinge moments on control surfaces;

Loads on the empennage due to buffeting;

Loads on any unusual device.

5. FLIGHT LOAD MEASUREMENTS

5.1. Measurements.

Flight load measurements (for example, through application of strain gauges, pressure belts, accelerometers) may include:

Pressures / air loads /net shear, bending and torque on primary aerodynamic surfaces;

Flight mechanics parameters necessary to correlate the analytical model with flight test results;

High lift devices loads and positions;

Primary control surface hinge moments and positions;

Unsymmetrical loads on the empennage (due to roll/yaw maneuvers and buffeting);

Local strains or response measurements in cases where load calculations or measurements are indeterminate or unreliable.

5.2. Variation of parameters.

The test points for the flight loads measurements should consider the variation of the main parameters affecting the loads under validation. Examples of these parameters include: load factor, speeds, altitude, aircraft c.g., weight and inertia, power settings (thrust, for wing mounted engines), fuel loading, speed brake settings, flap settings and gear conditions (up/down) within the design limits of the airplane. The range of variation of these parameters must be sufficient to allow the extrapolation to the design loads conditions. In general, the flight test conditions need not exceed approximately 80% of limit load.

6. RESULTS OF FLIGHT LOAD MEASUREMENTS

6.1. Comparison / Correlation.

Flight loads are not directly measured, but are determined through correlation with measured strains, pressures or accelerations. The load intensities and distributions derived from flight testing should be compared with those obtained from analytical methods. The uncertainties in both the flight testing measurements and subsequent correlation should be carefully considered and compared with the inherent assumptions and capabilities of the process used in analytic derivation of flight loads. Since in most cases the flight test points are not the limit design load conditions, new analytical load cases need to be generated to match the actual flight test data points.

6.2. Quality of measurements.

Factors which can affect the uncertainty of flight loads resulting from calibrated strain gauges include the effects of temperature, structural non-linearity's, establishment of flight/ground zero reference, and large local loads, such as those resulting from the propulsion system installation, landing gear, flap tracks or actuators. The static or dynamic nature of the loading can also affect both strain gauge and pressure measurements.

AMC 25.307

3. DEFINITIONS

- *Detail*. A structural element of a more complex structural member (e.g. joints, splices, stringers, stringer run-outs, or access holes).
- *Sub Component*. A major three-dimensional structure which can provide complete structural representation of a section of the full structure (e.g., stub-box, section of a spar, wing panel, wing rib, body panel, or frames).
- *Component*. A major section of the airframe structure (e.g., wing, body, fin, horizontal stabilizer) which can be tested as a complete unit to qualify the structure.
- *Full Scale*. Dimensions of test article are the same as design; fully representative test specimen (not necessarily complete airframe).
- *New Structure*. Structure for which behavior is not adequately predicted by analysis supported by previous test evidence. Structure that utilizes significantly different structural design concepts such as details, geometry, structural arrangements, and load paths or materials from previously tested designs.

4. INTRODUCTION

As required by subparagraph (a) of CS 25.307, the structure must be shown to comply with the strength and deformation requirements of Subpart C of CS-25. This means that the structure must:

(l) be able to support limit loads without detrimental permanent deformation, and:

(m) be able to support ultimate loads without failure.

This implies the need of a comprehensive assessment of the external loads (addressed by CS 25.301), the resulting internal strains and stresses, and the structural allowable. CS 25.307 requires compliance for each critical loading condition. Compliance can be shown by analysis supported by previous test evidence, analysis supported by new test evidence or by test only. As compliance by test only is impractical in most cases, a large portion of the substantiating data will be based on analysis.

There are a number of standard engineering methods and formulas which are known to produce acceptable, often conservative results especially for structures where load paths are well defined. Those standard methods and formulas, applied with a good understanding of their limitations, are considered reliable analyses when showing compliance with CS 25.307. Conservative assumptions may be considered in assessing whether or not an analysis may be accepted without test substantiation.

The application of methods such as Finite Element Method or engineering formulas to complex structures in modern aircraft is considered reliable only when validated by full scale tests (ground and/or flight tests). Experience relevant to the product in the utilization of such methods should be considered.

5. CLASSIFICATION OF STRUCTURE

(n) The structure of the product should be classified into one of the following three categories:

New Structure
Similar New Structure
Derivative/Similar Structure

(o) Justifications should be provided for classifications other than New Structure. Elements that should be considered are :

i. The accuracy/conservatism of the analytical methods, and

ii. Comparison of the structure under investigation with previously tested structure.

Considerations should include, but are not limited to the following:

external loads (bending moment, shear, torque , etc.);
internal loads (strains, stresses, etc.);
structural design concepts such as details, geometry, structural arrangements, load paths ;
materials ;
test experience (load levels achieved, lessons learned);
deflections ;
deformations ;
extent of extrapolation from test stress levels.

6. NEED AND EXTENT OF TESTING

The following factors should be considered in deciding the need for and the extent of testing including the load levels to be achieved:

(p) The classification of the structure (as above);

(q) The consequence of failure of the structure in terms of the overall integrity of the airplane;

(r) The consequence of the failure of interior items of mass and the supporting structure to the safety of the occupants.

7. CERTIFICATION APPROACHES

(a) Analysis, supported by new strength testing of the structure to limit and ultimate load. This is typically the case for New Structure. Substantiation of the strength and deformation requirements up to limit and ultimate loads normally requires testing of sub-components, full scale components or full scale tests of assembled components (such as a nearly complete airframe). The entire test program should be considered in detail to assure the requirements for strength and deformation can be met up to limit load levels as well as ultimate load levels. Sufficient limit load test conditions should be performed to verify that the structure meets the deformation requirements of CS 25.305(a) and to provide validation of internal load distribution and analysis predictions for all critical loading conditions. Because ultimate load tests often result in significant permanent deformation, choices will have to be made with respect to the load conditions applied. This is usually based on the number of test specimens available, the analytical static strength margins of safety of the structure and the range of supporting detail or sub-component tests. An envelope approach may be taken, where a combination of different load cases is applied, each one critical for a different section of the structure. These limit and ultimate load tests may be supported by detail and sub-component tests that verify the design allowable (tension, shear, compression) of the structure and often provide some degree of validation for ultimate strength.

(b) Analysis validated by previous test evidence and supported with additional limited testing. This is typically the case for Similar New Structure. The extent of additional limited testing (number of specimens, load levels, etc.) will depend upon the degree of change, relative to the elements of paragraphs 5(b)(i) and (ii). For example, if the changes to an existing design and analysis necessitate extensive changes to an existing test-validated finite element model (e.g. different rib spacing) additional testing may be needed. Previous test evidence can be relied upon whenever practical. These additional limited tests may be further supported by detail and sub-component tests that verify the design allowable (tension, shear, compression) of the structure and often provide some degree of validation for ultimate strength.

(c) *Analysis, supported by previous test evidence*. This is typically the case for Derivative/ Similar Structure. Justification should be provided for this approach by demonstrating how the previous static test evidence validates the analysis and supports showing compliance for the structure under investigation. Elements that need to be considered are those defined in paragraphs 5(b)(i) and (ii). For example, if the changes to the existing design and test-validated analysis are evaluated to assure they are relatively minor and the effects of the changes are well understood, the original tests may provide sufficient validation of the analysis and further testing may not be necessary. For example, if a weight increase results in higher loads along with a corresponding increase in some of the element thickness and fastener sizes, and materials and geometry (overall configuration, spacing of structural members, etc.) remain generally the same, the revised analysis could be considered reliable based on the previous validation.

(d) Test only. Sometimes no reliable analytical method exists, and testing must be used to show compliance with the strength and deformation requirements. In other cases it may be elected to show compliance solely by tests even if there are acceptable analytical methods. In either case, testing by itself can be used to show compliance with the strength and deformation requirements of CS-25 Subpart C. In such cases, the test load conditions should be selected to assure all critical design loads are encompassed. If tests only are used to show compliance with the strength and deformation requirements for single load path structure which carries flight loads (including pressurization loads), the test loads must be increased to account for variability in material properties, as required by CS 25.307(d). In lieu of a rational analysis, for metallic materials, a factor of 1.15 applied to the limit and ultimate flight loads may be used. If the structure has multiple load paths, no material correction factor is required.

8. INTERPRETATION OF DATA

The interpretation of the substantiation analysis and test data requires an extensive review of:

he representativeness of the loading ;

the instrumentation data ;

comparisons with analytical methods ;

representativeness of the test article(s) ;
test set-up (fixture, load introductions) ;
load levels and conditions tested ;
test results.

Testing is used to validate analytical methods except when showing compliance by test only. If the test results do not correlate with the analysis, the reasons should be identified and appropriate action taken. This should be accomplished whether or not a test article fails below ultimate load.

Should a failure occur below ultimate load, an investigation should be conducted for the product to reveal the cause of this failure. This investigation should include a review of the test specimen and loads, analytical loads, and the structural analysis. This may lead to adjustment in analysis/modeling techniques and/or part redesign and may result in the need for additional testing. The need for additional testing to ensure ultimate load capability depends on the degree to which the failure is understood and the analysis can be validated by the test.

AMC 25.571

(s) 1 INTRODUCTION

1.1 The contents of this AMC are considered by the Agency in determining compliance with the damage-tolerance and fatigue requirements of CS 25.571.

1.1.1 Although a uniform approach to the evaluation required by CS 25.571 is desirable, it is recognized that in such a complex field new design features and methods of fabrication, new approaches to the evaluation, and new configurations could necessitate variations and deviations from the procedures described in this AMC.

1.1.2 Damage-tolerance design is required, unless it entails such complications that an effective damage-tolerant structure cannot be achieved within the limitations of geometry, inspectability, or good design practice. Under these circumstances, a design that complies with the fatigue evaluation (safe life) requirements is used. Typical examples of structure that might not be conducive to damage tolerance design are landing gear, engine mounts, and their attachments.

1.1.3 Experience with the application of methods of fatigue evaluation indicates that a test background should exist in order to achieve the design objective. Even under the damage-tolerance method discussed in paragraph 2, 'Damage-tolerance (fail-safe) evaluation', it is the general practice within industry to conduct damage-tolerance tests for design information and guidance purposes. Damage location and growth data should also be considered in establishing a recommended inspection programme.

1.1.4 Assessing the fatigue characteristics of certain structural elements, such as major fittings, joints, typical skin units, and splices, to ensure that the anticipated service life can reasonably be attained, is needed for structure to be evaluated under CS 25.571(c).

1.2 Typical Loading Spectra Expected in Service. The loading spectrum should be based on measured statistical data of the type derived from government and industry load history studies and, where insufficient data are available, on a conservative estimate of the anticipated use of the loads (gust and maneuver), ground loads (taxiing, landing impact, turning, engine run up, braking, and towing) and pressurization loads. The development of the loading spectrum includes the definition of the expected flight plan which involves climb, cruise, descent, flight times, operational speeds and altitudes, and the approximate time to be spent in each of the operating regimes. Operations for crew training, and other pertinent factors, such as the dynamic stress characteristics of any flexible structure excited by turbulence, should also be considered. For pressurized cabins, the loading spectrum would include the repeated application of the normal operating differential pressure, and the super imposed effects of flight loads and external aerodynamic pressures.

1.3 *Components to be evaluated*. In assessing the possibility of serious fatigue failures, the design should be examined to determine probable points of failure in service. In this examination, consideration should be given, as necessary, to the results of stress analyses, static tests, fatigue tests, strain gauge surveys, tests of similar structural configurations, and service experience. Service experience has shown that special attention should be focused on the design details of important discontinuities, main attachment fittings, tension joints, splices, and cutouts such as windows, doors and other openings. Locations prone to accidental damage (such as that due to impact with ground servicing equipment near airplane doors) or to corrosion should also be considered.

1.4 Analyses and Tests. Unless it is determined from the foregoing examination that the normal operating stresses in specific regions of the structure are of such a low order that serious damage growth is extremely improbable, repeated load analyses or tests should be conducted on structure representative of components or sub-components of the wing, control surfaces, empennage, fuselage, landing gear, and their related primary attachments. Test specimens should include structure representative of attachment fittings, major joints, changes in section, cutouts, and discontinuities. Any method used in the analyses should be supported, as necessary, by test or service experience. Generally it will be required to substantiate the primary structure against the provisions of CS 25.571(b) and (c) by representative testing. The nature and extent of tests on complete structures or on portions of the primary structure will depend upon applicable previous design and structural tests, and service experience with similar structures. The scope of the analyses and supporting test programs should be agreed with the Agency.

1.5 Repeated Load Testing. In the event of any repeated load testing necessary to support the damage tolerance or safe-life objectives of CS 25.571(b) and (c) respectively not being concluded at the issuance of type certificate, at least one year of safe operation should be substantiated at the time of certification. In order not to invalidate the certificate of airworthiness the fatigue substantiation should stay sufficiently ahead of the service exposure of the lead airplane.

2 DAMAGE-TOLERANCES (FAIL-SAFE) EVALUATION

2.1 General. The damage-tolerance evaluation of structure is intended to ensure that should serious fatigue, corrosion, or accidental damage occur within the operational life of the airplane, the remaining structure can withstand reasonable loads without failure or excessive structural deformation until the damage is detected. Included are the considerations historically associated with fail-safe design. The evaluation should encompass establishing the components which are to be designed as damage-tolerant, defining the loading conditions and extent of damage, conducting sufficient representative tests and/or analyses to substantiate the design objectives (such as life to crack-initiation, crack propagation rate and residual strength) have been achieved and establishing data for inspection programs to ensure detection of damage. Interpretation of the test results should take into account the scatter in crack propagation rates as well as in lives to crack-initiation. Test results should be corrected to allow for variations between the specimen and the airplane component thickness and sizes. This evaluation applies to either single or multiple load path structure.

2.1.1 Design features which should be considered in attaining a damage-tolerant structure include the following:

a. Multiple load path construction and the use of crack stoppers to control the rate of crack. growth, and to provide adequate residual static strength;
b. Materials and stress levels that, after initiation of cracks, provide a controlled slow rate of crack propagation combined with high residual strength. For single

load path discrete items, such as control surface hinges, wing spar joints or stabilizer pivot fittings the failure of which could be catastrophic, it should be clearly demonstrated that cracks starting from material flaws, manufacturing errors or accidental damage (including corrosion) have been properly accounted for in the crack propagation estimate and inspection method;

c. Arrangement of design details to ensure a sufficiently high probability that a failure in any. critical structural element will be detected before the strength has been reduced below the level necessary to withstand the loading conditions specified in CS 25.571(b) so as to allow replacement or repair of the failed elements; and
d. Provisions to limit the probability of concurrent multiple damage, particularly after long service, which could conceivably contribute to a common fracture path. The achievement of this would be facilitated by ensuring sufficient life to crack-initiation. Examples of such multiple damage are
   i. A number of small cracks which might coalesce to form a single long crack;
   ii. Failures, or partial failures, in adjacent areas, due to the redistribution of loading following a failure of a single element; and
   iii. Simultaneous failure, or partial failure, of multiple load path discrete elements, working at similar stress levels.

In practice it may not be possible to guard against the effects of multiple damages and fail-safe substantiation may be valid only up to a particular life which would preclude multiple damages.

e. The airplane may function safely with an element missing. This feature would be admitted only, provided its separation will not prevent continued safe flight and landing and the probability of occurrence is acceptably low.

2.1.2 In the case of damage which is readily detectable within a short period (50 flights, say) for which CS 25.571(b) allows smaller loads to be used, this relates to damage which is large enough to be detected by obvious visual indications during walk around, or by indirect means such as cabin pressure loss, cabin noise, or fuel leakage. In such instances and in the absence of a probability approach the residual load levels except for the trailing edge flaps may be reduced to not less than the following:

a. The maximum normal operating differential pressure (including the expected external aerodynamic pressures under 1g level flight) multiplied by a factor of 1·10 omitting other loads.

b. 85% of the limit flight maneuver and ground conditions of CS 25.571(b)(1) to (6) inclusive, excluding (5)(ii) and separately 75% of the limit gust velocities (vertical or lateral) as specified at speeds up to VC in CS 25.571(b)(2) and (b)(5)(i). On the other hand if the probability approach is used the residual load levels may not in any case be lower than the values given in paragraph 2.7.2 of this AMC for one flight exposure. In the case where fatigue damage is arrested at a readily detectable size following rapid crack growth or a sudden load path failure under the application of high loads, the structure must be able to withstand the loads defined in CS 25.571(b)(1) to (6) inclusive up to that size of damage. For the subsequent growth of that damage, lower loads as stated above may be used.

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2.2 Identification of Principal Structural Elements. Principal structural elements are those which contribute significantly to carrying flight, ground, and pressurization loads, and whose failure could result in catastrophic failure of the airplane. Typical examples of such elements are as follows:

2.2.1 Wing and empennage

a. Control surfaces, slats, flaps and their attachment hinges and fittings;
b. Integrally stiffened plates;
c. Primary fittings;
d. Principal splices;
e. Skin or reinforcement around cutouts or discontinuities;
f. Skin-stringer combinations;
g. Spar caps; and
h. Spar webs.

2.2.2 Fuselage

a. Circumferential frames and adjacent skin;
b. Door frames;
c. Pilot window posts;
d. Pressure bulkheads;
e. Skin and any single frame or stiffener element around a cutout;
f. Skin or skin splices, or both, under circumferential loads;
g. Skin or skin splices, or both, under fore-and-aft loads;
h. Skin around a cutout;
i. Skin and stiffener combinations under fore-and-aft loads; and

j. Window frames.

2.3 Extent of Damage. Each particular design should be assessed to establish appropriate damage criteria in relation to inspectability and damage-extension characteristics. In any damage determination, including those involving multiple cracks, it is possible to establish the extent of damage in terms of detectability with the inspection techniques to be used, the associated initially detectable crack size, the residual strength capabilities of the structure, and the likely damage-extension rate considering the expected stress redistribution under the repeated loads expected in service and with the expected inspection frequency. Thus, an obvious partial failure could be considered to be the extent of the damage or residual strength assessment, provided a positive determination is made that the fatigue cracks will be detectable by the available inspection techniques at a sufficiently early stage of the crack development. In a pressurized fuselage, an obvious partial failure might be detectable through the inability of the cabin to maintain operating pressure or controlled decompression after occurrence of the damage. The following are typical examples of partial failures which should be considered in the evaluation:

2.3.1 Detectable skin cracks emanating from the edge of structural openings or cutouts;

2.3.2 A detectable circumferential or longitudinal skin crack in the basic fuselage structure;

2.3.3 Complete severance of interior frame elements or stiffeners in addition to a detectable crack in the adjacent skin;

2.3.4 A detectable failure of one element where dual construction is utilized in components such as spar caps, window posts, window or door frames, and skin structure;

2.3.5 The presence of a detectable fatigue failure in at least the tension portion of the spar web or similar element; and

2.3.6 The detectable failure of a primary attachment, including a control surface hinge and fitting.

2.4 Inaccessible Areas. Every reasonable effort should be made to ensure inspectability of all structural parts, and to qualify them under the damage-tolerance provisions. In those cases where inaccessible and uninspectable blind areas exist and suitable damage tolerance cannot practically be provided to allow for extension of damage into detectable areas, the structure should be shown to comply with the fatigue (safe-life) requirements in order to ensure its continued airworthiness. In this respect particular attention should be given to the effects of corrosion.

2.5 Testing of Principal Structural Elements. The nature and extent of tests on complete structures or on portions of the primary structure will depend upon applicable previous design, construction, tests, and service experience, in connection with similar structures. Simulated cracks should be as representative as possible of actual fatigue damage. Where it is not practical to produce actual fatigue cracks, damage can be simulated by cuts made with a fine saw, sharp blade, guillotine, or other suitable means. In those cases where bolt failure, or its equivalent, is to be simulated as part of a possible damage configuration in joints or fittings, bolts can be removed to provide that part of the simulation, if this condition would be representative of an actual failure under typical load. Where accelerated crack propagation tests are made, the possibility of creep cracking under real time pressure conditions should be recognized especially as the crack approaches its critical length.

2.6 Identification of Locations to be evaluated. The locations of damage to structure for damage-tolerances evaluation should be identified as follows:

2.6.1 Determination of General Damage Locations. The location and modes of damage can be determined by analysis or by fatigue tests on complete structures or subcomponents. However, tests might be necessary when the basis for analytical prediction is not reliable, such as for complex components. If less than the complete structure is tested, care should be taken to ensure that the internal loads and boundary conditions are valid. Any tests should be continued sufficiently beyond the expected service life to ensure that, as far as practicable, the likely locations and extent of crack initiation are discovered.

a. If a determination is made by analysis, factors such as the following should be taken into a. account:
   i. Strain data on undamaged structure to establish points of high stress concentration as well as. the magnitude of the concentration;
   ii. Locations where permanent deformation occurred in static tests;
   iii. Locations of potential fatigue damage identified by fatigue analysis; and
   iv. Design details which service experience of similarly designed components indicate are prone to fatigue or other damage.
b. In addition, the areas of probable damage from sources such as corrosion, disbonding, accidental damage or manufacturing defects should be determined from a review of the design and past service experience.

2.6.2 Selection of Critical Damage Areas. The process of actually locating where damage should be simulated in principal structural elements identified in paragraph 2.2 of this AMC should take into account factors such as the following:

a. Review analysis to locate areas of maximum stress and low margin of safety;
b. Selecting locations in an element where the stresses in adjacent elements would be the maximum with the damage present;
c. Selecting partial fracture locations in an element where high stress concentrations are present in the residual structure; and
d. Selecting locations where detection would be difficult.

2.7 Damage-tolerance Analysis and Tests. It should be determined by analysis, supported by test2.7 evidence, that the structure with the extent of damage established for residual strength evaluation can withstand the specified design limit loads (considered as ultimate loads), and that the damage growth rate under the repeated loads expected in service (between the time at which the damage becomes initially detectable and the time at which the extent of damage reaches the value for residual strength evaluation) provides a practical basis for development of the inspection programme and procedures described in paragraph 2.8 of this AMC. The repeated loads should be as defined in the loading, temperature, and humidity spectra. The loading conditions should take into account the effects of structural flexibility and rate of loading where they are significant.

2.7.1 The damage-tolerance characteristics can be shown analytically by reliable or conservative methods such as the following:

a. By demonstrating quantitative relationships with structure already verified as damage tolerant;
b. By demonstrating that the damage would be detected before it reaches the value for residual strength evaluation; or
c. By demonstrating that the repeated loads and limit load stresses do not exceed those of previously verified designs of similar configuration, materials and inspectability.

2.7.2 The maximum extent of immediately obvious damage from discrete sources should be determined and the remaining structure shown to have static strength for the maximum load (considered as ultimate load) expected during the completion of the flight. In the absence of a rational analysis the following ultimate loading conditions should be covered:

a. At the time of the incident:

i. The maximum normal operating differential pressure (including the expected external aerodynamic pressures during 1 g level flight) multiplied by a factor 1·1 combined with 1 g flight loads.

ii. The airplane, assumed to be in 1g level flight should be shown to be able to survive the over swing condition due to engine thrust asymmetry and pilot corrective action taking into account any damage to the flight controls which it is presumed the airplane has survived.

b. Following the incident: 70% limit flight maneuver loads and, separately, 40% of the limit gust velocity (vertical or lateral) as specified at VC up to the maximum likely operational speed following failure, each combined with the maximum appropriate cabin differential pressure (including the expected external aerodynamic pressures). Further, any loss in structural stiffness which might arise should be shown to result in no dangerous reduction in freedom from flutter up to speed VC/MC.

2.8 Inspection. Detection of damage before it becomes dangerous is the ultimate control in ensuring the damage-tolerance characteristics of the structure. Therefore, the applicant should provide sufficient guidance information to assist operators in establishing the frequency, extent, and methods of inspection of the critical structure, and this kind of information must, under CS 25.571(a) (3), be included in the maintenance manual required by CS 25.1529. Due to the inherent complex interactions of the many parameters affecting damage tolerance, such as operating practices, environmental effects, load sequence on crack growth, and variations in inspection methods, related operational experience should be taken into account in establishing inspection procedures. It is extremely important to ensure by regular inspection the detection of damage in areas vulnerable to corrosion or accidental damage. However for crack initiation arising from fatigue alone, the frequency and extent of the inspections may be reduced during the period up to the demonstrated crack-free life of the part of the structure, including appropriate scatter factors (see paragraph 3.2). Comparative analysis can be used to guide the changes from successful past practice when necessary. Therefore, maintenance and inspection requirements should recognize the dependence on experience and should be specified in a document that provides for revision as a result of operational experience, such as the one containing the Manufacturers Recommended Structural Inspection Programme.

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