

Article

Virtual Full Scale Static Test of a Civil Tilt Rotor Composite Wing in Non-Linear Regime

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Abstract: This study addresses the crucial role of post-buckling behavior analysis in the structural design of composite aeronautical structures. Traditional engineering practices tend to result in oversized composite components, increasing structural weight. EASA AMC 20-29's Building Block Approach suggests phased testing, but its time and cost challenges necessitate a shift to high-fidelity post-buckling analyses, exemplified by MSC NASTRAN SOL 400. This approach, showcased in the analysis of the Next Generation Civil Tilt Rotor Technology Demonstrator's wing (NGTCTR-TD), effectively de-risks static tests, contributing to a more efficient certification process. The study demonstrates how advanced simulations provide detailed insights into local buckling phenomena, allowing precise stress distribution analysis. These analyses eliminate the risk of structural failure, paving the way for safer, more efficient, and cost-effective airframe structures. Future developments aim to validate numerical analyses with experimental data, further emphasizing the reliability and benefits of high-fidelity simulations.

Keywords: full scale static; tiltrotor; wing; non-linear analysis



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1. Introduction

The analysis of post-buckling behavior in composite structures is a critical aspect of structural design in the aeronautics field. Conservative approaches, which are usually used in engineering, tend to oversize composite components to ensure buckling resistance. This results in much heavier structures than would be the case if the true post-buckling capacity of the structure were fully utilized. Budiansky's work [1] contributes to the theoretical foundation of post-buckling behavior in elastic structures. This classic text outlines fundamental principles, providing a theoretical framework for subsequent research in the field. Dickson et al. [2] contribute to the design methodologies for stiffened composite panels in the post-buckling range. The chapter provides practical strategies for designing composite structures that consider post-buckling behavior, enhancing the reliability and efficiency of aerospace structures. Knight and Starnes [3] explore the post buckling behavior of curved stiffened graphite-epoxy panels under axial compression. The study contributes valuable empirical data, shedding light on the behavior of specific composite structures subjected to compressive loads. Davis [4] conducts a detailed analysis and test correlation of a stiffened composite wing pane. The paper bridges theoretical simulations with practical tests, validating numerical methodologies against real-world scenarios and contributing to the accuracy of post-buckling simulations. Chen and Virgin [5] delve into the finite element analysis of post-buckling dynamics in plates, employing an asymptotic approach. Their work provides insights into the mathematical modeling of post-buckling

phenomena, offering a foundation for understanding the structural behavior beyond the critical buckling point. Cunha et al. [6] propose a robustness-based design strategy for composite structures. The paper introduces a methodology that considers the uncertainties in material properties, manufacturing processes, and operating conditions, contributing to the development of reliable and resilient composite structures. Wysmulski [7] focuses on the analysis of buckling and post-buckling in compressed composite columns. The paper offers insights into the behavior of composite structures under compression, with implications for design methodologies and structural integrity. Wang et al. [8] present a detailed analysis of buckling and post-buckling phenomena in a composite wing box subjected to loads with torsion-bending coupling. The study explores the complex interactions in large-scale structures, providing crucial insights for design considerations. Wu et al. [9] investigate the compressive buckling and post-buckling behaviors of J-type composite stiffened panels before and after impact load. The paper addresses the dynamic aspects of post-buckling, considering the effects of impact, and provides insights into structural resilience. EASA AMC 20-29 [10] provides standards for demonstrating compliance with airworthiness requirements, emphasizing the importance of robust methodologies in structural design and analysis. It addresses aspects crucial to ensuring the structural integrity and safety of aircraft, aligning with EASA's regulatory framework for aviation. The Building Block Approach in EASA AMC 20-29 involves a phased testing strategy for complex or large aircraft structures. It recommends a series of tests at different levels, allowing for incremental assessment. This approach aims to manage the certification and commercialization process efficiently, balancing the need for comprehensive testing with considerations of time and cost. It ensures that each phase builds on the knowledge gained from the previous one, providing a systematic and reliable method for certifying the structural integrity of aircraft components. Despite its reliability, the building block approach is very time consuming and expensive. The need for high-fidelity post-buckling analyses in airframe structures is paramount to reducing costs associated with expensive full-scale tests. High-fidelity post-buckling analyses enable a more detailed understanding of structural behavior, allowing for targeted and cost-effective design improvements without relying solely on extensive and expensive physical tests. Moreover, numerical simulations facilitate an iterative design process, where engineers can quickly assess multiple design alternatives and refine them based on simulated post-buckling behavior. This iterative approach is more efficient than relying solely on physical testing for design optimization. High-fidelity analyses can identify potential post-buckling issues in the design phase and allow for parametric studies to explore a wide range of design variables and loading conditions, enabling corrective measures to be implemented early. This proactive approach helps avoid costly modifications and retesting later in the development process. Finally, advanced simulations provide a wealth of data that can be used to support the certification process and assist in planning physical tests more strategically. Regulatory authorities, such as EASA, increasingly recognize the value of validated numerical simulations, potentially expediting the certification process: for example, engineers can use simulation results to guide the selection of critical test parameters, reducing the number of tests needed and focusing on areas where physical validation is essential.

Accurate post-buckling analyses provide insights into safety margins, allowing for more precise determination of design limits. This knowledge contributes to safer designs without overengineering, optimizing the balance between safety and cost.

In this work, the high fidelity post buckling analysis was set up in order to de-risk the static test of the composite wing of the Next Generation Civil Tilt Rotor Technology Demonstrator. Among the available commercial software to study the behavior of airframe structures, MSC NASTRAN SOL 400 [11,12] allows nonlinear analysis capabilities, both from a material and geometrical point of view.

The static test of the NGCTR-TD wing is a crucial test to obtain the Permit to Fly. The wing of NGCTR-TD is an innovative composite structure, characterized by a highly integrated concept for the upper skin and spars, featuring a curved spar to enhance fuel

capacity. The wing was designed as to have proper torsional and flexural stiffness to comply with aeroelastic requirements [13]; this aeroelastic tailoring is common with other authors [14]. For the purpose of the static test, a number of test loading conditions were selected among the hundreds of thousands coming from the Design Limit Loads Book, making a reduced set of loads to be tested at limit and ultimate loads conditions. A couple of testing loads, one coming from flight loads and the other coming from ground loads, were selected for the Ultimate load test. For this couple of conditions, a post buckling analysis has been performed in order to assess the wing behavior beyond more than 1.5 times the limit loads. Starting from the detailed FEM developed for the linear buckling analyses (SOL 105), a series of modifications were performed in order to allow the nonlinear calculation to be performed. In addition to geometric nonlinearities, material nonlinearities were also considered in order to obtain a more realistic behavior of the structure. The results of the parts considered most critical for each loading condition are reported.

2. Materials and Methods

2.1. Wing Design Description

The wing structure was built in full compliance with airworthiness requirements and technical specifications for helicopters and tiltrotors. Basically, the structure features three spars (multi spar structure) consisting of solid composite laminates and sandwich structures. Other parts were made of metal-based alloys. The ultimate design was developed taking into strong consideration especially the requirements of buckling resistance and stiffness, trying through a series of optimization processes to achieve the maximum results with the minimum possible structural mass [15]. All this is because weight is one of the main constraints of aircraft structures, so the design philosophy is to achieve high-performance structures that are as light as possible.

A most critical issue in the design phase of the wing involved achieving torsional stiffness standards, as the structure includes a series of access panels on the lower skin that worsen its torsional response. The design type used presents an excellent compromise between space available for the installation of the various control equipment and systems and the maximum strength achievable from the wing skin.

In Figure 1 is reported the wing structure Digital Mock-Up (DMU).

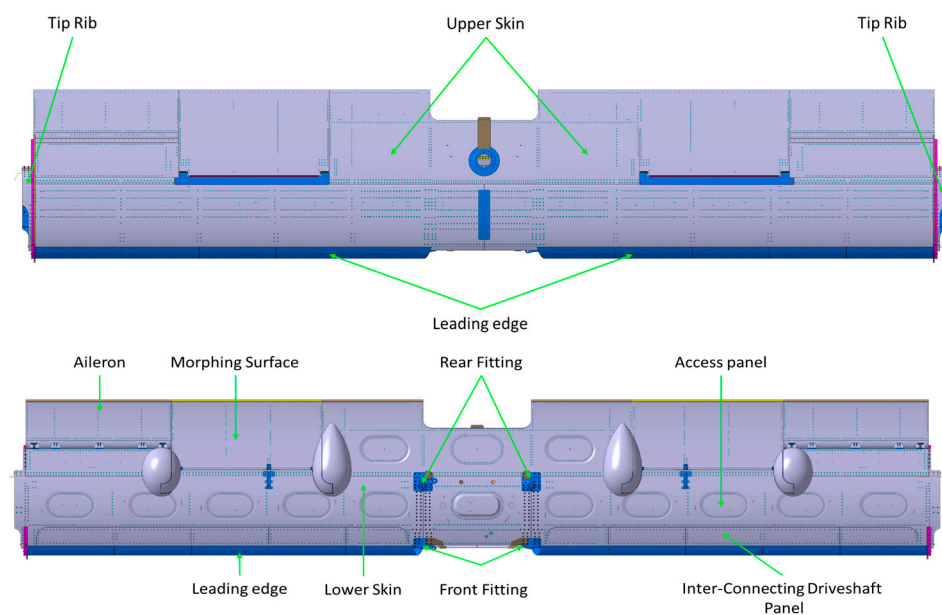


Figure 1. Cont.

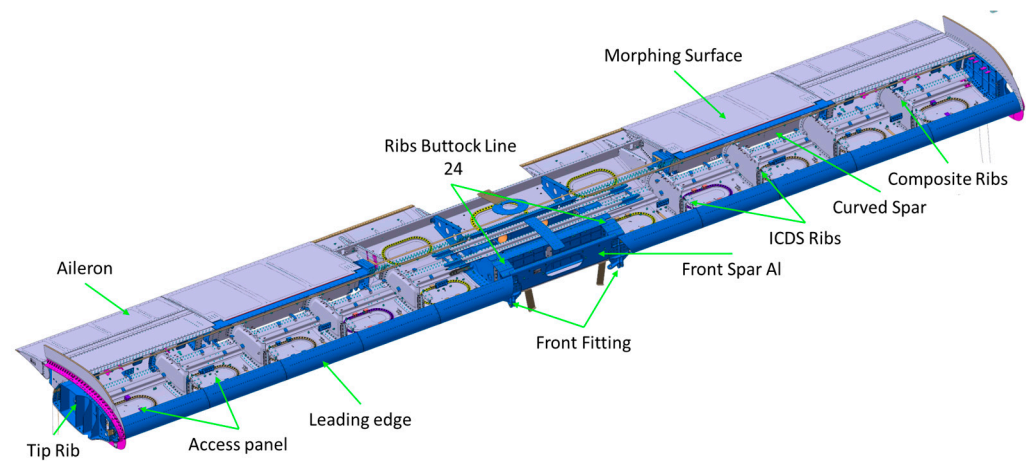


Figure 1. Wing structure description.

2.2. Experimental Environment

Aircraft are designed to withstand various types of loads, including aerodynamic loads, inertia loads, and operational loads (Figure 2). Aerodynamic loads result from dynamic pressure and encompass forces such as wing lift and drag, as well as moments like wing torsion and bending. Inertia loads arise from the acceleration of aircraft components.

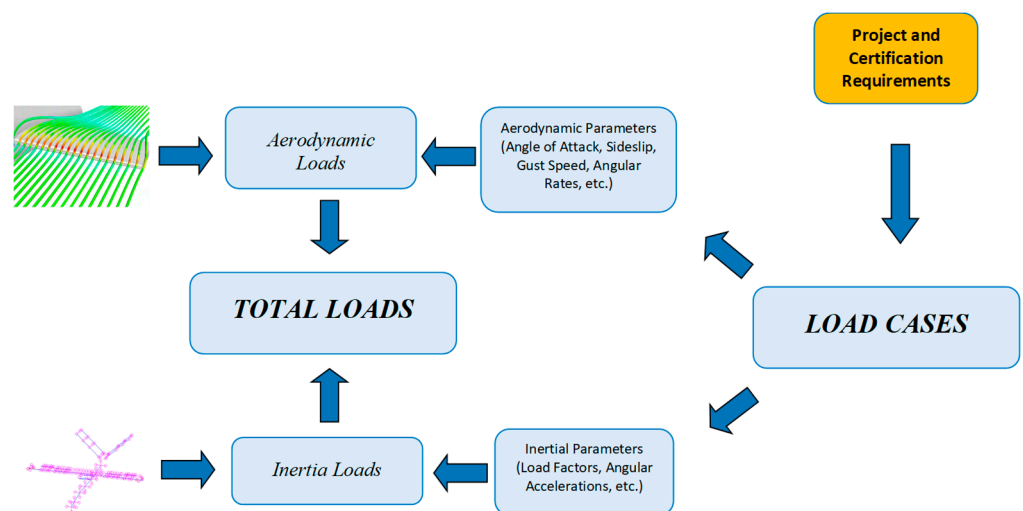


Figure 2. Total Loads Acting on the structure.

Operational loads, distinct from aerodynamics and inertia, stem from the aircraft's use. The airframe's primary role is crucial, and the location and shape of major load paths significantly influence weight. Structural engineers must design the structure to handle anticipated operational loads, ensuring it is not overdesigned. An aircraft stronger than necessary will be heavier and carry excess weight during each flight, impacting the useful load. The wings, stabilizing surfaces, engines, and landing gear also influence weight and the center of gravity, potentially requiring solutions such as heavy ballast. Efficiently managing these factors is essential for successful aircraft design.

The selection of critical cases to be included in a static test of an aircraft wing is a fundamental process in ensuring that the wing structure is adequately tested for the most extreme conditions it may encounter throughout its lifecycle.

For the NGCTR-TD wing, a test rig is designed in order to properly apply the loads and to faithfully replicate the boundary conditions (Figure 3): the wing-fuselage interface is constituted by the same production components (fittings and links) whereas the load path at the interface between the Nacelle Primary Structures (NPS) and the wing, is reproduced

by means of a welded structure, named “dummy NPS” which was designed having a stiffness which safeguards the bolts load distribution at the interfaces with the wing tip rib. In lieu of physical counterbalancing, the rig’s tare will be removed by the controlled adjustment of the load actuators before testing. A thorough correlation in terms of stiffness and load distribution at the interface has been carried out between the NPS of the actual aircraft, constructed with specific shapes to accommodate the transmission group and the engine using high-quality metals, and the “Dummy NPS” made with simplified shapes and less premium metals. The objective was achieved through several iterations of correlation by adjusting the thicknesses and elasticity of the metals in the “Dummy NPS”.

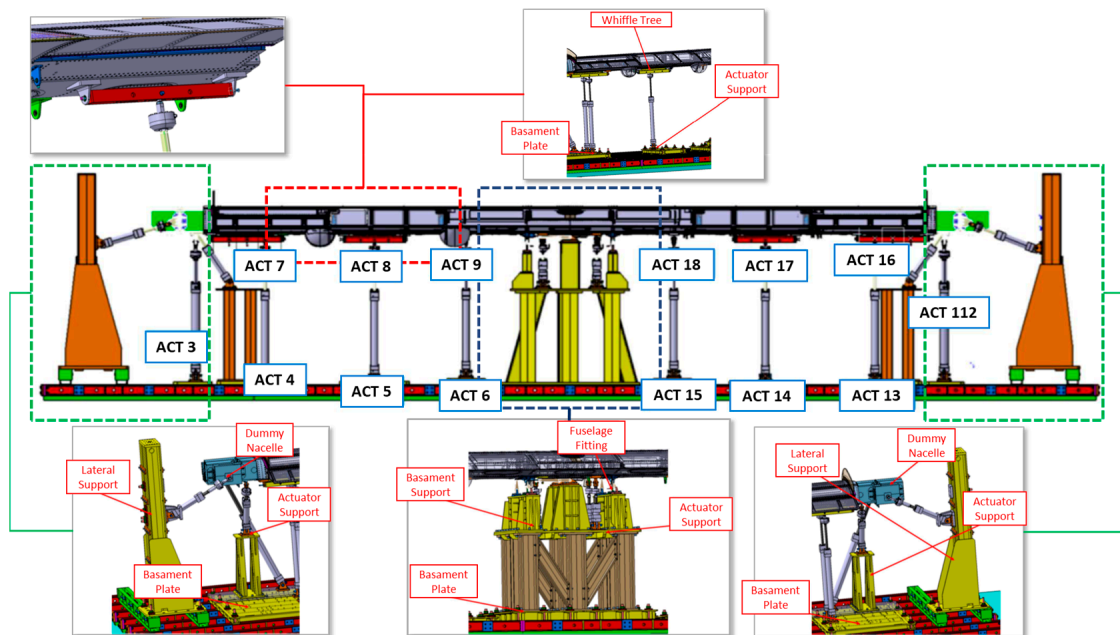


Figure 3. Overview of NGCTR-TD Test Rig.

During the static test of the wing, specific safety factors inherent to composite materials will be employed to ensure that the structure can withstand stresses and loads, preventing any failures throughout its operational life cycle.

Given the need to ensure high motorcycle safety standards and the difficulty with which it is possible to predict the failure of a part made of composite material, it was necessary to use a series of multiplicative factors by which the loads acting on the structure were amplified. With these factors, the possibility of degradation of the composite material due to different operational scenarios of the wing was also taken into account.

Design Safety Factor: Represents a multiplying factor applied according to regulation CS-29.303 to the maximum loads anticipated during the design phase, as stipulated by aviation regulations.

- The design safety factor is 1.50.

Environmental Effects Safety Factor: Composite materials are sensitive to environmental effects such as humidity and temperature. It is also important to consider these effects over time with the deterioration of the composite. Two environmental safety factors are taken into account:

- The safety factor for cold load condition = 1.10 (according to background of manufacturer of composite materials).
- The safety factor for hot load condition = 1.146 (according to background of manufacturer of composite materials).

Manufacturing Scatter Factor: since the production process can introduce variability, a safety factor may be included to compensate for possible defects or imperfections during production.

- the Manufacturing Scatter Factor = 1.10 (based on the test campaign performed by the manufacturer in order to evaluate the replicability of manufacturing process).

Table 1 report the summary of factor used to calculate the max level of load to be tested both for flight and ground conditions.

Table 1. Multiplication factors for calculating maximum loads.

	Design Safety Factor	Environmental Knockdown Factor Cold Load Condition	Environmental Knockdown Factor Hot Load Condition	Manufacturing Scatter Factor
Flight Load	1.50	1.10	-----	1.10
Ground Load	1.50	-----	1.46	1.10

2.3. Finite Element Model Description

The purpose of the finite element analysis is to reproduce the operating conditions considered in the experimental test to be carried out on the wing of the NGCTR-TD. This is to enable the behavior of the structure subjected to the selected loading conditions to be predicted using numerical methodology.

Therefore, the finite element model attempts, by using MSC NASTRAN software, to reconstruct as best as possible the entire experimental environment, which consists of two parts:

- Test rig.
- Wing.

As previously described, the test rig structure consists of a number of components that have a dual function:

- Support the wing by reproducing the connection areas of the wing with the fuselage and nacelles.
- Apply forces to the wing by means of actuators that together allow the two flight and ground loading conditions selected as the most critical to be reproduced.

The mathematical model of the test rig (Figure 4) was made using mainly one-dimensional elements that represent with appropriate section properties the geometry of the different components. The area reproducing the fuselage connection region and the bases of the lateral support columns are modeled with two-dimensional elements. The decision to employ a beam structure for modeling the “Rig” was made to streamline calculation times. Additionally, this simplified approach enables swift adjustments to the loading configuration to accommodate various load cases. Importantly, this choice did not impact the outcome of the results.

The test rig base consists of a truss structure modeled with one-dimensional elements constrained to the ground in the vertical direction through GAP elements (unidirectional constraint) and in the two directions in the horizontal plane at points aligned at its plane of symmetry. To guarantee that GAP elements work as a one-sided constraint, considering the impossibility of defining zero or infinite stiffness, a displacement (Δu)—load curve was defined for them such in (Figure 5):

- For $\Delta u < 0.0$ the stiffness has a very low value (numerically zero).
- For $\Delta u > 0.0$ the stiffness has a very high value (numerically infinite).

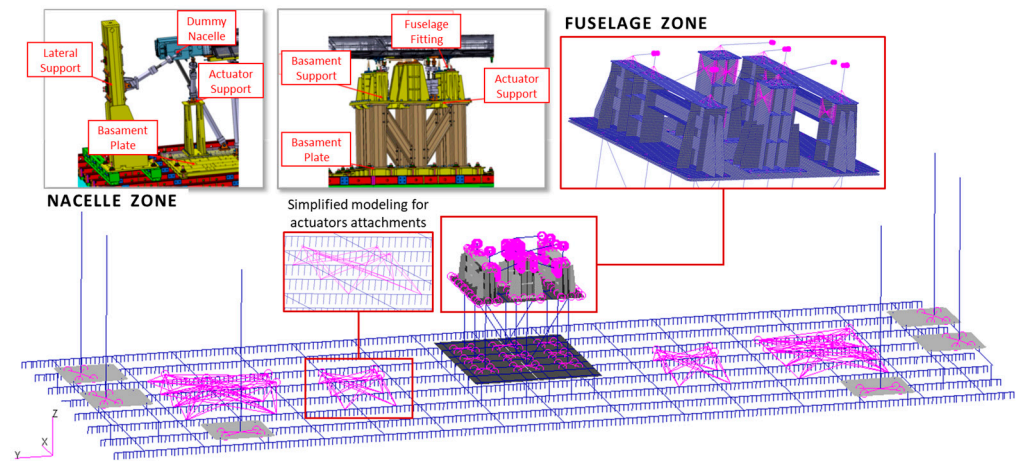


Figure 4. Finite element model for test rig.

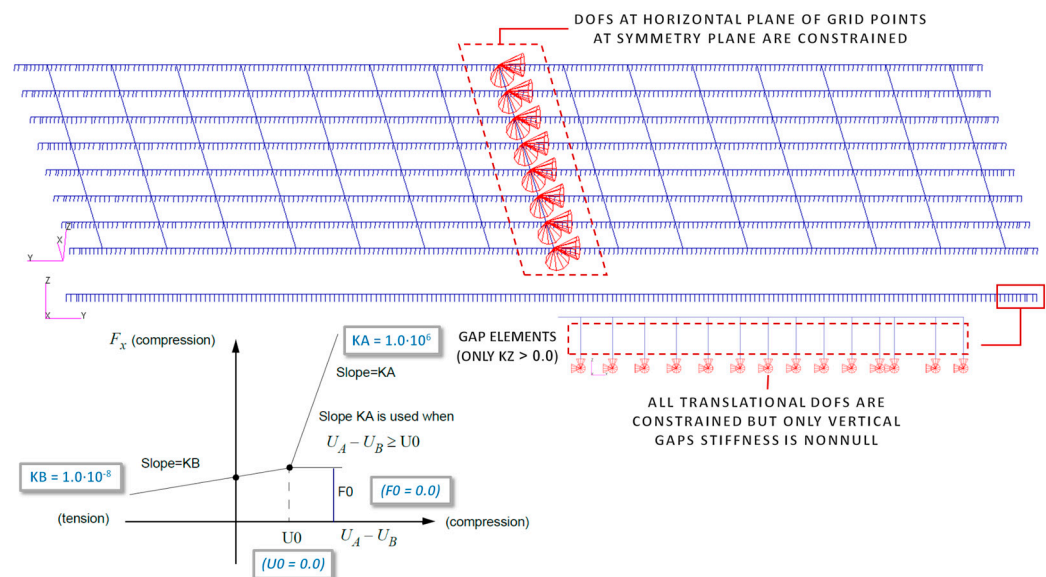


Figure 5. Finite elements modeling for test rig base and constraint conditions.

The actuators were not modeled but were represented by the forces of equal modulus and opposite direction that each of them applies at its points of connection with wing and test rig.

The connection structures of the actuators were defined by means of appropriate rigid elements (RBE2) whose independent nodes are positioned exactly at the spherical joints present in them; the dependent nodes of these rigid elements are the respective connection nodes of the structure of the attachments to the wing and test rig.

The wing model is much more complex in that all its various components have been modeled in detail. In particular, composite components are present like the following:

- Upper and lower skin of wing, trailing edge and flaps.
- Right and left side spars.
- Right and left side ribs.
- as well as metallic components like the following:
- Central and tip ribs.
- Central region of the spars.
- Connection fittings with fuselage.

The composite components were modelled with two-dimensional elements as well as many of the aluminum elements. Aluminum components such as center and side ribs, fittings, and the central part of the spars were modeled with solid elements.

The different components of the wing are connected by fasteners modeled by using rigid (RBE2) and elastic (CBUSH) elements by which properties are properly calculated considering panel thicknesses, fastener diameter and material properties: In detail, this type of modeling requires four nodes [2]:

- A node belonging to the first component to be connected.
- Two overlapping nodes placed equidistant from the two regions to be connected.
- The fourth node belongs to the other component.

The first node is connected to one of the two central nodes via an RBE2 element as the fourth grid point and the other central node. The two overlapping central nodes are connected by a BUSH element with nonzero stiffness in its translational degrees of freedom defined in an appropriate analysis reference system whose X direction is the axial direction of the BUSH element that is coincident with the direction identified by the first and fourth node (Figure 6).

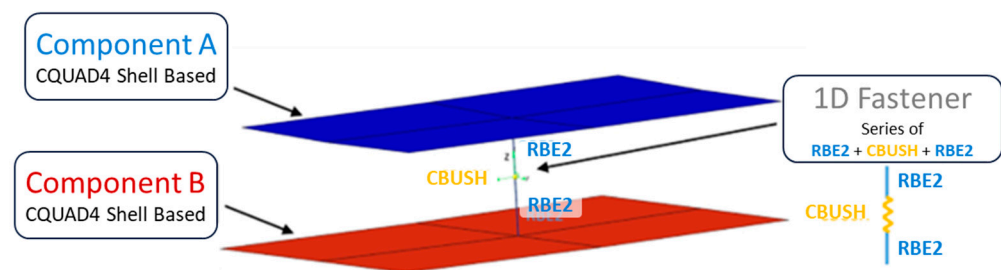


Figure 6. Finite element modeling for fasteners.

For composite components and many of the metal parts, a material with linear behavior was considered; differently, considering them particularly critical, for the following metal components (Figure 7):

- Central and tip ribs.
- Central region of the spars.
- Fitting connecting the wing to the fuselage.

Material with nonlinear behaviors have been considered [3]. The following material have been considered:

- Al7050-T7452 for the central ribs.
- Al7050-T7451 for the other components.

Experimental strain-stress curves are normally provided as engineering data while the most common finite element programs expect such data in terms of “true strain—true stress”. The following relationships (1) and (2) allow you to switch from one data type to another:

$$\varepsilon_{True} = \ln(1 + \varepsilon_{Eng}) \quad (1)$$

$$\sigma_{True} = \sigma_{Eng} \cdot (1 + \sigma_{Eng}) \quad (2)$$

In addition, the strain-stress curves can be given either in terms of absolute strain or plastic strain. The latter format is the one suggested since it allows the linear part to be separated from the nonlinear part; the plastic strain is calculated from the relationship (3):

$$\varepsilon_{ln}^{Plastic} = \ln(1 + \varepsilon_{Eng}) - \frac{\sigma_{True}}{E_{Modulus}} \quad (3)$$

The graphs below (Figure 8) show the above transformations from the data found in the literature [3] for Al7050-T7451 and Al7050-T7452 up to the input provided to the code.

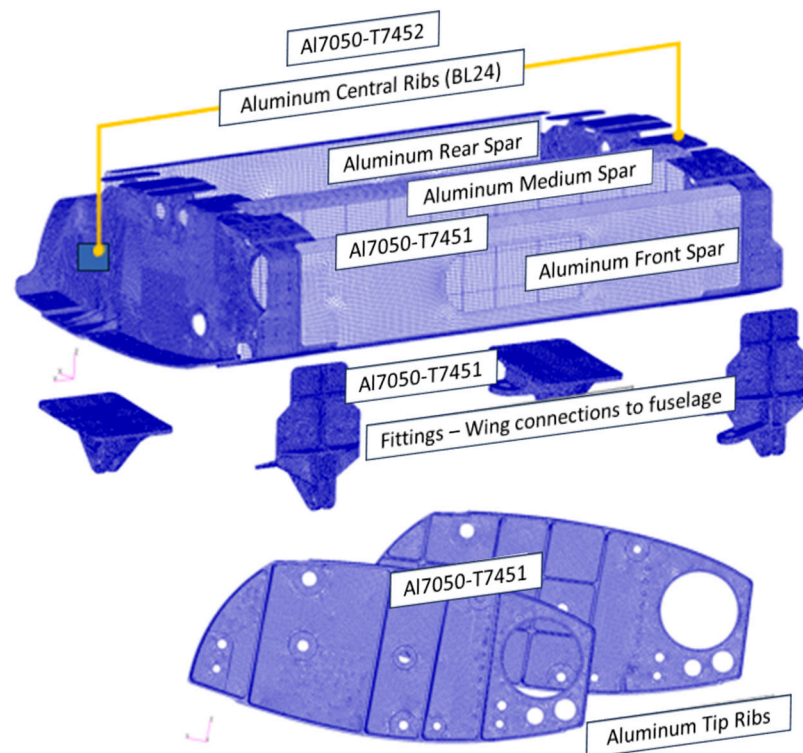


Figure 7. Nonlinear material types used in some critical metallic component.

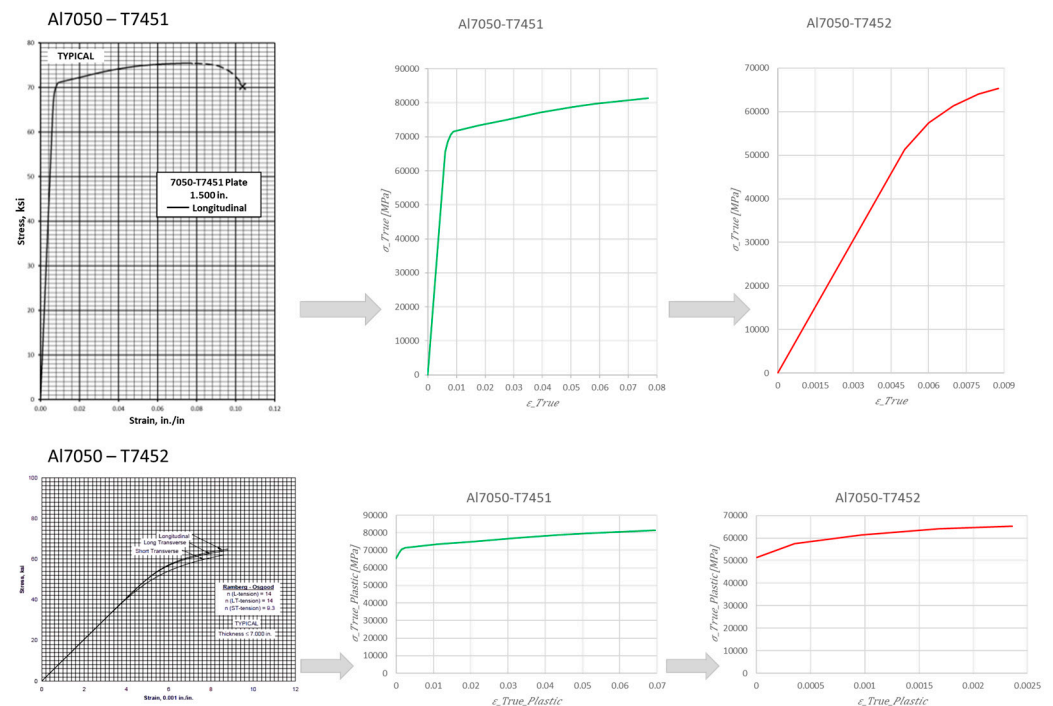


Figure 8. The literature material properties to input material data.

The linear part of the material behavior is defined in the MAT1 entry while the nonlinear one in MATEP entry.

It should be noted that in the various calculations performed on the structure, unrealistic situations in terms of local deformations were observed. This is related to the fact that initially for reasons of convenience and efficiency of the calculation it was avoided to define the different contact conditions present between the various components of the structure.

Depending on the type of deformation, contact (glued or touching) was then introduced in order to solve the various issues that occurred from time to time. Of course, the most realistic solution would have been to apply contact, mainly touching, between all adjacent components. But in a model of about two million nodes, we tried to preserve computational efficiency by limiting such modeling to only the most affected areas.

To complete the description of the model, Table 2 is given with the number of finite element entities in the model.

Table 2. Finite element entities in the model.

MSC Nastran Entry	Total n.	Description
BCBODY1	20	Contact Bodies
BCONECT	16	Contact Pairs
CBAR	19,223	1D beam elements
CGAP	1704	Node to node contacts elements
CHEXA	110,490	Cubic solid elements
CPENTA	5284	Pentahedral solid elements
CQUAD4	1,403,848	2D quadrilateral shell elements
CROD	4	1D axial elements
CTETRA	1,525,417	Tetrahedral solid elements
CTRIA3	743	2D triangular shell elements
GRID	1,988,452	Grid points
RBAR	2	Rigid elements (2 nodes)
RBE2	21,964	Rigid elements (≥ 2 nodes)
RBE3	22	Interpolation Elements

2.4. Analysis Setting

The study that you want to perform on the structure is the one related to the analysis of the structure in post-buckling conditions. This type of analysis assumes that at least large displacements must be activated in the solution process (PARAM, LGDISP, 1).

As mentioned, the actuators were accounted for through the equal and opposite forces they apply at the connection points with the wing and the test rig. At these points there are spherical joints that as the deformation of the wing proceeds result in a change in the direction of each actuator and thus also in the direction of the forces representing it.

In a nonlinear calculation, in case of large displacements and rotations, the dependence of module and direction of a force on displacement leads to define it as “follower force” (Figure 9). Such loads include pressure loads, certain types of concentrated forces or moments, centrifugal forces, and temperature.

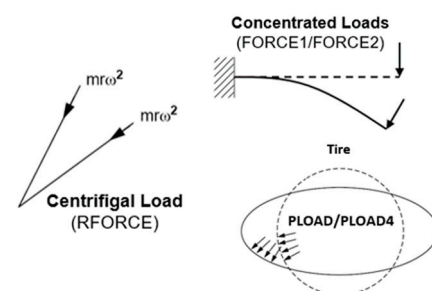


Figure 9. Typical follower forces and their definition in MSC Nastran.

In the case under consideration, the equal and opposite forces to be applied to the connected structures cannot be defined by means of the more common FORCE input card since there is no data in it that relates it to the deformation of the structure; otherwise, the FORCE1 input card allowing the definition of the direction of the load by means of the position of the two end nodes of the actuator, makes it possible through the knowledge of their current displacements to identify the change in the line of action of the force. The

opposite direction of the forces applied to the ends of the actuator is defined by reversing the order of the two nodes in the corresponding input.

By default, follower forces also affect the stiffness matrix in MSC Nastran introducing non-symmetric terms in it (PARAM, FOLLOWK, YES); depending on the value of these terms, the matrix can be symmetrized avoiding any effect on the calculation time.

The complete distribution of the two load conditions selected to be used in the experimental test is represented in Figure 10.

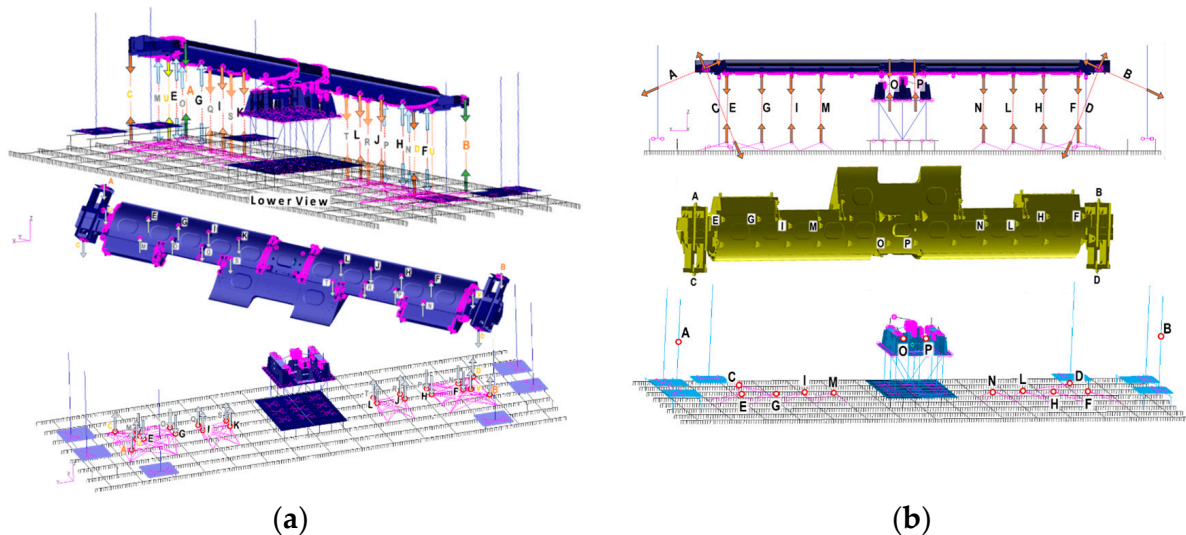


Figure 10. Actuator forces: (a) Ground Load condition, (b) Flight Load conditions.

In nonlinear analysis, the correct definition of the load history is very important because the sequence in which the various loads are applied (separately or simultaneously) determines the result obtained; in fact, the effect of each individual load is influenced by the condition of the structure when it is applied.

As previously indicated, the final load applied in the two load conditions is the result of an increase in the ultimate load (150% of the limit load) related to the inclusion of corrective factors that considers the environment (Environmental knockdown factor) and manufacturing (Manufacturing scatter factor) issues. It leads to the loads listed in Table 3.

Table 3. Load level applied for each of the load condition.

Load Condition	Applied Load
Flight Loads	181.5% of the limit loads
Ground Loads	189.1% of the limit loads.

In this case, to allow the definition of the correct initial position of the wing and actuators, a gravity load has been applied in advance. Obviously, it will continue to act even when the actuator loads are applied.

To set up the non-linear analysis as best as possible, a linear buckling analysis has been executed to verify where these critical conditions occur and if they are within the considered load range. Considering that we are only interested in the critical loads related to the two load conditions due to the actuators, the input structure in buckling analysis must ensure the following:

- The gravity load acts only on the initial stiffness matrix as preload acting it constantly throughout the load history.
- Actuator load is the only input in critical load calculation.

The relationship to be considered for the calculation of critical loads must be as follows:

$$\text{Det} \left[K + K_d^{\text{Gravity}} + \lambda \cdot K_d^{\text{Actuators}} \right] = 0 \quad (4)$$

where K_d represents the differential stiffness due to the specific load.

Considering the standard input for a buckling analysis in MSC Nastran (SOL 105) the Case Control Section structure should be modified in the following way:

- A first linear static SUBCASE must be defined for the gravity load.
- A second linear static SUBCASE must be defined for the specific actuators load configuration.
- The third SUBCASE that activates the buckling analysis must contain the following:
 - $\text{STATSUB(PRELOAD)} = \text{ID}_{\text{SUBCASE Gravity Load}}$
 - $\text{STATSUB(BUCKLING)} = \text{ID}_{\text{SUBCASE Actuators Load}}$

According to the results obtained in buckling analysis and to some convergence issues that occurred running the nonlinear analysis, the history loads relative to the two load conditions have been defined using the STEP sequence summarized in Table 4:

Table 4. STEPs for load histories definition for the Ground and Flight load conditions.

STEP n.	Flight Loads (% Limit Load)	Ground Loads (% Limit Load)
1	Gravity Load	Gravity Load
2	Gravity + 180% Actuator Loads	Gravity + 130% Actuator Loads
3	Gravity + 181.5% Actuator Loads	Gravity + 180% Actuator Loads
4	N.A.	Gravity + 189.1% Actuator Loads

Each STEP considers as initial configuration of the structure the deformed configuration and corresponding stresses/strains level calculated in the previous one.

The above subdivisions are related to the fact that, for example, for ground loads the first buckling condition is occurring a little bit over the 131% of the limit load. For this reason, differently from the other STEPs in which a FIXED option has been used in NLSTEP entry (constant load increments), in the second one an adaptive load increment scheme (ADAPT option in NLSTEP entry) has been considered to overcome the convergence issues encountered.

On the contrary, instability issues for flight loads begin to occur around 170% and have less influence on the convergence of the analysis. This allowed the use of a FIXED type of load scheme in all the STEPs.

Note that the load splitting was performed in such a way as to allow easier reading of the load levels reached; for this reason, in both cases the final step is always the one leading from 180% to the final value.

The strength of the composite components is verified in terms of laminate failure. It means that the output request needed for it is that one of the strains calculate on the equivalent shell elements; the option FIBER in the STRAIN command allows obtaining the output at the upper and the lower surface of each element. The obtained values for strain are compared with the allowable strain determined for the laminate in the directions 0° – 90° and 45° – 135° . This request has been extended only to the elements that are part of the composite regions.

The STRESS output is extended to the metallic components and allows calculating the stress tensor, the equivalent stress; plastic strain is also calculated for components in which material nonlinear behaviors have been considered.

3. Results

In this section the results obtained to the most critical load conditions (Flight Load and Ground Load) will be shown. The two loading conditions were constructed in such a way that they each stress the upper and lower areas of the wing differently. Specifically, with the

Flight condition the wing structure tends to deform in such a way as to compress the upper skin and stretch the lower skin, while in the Ground condition the exact opposite occurs. To obtain an idea of the behavior of the structure subjected to the chosen loads, a linear buckling analysis (sol 105 MSC NASTRAN) was preliminarily executed by considering the limit loads (L.L.) as applied static load. This made it possible to check, first of all, whether the applied loads brought the structure into unstable conditions. Furthermore, the results of these linear analyses allowed the subsequent post buckling analysis to be set up in the best possible way. Considering the multiple factors reported in Table 1, the maximum value of load used during the simulation is 181.5 to F.L. and 189.1 to G.L. During the application of Flight Load, the ends of the wing move upward introducing compressive loads mainly in the panels located in its upper part which will be, therefore, the one potentially most affected by buckling phenomena Figure 11.

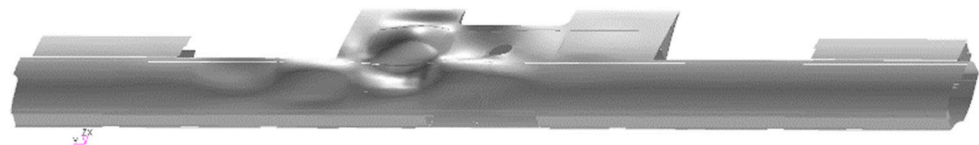


Figure 11. Flight Load Condition: Mode shapes for eigenvalue 1.695497.

Instead, the behavior of the structure under the action of ground load shows several instability situations concentrated in the panels of the lower part of the wing. This is closely related to the characteristic of the load tends to compress that region of the wing. This phenomenon can be seen clearly for the same eigenvalues in Figure 12.

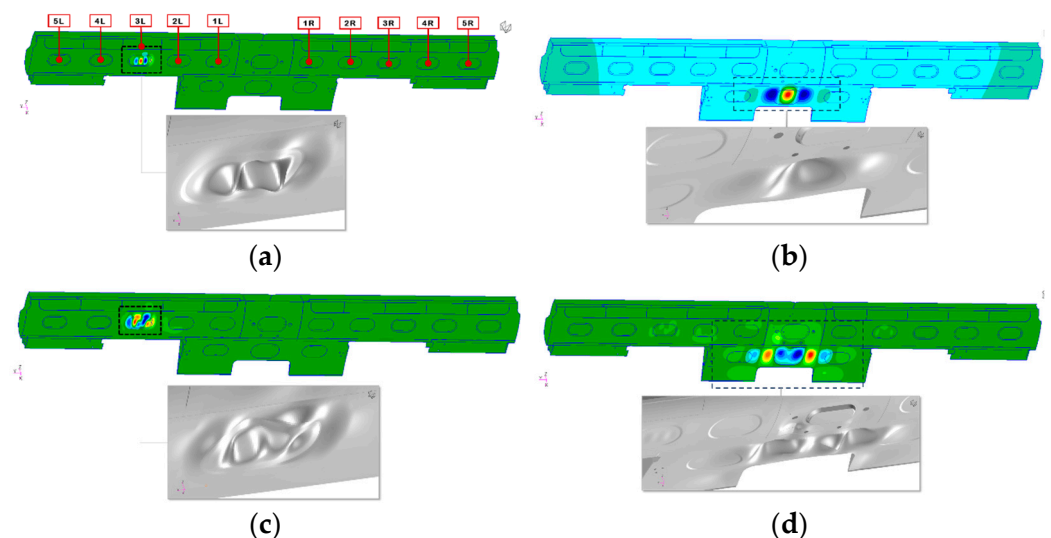


Figure 12. Ground Load Condition Deformed shapes on Lower skin panels for same eigenvalues: (a) 1.310801, (b) 1.573565, (c) 1.611699, (d) 1.894201.

Verified by the presence of such critical eigenvalues, the nonlinear post buckling calculation allowed us to verify whether the buckling was critical for the entire structure or whether it was simply local buckling.

In order to obtain an idea of the post-buckling response to critical loads, it was considered useful to start from the qualitative analysis of the global deformations of the structure, reporting them in such a way as to make it possible to appreciate their evolution as the load changes and to verify the possible onset of the buckling phenomenon. In the following figure some steps of non-linear analysis are reported in order to highlight a load range for the buckling onset. Figure 13 for Flight Load and Figure 14 for Ground Load shows some frames of the nonlinear analysis to highlight a load range for the onset of buckling. In F.L. conditions up to 150% LL, no buckling is observed. Above the value, local

buckling starts and involves the upper skin in the trailing edge. While for the G.L. case, the local buckling concerns the lower skin and the access panels placed on it.

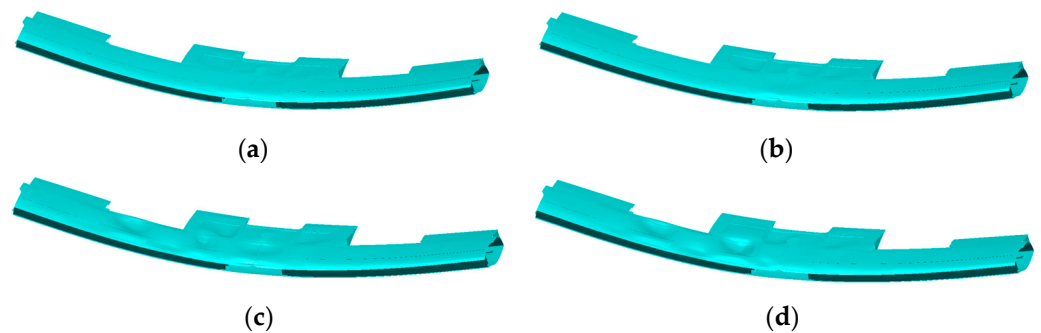


Figure 13. Global Deformation under Flight Load condition: (a) 100% F.L., (b) 150% F.L., (c) 180% F.L., (d) 181.5% F.L.

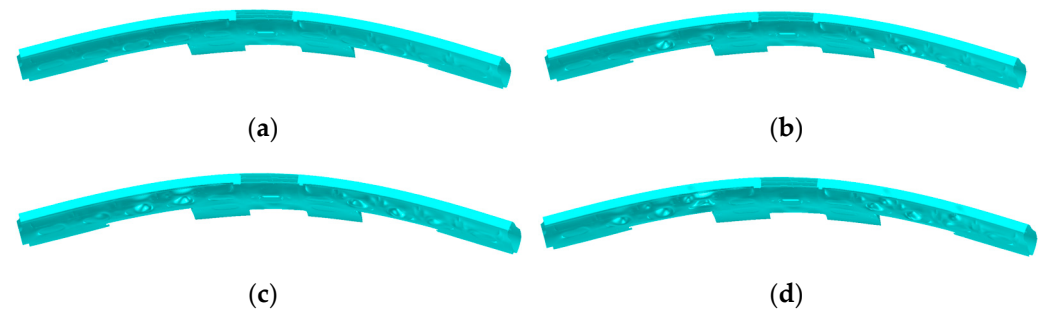


Figure 14. Global Deformation under Ground Load condition: (a) 100% G.L., (b) 150% G.L., (c) 180% G.L., (d) 189.1% G.L.

Through the analysis of the deformed shape, the region of local buckling can be seen. To better understand the behavior structure in the non-linear regime, the deformation for critical area is reported for each critical load condition (Figures 15 and 16). The results show that the local deformation occurs gradually and very slowly and therefore cannot be related to global buckling phenomena.

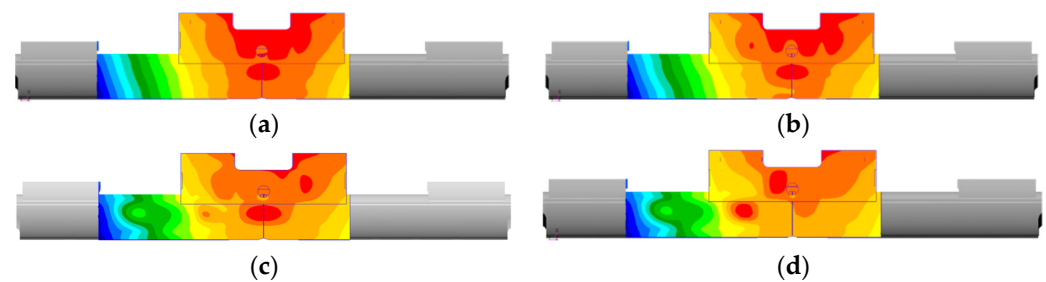


Figure 15. Local deformation on the upper central skin panels (a) 100% F.L., (b) 150% F.L., (c) 180% F.L., (d) 181.5% F.L.

To verify how these local effects affected the overall behavior of the structure, the following are graphs in which the trend of the maximum displacement (measured at the end of the wing) is plotted as a function of percentage changes in the applied load (Figures 17 and 18).

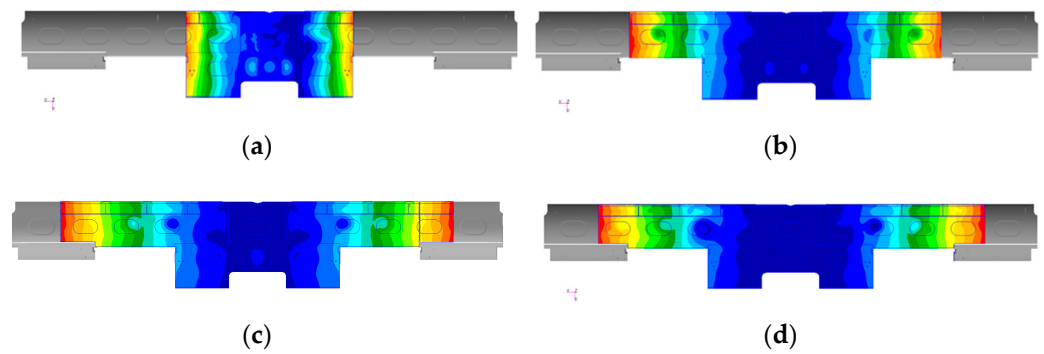


Figure 16. Local deformation on the lower skin panels (a) 100% F.L., (b) 150% G.L., (c) 180% G.L., (d) 189.1% G.L.

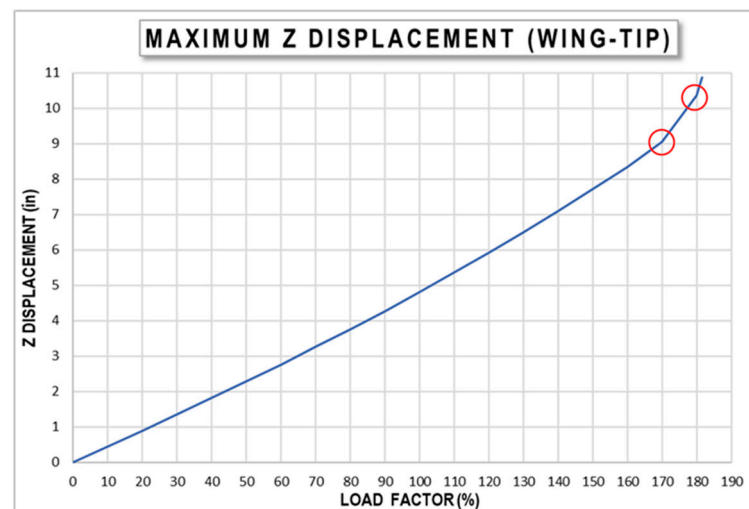


Figure 17. Maximum z displacement (Wing TIP) Flight Load condition.

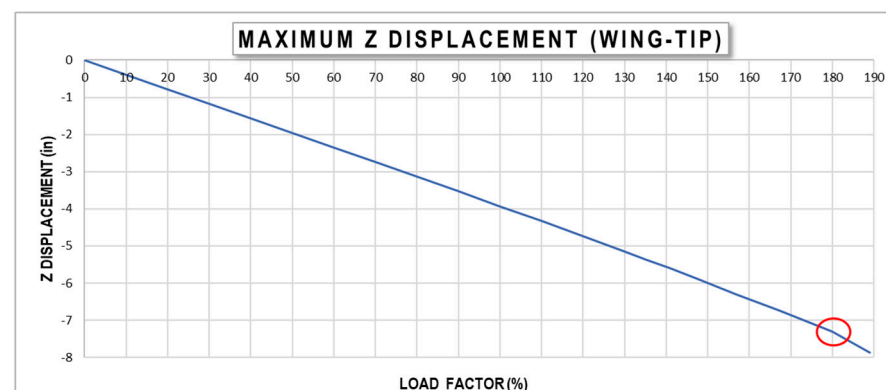


Figure 18. Maximum z displacement (Wing TIP) Ground Load condition.

As can be seen from the graphs there is an area for both loading conditions examined where a sharp change in the slope of the curve occurs, which can be justified by the fact that near the corresponding load value the structure globally begins to be affected by local buckling instabilities. The points of change in the curve slopes were highlighted in the plots:

- F.L.: 170% and 180% of the limit load.
- G.L.: 180% of the limit load.

By running the study of non-linear analysis results, it was possible to verify the stress distribution and plastic strain on metallic parts, while for composite parts it was considered

the strain distribution only. Considering tip rib and fittings, a different response can be observed according to the load condition. Figures 19 and 20 show how for the F.L. condition the maximum value of stress is higher than the maximum stress that structure can support and that is when the plasticization is present. On the other hand, under G.L. conditions, the stress level at the last applied load step always remains less than the maximum tolerable and thus no plasticization is engaged.

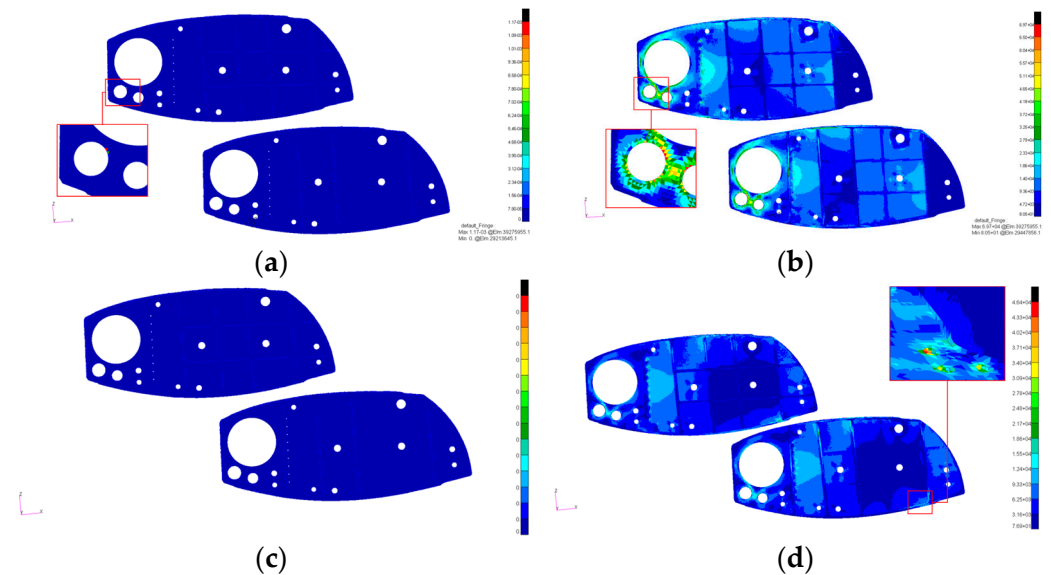


Figure 19. TIP RIB Results: (a) Plastic-Strains 181.5% F.L., (b) Stresses 181% F.L., (c) Plastic-Strains 189.1% G.L., (d) Stresses 189.1% G.L.

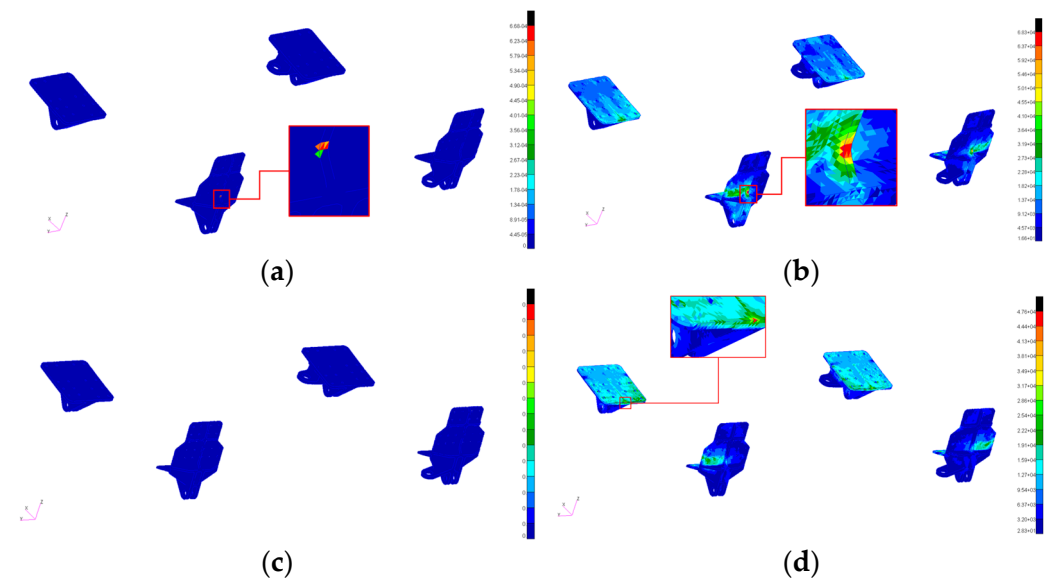


Figure 20. FITTINGS Results: (a) Plastic-Strains 181.5% F.L., (b) Stresses 181% F.L., (c) Plastic-Strains 189.1% G.L., (d) Stresses 189.1% G.L.

Evaluating data on BL 24 and spar, it was possible to see that the behavior is the same under both loading conditions. In particular, on the BL 24 at the last step of applied load, a high stress value was observed with beginning of plasticization (Figure 21), while for Spar the stresses do not exceed the limit value (Figure 22). It is important to specify that the pick of stress above the mark is located in a small area on BL24, Tip Ribs, and fittings as well as for plasticization. There are only a few elements with a high value and therefore they can be neglected.

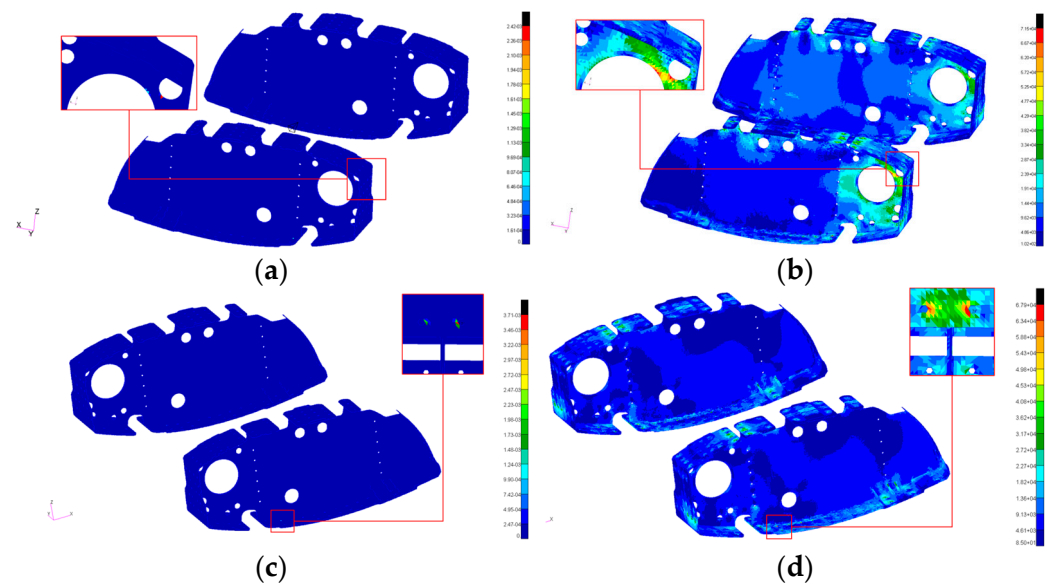


Figure 21. BL24 Results: (a) Plastic-Strains 181.5% F.L., (b) Stresses 181% F.L., (c) Plastic-Strains 189.1% G.L., (d) Stresses 189.1% G.L.

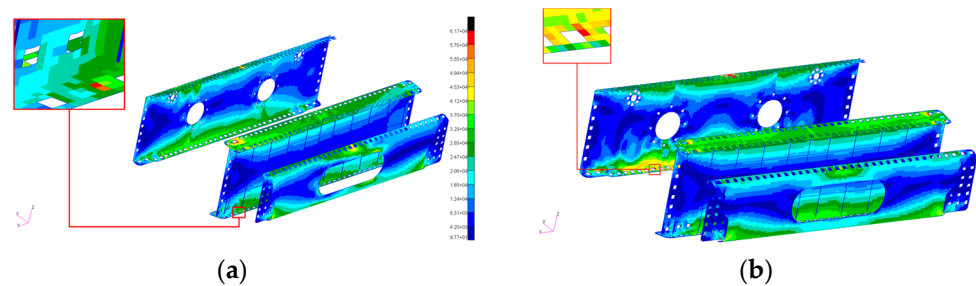


Figure 22. SPAR Stress distribution: (a) 181.5% F.L., (b) 189.1% G.L.

Analyzing the composite parts response, it was possible to verify that the structure behavior is more critical for the compression zone during the application load for each load condition. Figure 23 shows the strain distribution on upper skin for F.L. and lower skin for G.L. In both cases the maximum values are not excessively high, and above all it is concentrated on only a few elements of the mesh and therefore may not be considered significant; all that can be related to contact issue (bonding and bolted method).

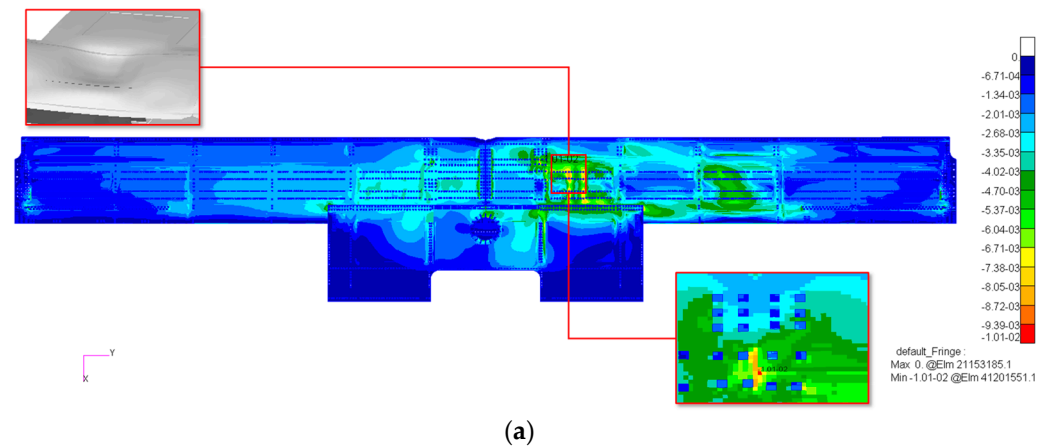


Figure 23. Cont.

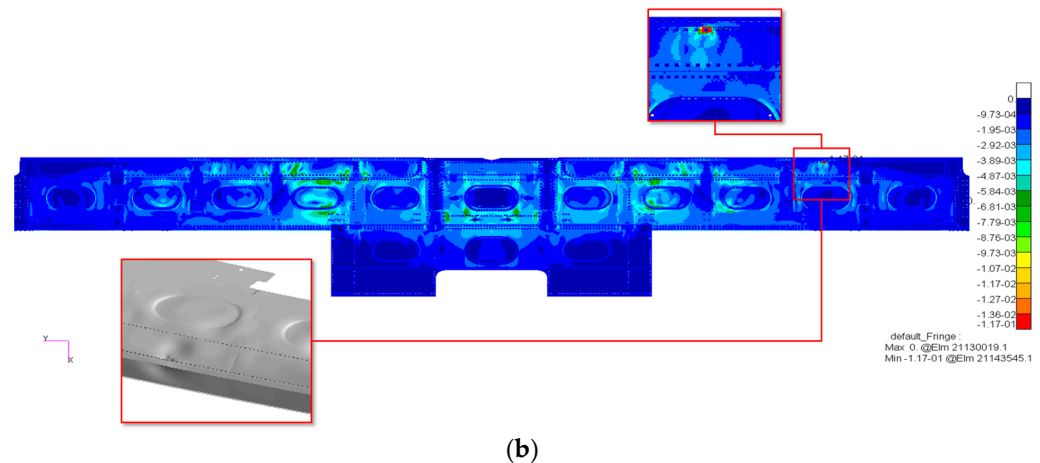


Figure 23. Strain distribution: (a) UPPER SKIN 181.5% F.L., (b) LOWER SKIN 189.1% G.L.

4. Conclusions

This study focuses on the critical aspect of post-buckling behavior in composite structures, particularly emphasizing its significance in aeronautical structural design. The conventional engineering approach tends to result in over-sizing composite components to ensure buckling resistance, leading to heavier structures. Foundational works by Budiansky [4] and Dickson et al. [5] contribute theoretical principles and practical design methodologies, respectively, to enhance the reliability and efficiency of aerospace structures.

The Building Block Approach outlined in EASA AMC 20-29 [13] emphasizes phased testing for large aircraft structures, but its time-consuming and costly nature necessitates a shift towards high-fidelity post-buckling analyses. Such analyses, exemplified by MSC NASTRAN SOL 400, enable detailed insights into structural behavior, facilitating targeted and cost-effective design improvements. A completely innovative contribution to the study of post-buckling behavior by nonlinear regime analysis was provided by the reproduction in a numerical model environment of the complete rig on which the wing will be placed during the experimental tests. Furthermore, in order to reproduce as realistically as possible the behavior of the moving actuators during the tests, Following Force was used. This allowed us to vary, during the simulation, the direction of load application on the wing thus taking into account its deformation. This approach reduces risks during experimental testing and provides a clear and detailed view of what the response of the wing to loads will be during the experimental phase. These simulations support an iterative design process, allowing engineers to efficiently explore alternatives and refine designs based on simulated post-buckling behavior.

In the context of the Next Generation Civil Tilt Rotor Technology Demonstrator's wing, a high-fidelity post-buckling analysis using MSC NASTRAN SOL 400 was instrumental in de-risking the static test. By considering geometric and material nonlinearities, the analysis delved into post-buckling regimes well beyond ultimate loads, offering detailed stress distribution insights. This approach eliminated the risk of structural failure due to buckling and allowed for a more efficient permit to flight process. Pushing the structure beyond ultimate loads in the post-buckling regime decreases risks during static tests, crucial for certification.

The results show how it has been possible to study in detail the local buckling phenomena arising on the structure and how it affects the overall response of the wing by using a non-linear analysis. It was possible to know precisely the stress distribution in the post-buckling regime and to know the areas where the peaks are generated. All this made it possible to rule out structural failure due to buckling, which would not have been possible if we had stopped at a linear analysis.

Future developments will involve comparing numerical analyses with experimental data, further validating the reliability of high-fidelity simulations in enhancing the safety, efficiency, and cost-effectiveness of airframe structures.

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