

AUTOMATED SIZING PROCESS OF A COMPLETE AIRCRAFT STRUCTURE FOR THE USAGE WITHIN A MDO PROCESS

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Abstract

The following paper gives an overview of an automated sizing process of a complete aircraft structure which is part of a multidisciplinary optimization process that has been developed at the German Aerospace Center (DLR). It covers all aspects of aircraft model generation, model analysis and the implemented sizing concepts. Where the main focus concentrates on metallic based sizing within the global optimization process, an alternative composite sizing concept is also described in detail. Furthermore, exemplary results of the automated sizing process will be illustrated.

1. INTRODUCTION

In the early phase of an aircraft design process it is necessary to evaluate the performance of many aircraft configurations and structural concepts. Thereby, fast and efficient simulation tools and analysis processes are required to reduce the effort for the model generation, simulation and optimization. Within the DLR project Digital-X a multidisciplinary design optimization (MDO) process has been developed including disciplines, such as structure mechanics, aeroelastics and aerodynamics. To ensure the consistency in the parametric description of the aircraft model, the Common Parametric Aircraft Configuration Schema (CPACS) is used, which was developed by the DLR [1]. All disciplinary tools can build up their models from this parametrization. The developed MDO chain is illustrated in Fig. 1.

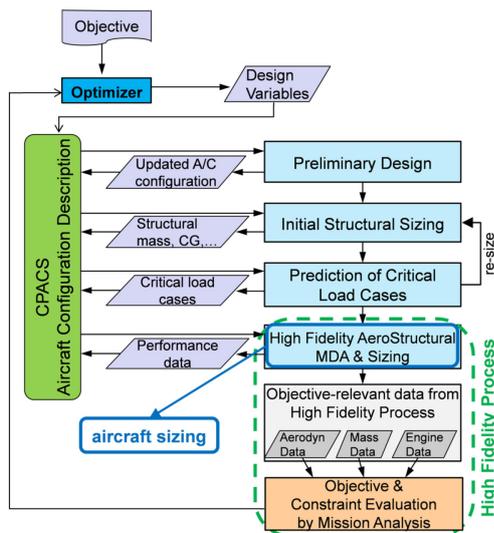


Fig. 1 Developed global MDO design approach

After the computation of the preliminary design, the critical load cases are calculated. The loads calculation process

also includes a preliminary structural sizing. Afterwards, the high fidelity process is performed including a detailed analysis and sizing of the complete aircraft structure, which is shown in Fig. 2 and presented comprehensively in this paper.

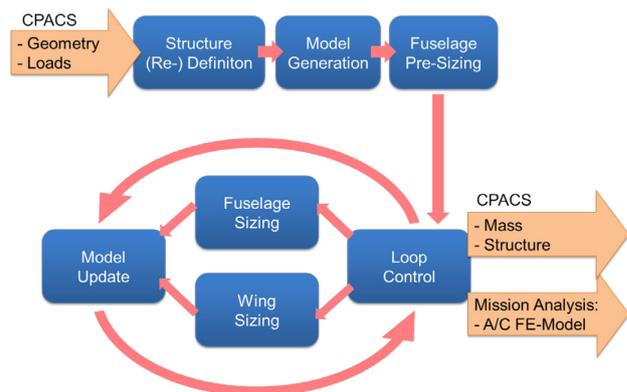


Fig. 2 Generic scheme of implemented sizing process

Compared to the loads process, the fuselage is modelled explicitly in the high fidelity process, which allows for a detailed consideration of the fuselage/wing interface. Therefore, it is possible to analyze more realistic designs and take into account the interaction of local effects in the global structural response. Prior to the model generation within the high fidelity process additional tools adjust the CPACS data set, i.e. the number and positions of the ribs within the wing box and the fuselage frame positions. This is necessary to ensure the compatibility between the wing and the fuselage in case the global optimizer changes the overall aircraft geometry. Afterwards, a detailed finite element model of the aircraft structure including the fuselage, the wing and the empennage is generated, analyzed and sized automatically using the fully parametric tools DELiS/SBOT and TRAFUMO/S-BOT+. The loads used for the structural sizing are obtained from the upstream loads process and transferred via CPACS. For the fuselage sizing a combined analytical and

numerical approach is used and focus is put on the run time performance. The results of the high fidelity process are used for the subsequent mission performance analysis of the aircraft.

The following paragraphs of this paper represent the sequential process flow and provide a detailed description of each sub process including its methods and tools.

2. AIRCRAFT STRUCTURE DEFINITION

As shown in Figure 1, the global optimizer of the MDO process accesses a set of global design variables. All following sub processes are executed sequentially and have to handle variations of the global design variables in a consistent way. In the presented MDO process the wing planform is subject to optimization and wing position and stabilizers are adapted to meet flight quality requirements in the preliminary design step, whereas the fuselage geometry is not changed. This may cause inconsistencies in the aircraft structure definition. To ensure consolidated interfaces of fuselage and wing structure the CPACS data set has to be updated.

The tool F-DESIGN (Fuselage DESIGN) [2] automatically updates the CPACS fuselage structure definition according to the wing and empennage geometry.

As described in [2], the fully parameterized structural interface of wing and fuselage bases on the geometrical position of the wing spars intersecting the fuselage. A suitable structural connection of fuselage and wing requires the so-called fuselage main frames to be located at the exact spar intersection positions as illustrated in Fig. 3.

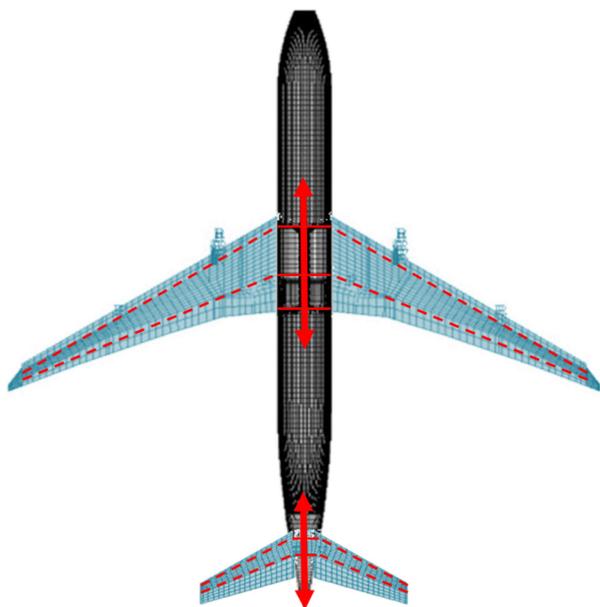


Fig. 3 Possible change of wing spar positions due to variation of design variables in global optimization

During the optimization process the wing spar positions may change either by a modified global wing position or by a modified wing root chord (see Fig. 3). Therefore, F-DESIGN evaluates the CPACS definition of the wing spars and adapts the complete definition of the structural

interface of wing and fuselage to reestablish consistency. The fully parameterized structural interface of empennage and fuselage [2] is analyzed and updated in the same way.

Due to the optimization of the wing planform and wing span and the fixed position of secondary structures like landing gear and engine in the developed MDO process (no explicit design variables), the initial rib structure of the wing defined in CPACS may violate common structural wing design rules (e.g. a rib should be aligned with the spanwise engine position to introduce high thrust loads safely into the structure) and must be updated before generating the model for the sizing process. Therefore, a tool was created as functionality within the wing model generator described in paragraph 3.2. which changes the concerned rib structure positions and writes them back to the CPACS data set. The rib structure generation is based generally on some simple rules which are:

- 1) Ribs are separated into **system ribs** which serves for correct load introduction or connection of secondary structures (e.g. landing gears, flaps, engine) and **slave ribs** which are filled to reduce the buckling span of wing skins
- 2) System ribs are fixed and should not be moved or changed in any way
- 3) Filling the slave ribs is made by an initial rib spacing which should be set to a well-validated value
- 4) Slave ribs can be deleted due to violation of geometrical constraints (e.g. at the transition between flow aligned ribs and ribs aligned normally to the wing front spar at the outer wing which can cause geometrical rib intersection)
- 5) If not defined explicitly, all orientations of slave ribs are assigned by the orientation of its corresponding system rib (in inboard direction)

In Fig. 4 an adapted inner wing structure is illustrated based on changing outer wing shape parameters and a consequent update of the internal rib structure.

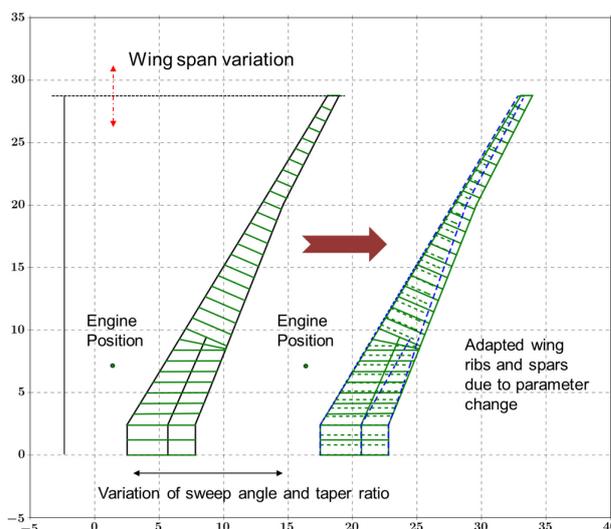


Fig. 4 Visualization of internal wing structure change due to outer shape parameter change and adaption of the ribs

3. MODEL GENERATION

3.1. Fuselage model generator

The fuselage model is created by the TRAFUMO (**T**Ransport **A**ircraft **F**uselage **M**odel) model generator [3]. Based on the CPACS definition, the tool TRAFUMO generates a complete FE model of the fuselage including coupled mass elements for load application in a fully automated process.

First, TRAFUMO reads in the CPACS file using a PYTHON based wrapper module, computes a geometry model of the fuselage and finally writes an ANSYS input file in the APDL format. This file is used in the ANSYS Pre-Processor PREP7 for the FE model generation together with an in house developed toolbox called PROSHAPE, that extends the capabilities of the standard APDL commands.

For the computation of the so-called geometry lofts of the fuselage hull and the floors in 3D-space, TIGL library functions [4] are used extensively. Subsequently, frames, stringers and floor structures are positioned on the fuselage geometry via cutting planes and their extrusion paths on the lofts are calculated as shown in Fig. 5

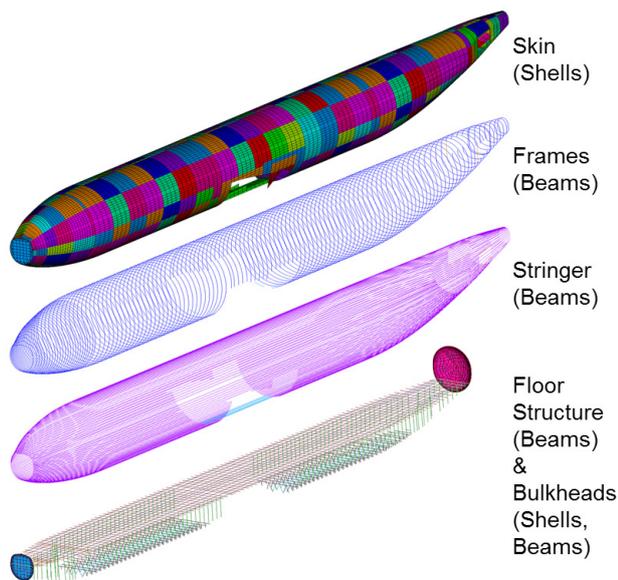


Fig. 5 Structural components of the basic fuselage model generated using TRAFUMO.

According to their CPACS definition, the cross section properties of all structural elements are calculated and linked to lines and areas for later meshing. For the static sizing purpose as applied in the presented MDO process TRAFUMO generates meshes in so-called GFEM quality, where each bay on the fuselage hull between two adjacent frames and stringers forms one 4-node shell element. Likewise, frames, stringers and floor structures form one beam element between two geometrical crossing points. Bulkheads, center fuselage area and tailplane attachment area are created in a subsequent step as they require the basic FE model's geometry. The latter are described in more detail in paragraph 3.3.

At that point, the generated fuselage model is suited for the subsequent sizing process as described in paragraph 4

3.2. Wing model generator

To obtain a structural model of the wing, the abstract wing description in the CPACS dataset must be interpreted and translated into a finite element conform syntax. Therefore, the tool DELiS (**D**esign **E**nvironment of **L**ightweight **S**tructures) is being developed and enhanced at the DLR Institute of Composite Structures and Adaptive Systems. Based on the Python programming language, DELiS creates an abstract, object oriented model of the wing structure. It contains all structurally relevant CPACS information and enriches it with required data for finite elements. A resulting finite-element model of a wing with its main structures can be seen in Fig. 6

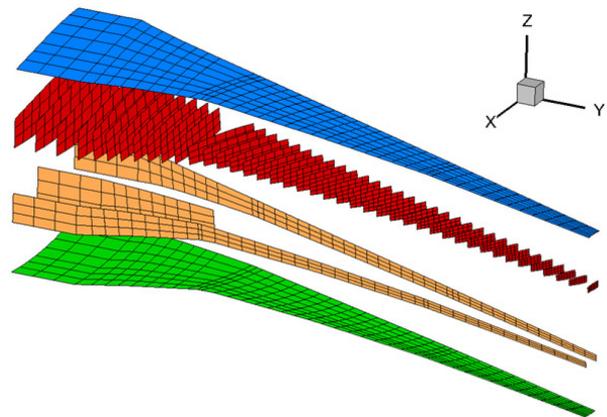


Fig. 6 Generated FE model of the wing with inner parts (skins, ribs and spars)

In the first step DELiS reads the hierarchical CPACS structure and creates initial top level CPACS objects, components and structure elements. These objects are extended with metadata describing associations between structure elements. In the second step the wing is partitioned by a grid of spars and ribs on the wings plan form. This grid is refined by so called imaginary ribs and spars to achieve the desired mesh density. The skins of a wing are mapped on one or several bays of adjacent spars and ribs. Compared to the stiffener elements in the fuselage, stringer elements on the upper and lower wing skin are also represented internally in DELiS but are not explicitly modeled in a finite element model for a subsequent sizing and analysis. However, they are included in the wing stiffness properties in the sizing process through an analytical approach which transforms the stiffener as an additional smeared skin layer in the upper and lower wing skin with the original global stiffness properties of the stringer elements like if they are used as beam elements.

Lastly, points on the outer hull, the so called jig-shape, for the wing are calculated. For this task the TIGL library [4] provides appropriate methods to obtain the geometry in 3D space. These keypoints are aggregated to lines and areas for each structure element. Besides geometry, the finite element properties for profiles and sheets are generated and mapped to the respective set of lines and areas. This collection of geometry and FE-properties

yields a compact interface to common structure solvers. In this paper, the well-known FE solver ANSYS was used within the static sizing process.

3.3. Coupling of fuselage and wings

After creating the fuselage, wing and empennage structural models separately, the models need to be coupled. During the fuselage modelling process described in paragraph 3.1, the complex load introduction regions that transfer the loads from the wing and the empennage into the fuselage are generated according to the CPACS geometry definition of the wing and the empennage [3], see Fig. 8 and Fig. 9.

Additional ANSYS APDL routines were implemented for the automated coupling of the structural models. The coupling routines are designed to account for different model fidelities. Hence, coupling by using common nodes, as would be appropriate for simple beam models, is replaced by consolidated interfaces that allow coupling of models with different element types and mesh sizes.

The actual structural coupling at the interfaces is established by constrained pairs of nodes. For each pair of constrained nodes, six equations for the spatial degrees of freedom are generated which define the rigid body motions. User defined groups of nodes for all structural parts of the coupling interface ease the process.

The following figures Fig. 7 to Fig. 9 show the coupling interfaces at the center fuselage and tailplane attachment area and the resulting structural model of the complete aircraft.

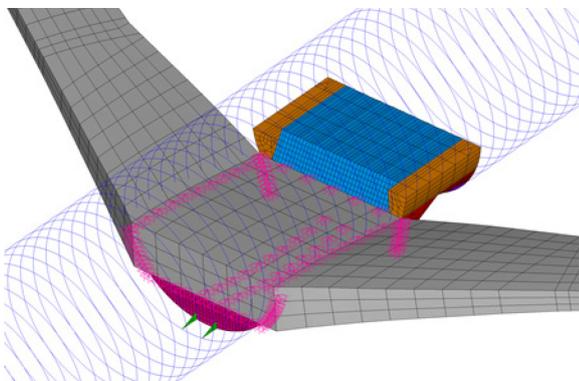


Fig. 7 Coupling Interface at the center wing section

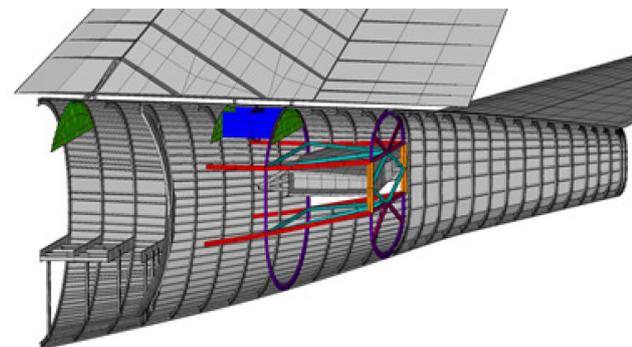


Fig. 8 Visualization of the structural components of the rear fuselage section to couple with the empennage

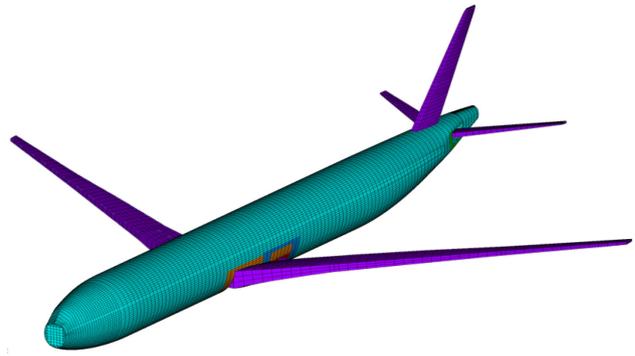


Fig. 9 Full coupled aircraft model

4. STATIC SIZING PROCESS WITH METALLIC DESIGN

4.1. Transferring loads

Besides generating the aircraft model, loads need to be applied to the structure for a correct initialization of the implemented sizing process. As shown in Fig. 1, an integrated loads process was carried out beforehand to estimate the critical loads for the given aircraft configuration. The output of the loads process [5] are the different load distributions for the defined load conditions (e.g. cruise, gust, maneuver) in terms of shear, moment and torsion (SMT). In this approach commonly used in research and industrial aircraft design, all acting loads are applied to nodes located at a so called load reference axis. In the following the nodes are called dynamic aircraft model (DAM) points. The latter are connected to the aircraft structure via non-stiffening elements to ensure transmission of the applied loads. For the fuselage, a DAM point is attached to all nodes of the closest frames. For the wing and empennage, the nearest rib is used for attaching the DAM point. Fig. 10 and Fig. 11 show the implemented approach for the fuselage and wing. Based on the given loads, the structural properties will be adapted by the sizing process described in the following paragraphs.

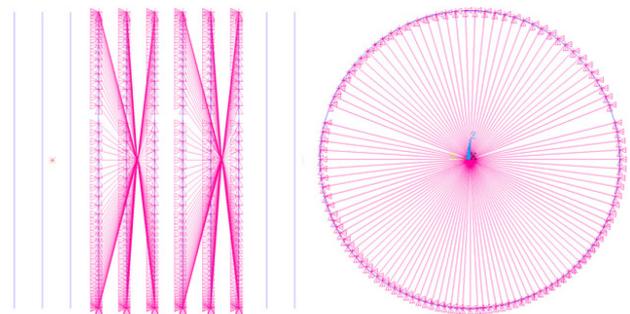


Fig. 10 Exemplary DAM points within the fuselage connected to the nearest frames via rigid body elements in side (left) and front view (right)

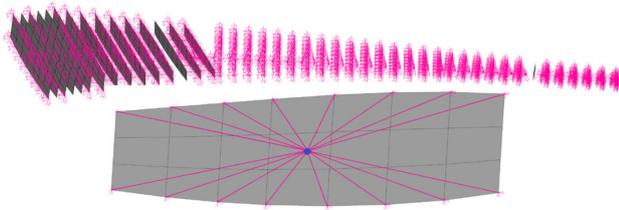


Fig. 11 DAM points within the wing model connected via rigid body element into the wing rib structure

4.2. Analytical pre-sizing of the fuselage

In a large MDO process the run time of the sub processes has to be kept as short as possible to ensure an acceptable global run time for the optimization task.

The tool STEFFS (Skin Thickness Estimation For Fuselage Sizing) has been developed at the Institute of Structures and Design to reduce overall process run time of the static sizing process of fuselage structures. The Python based tool uses solely analytical methods and provides a very fast sizing of the fuselage skin based on a CPACS dataset. However, the analytical sizing method has to use idealizations and therefore cannot account for complex geometries and structural details, e.g. the complex center fuselage area. Hence, using a numerical sizing process is still essential to comply with the level of detail expected by the static sizing process presented in this paper.

When used as a pre-sizing method STEFFS improves the start values for the numerical sizing process. These results in a lower number of iterations necessary to reach convergence, thus reducing the overall sizing process run time. Moreover, STEFFS determines the most critical fuselage load cases. This information can be used to reduce the number of load cases for the numerical sizing process which is a second means to reduce the overall process run time of the fuselage sizing process.

The analytical pre-sizing method of STEFFS uses the same sizing criteria as the numerical sizing tool S-BOT+ [3]. Based on fully stressed design principles for isotropic materials, the fuselage skin is sized to meet ultimate strength and stability criteria (local skin buckling). STEFFS considers the CPACS definition of the fuselage geometry and stiffening structure. The results are validated against a second analytical tool developed at the Institute of Aircraft Design (IFB) at the University of Stuttgart as described in [3] and show very good agreement for standard fuselage sections.

Fig 12. shows an exemplary fuselage skin thickness sizing result. The fuselage is pictured unrolled with the top at $\phi=0^\circ/360^\circ$ and the bottom at $\phi = 180^\circ$.

The analytical sizing results and information on critical fuselage load cases are updated to CPACS automatically and are provided for the numerical sizing process described in paragraph 4.3.

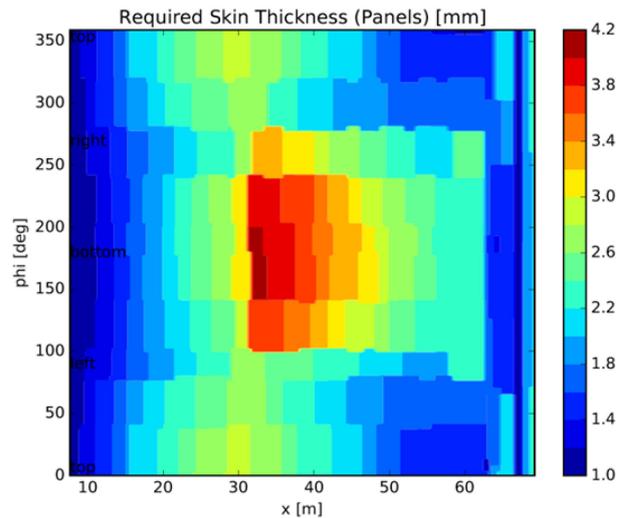


Fig. 12 STEFFS exemplary fuselage skin thickness sizing result for a generic two aisle aircraft

4.3. Numerical sizing of wings and fuselage

After the analytical pre-sizing of the fuselage is completed, the fuselage and the wings (main wing and empennage) are sized separately in the numerical sizing process as shown in Fig. 2. A state-of-the-art fully stressed design approach (FSD) is implemented as iterative method in the used numerical sizing tools called S-BOT for the wings and S-BOT+ for the fuselage model [3]. This approach relies on iteratively computing the reserve factors for each element of the aircraft model based on the load distribution of all load cases applied. Reserve factors which represent the remaining resistance of the structure against maximum allowable loads are calculated in terms of strength and stability. The von-Mises stress is evaluated for strength considerations, whereas handbook methods (local skin buckling) are used for stability evaluation.

Based on the reserve factors new properties are computed for each element involved in the sizing process and updated subsequently in the finite-element models. A final FE calculation of the entire aircraft model is performed for both fuselage and wing sizing to take into account the stiffness changes and their effect on the loads interaction between fuselage, wing and empennage. This approach is executed iteratively until the total structural mass change equals the defined convergence criteria.

5. ENVIRONMENT FOR SIZING AND OPTIMIZATION OF COMPOSITE AIRCRAFT STRUCTURES

In addition to the sizing environment for metallic structures, an additional environment for the design and structural optimization of composite aircraft components called VErSO (shown in Fig. 13) has been developed.

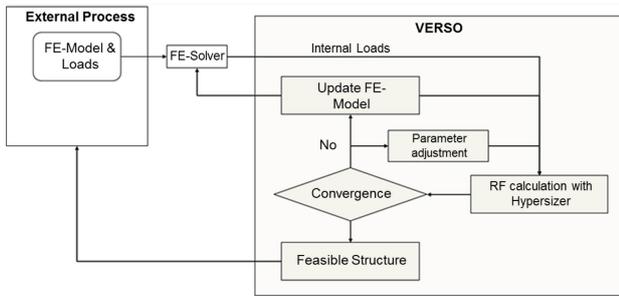


Fig. 13 Composite Optimization Process & Environment

During the optimization, three types of parameters are optimized:

- Structural Design Concept of each component, i.e. the decision if a panel is stiffened with a T or an I stiffener
- Structural concept parameters, i.e. stringer pitch, stringer height, thicknesses, etc.
- Material parameters, i.e. ply share

With this approach, a wide range of parameters can be considered during the optimization calculate the best design.

The Reserve Factors (RF) necessary for the optimization are calculated by using the commercial software HyperSizer. In HyperSizer, the Finite Element model of the wing or aircraft is imported, and each panel is assigned to a software internal component. With the imported loads and current structural parameters RF are calculated for a wide range of sizing criteria. Here, local skin buckling and strength is evaluated as well as the local buckling of stringer objects like web or flanges. Furthermore, global buckling and column buckling, Damage Tolerance and mixed stability-strength criteria are evaluated. By treating a component as unit of skin and stringer, a more optimized panel compared to a classical sizing is calculated. This is due to load redistribution between skin and stringer.

The second core component of VERSO is the parameter adjustment module. If HyperSizer is used stand-alone, each component is optimized for itself. Adjacent components are not taken into account. To ensure compatibility of structural components, structural parameters are modified based on their current values. The values are used as support points for polynomial curves, which are laid thru the components. The values of each component are then modified, based on the polynomial smoothing curves. With this approach it is possible, that configurations with negative RF appear. These negative RF disappear with increasing number of iterations. The adjustment curves are not only used for structural parameters, but also for material parameters. In order to ensure material compatibility between two components, the allowed ply distribution in skin and stringer is calculated for each component based on the adjacent ones.

For an efficient optimization, a smeared approach for the composite materials is used. The composite layups are homogenized, and only thickness and ply share are optimized. After the optimization, the smeared laminates are transformed to discrete layups. Hereby, common composite design rules are considered. Examples are common outer plies or a maximum number of adjacent plies of the same orientation.

In order to demonstrate the capabilities of the environment, a structural optimization of the wing the complete aircraft has been performed. The optimization run successfully and all results were plausible. In Fig. 14 the critical sizing criteria of the composite aircraft wing are shown. While the upper cover is mainly driven by local stability failure, the lower cover (not shown in the figure) is dimensioned by strength failure. A more detailed analysis of the wing and the results can be found in [6].

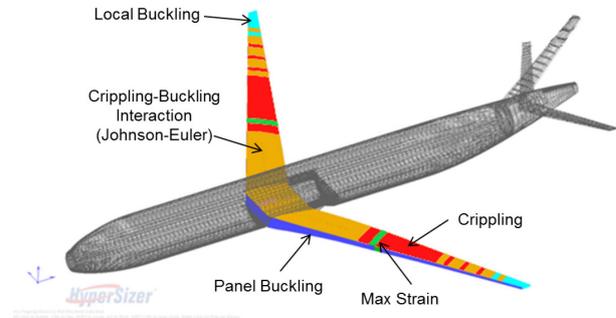


Fig. 14 Critical Sizing Criteria of the composite XRF1 wing with VerSO

6. RESULTS

The following paragraph focuses on exemplary results of the presented sizing process within the MDO process. It must be emphasized that the aim of the paragraph is not to show improved aircraft structure designs rather than demonstrating that the process is able to create reliable results in terms of structural sizing as part of a complex multidisciplinary design environment. As part of the Digital-X project, the XRF1 research configuration was used for sizing. The total number of applied load cases provided from the preceding dynamic loads process was of the order of 50 to 60 different cases containing maneuver and gust loads. In Fig. 15 an exemplary 2.5g maneuver load case is shown where the loads are applied on the load reference axes in the finite element model of the full aircraft. As boundary condition of the model, the last frame of the fuselage was clamped.

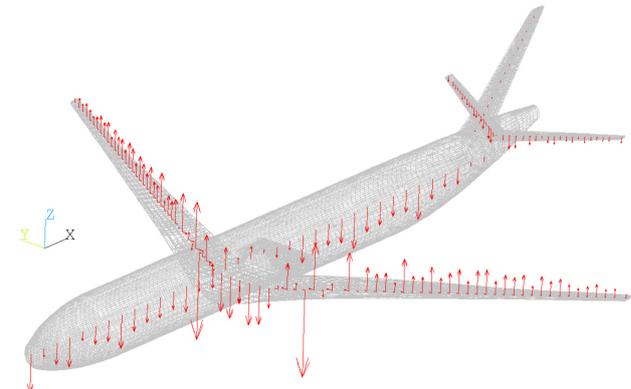


Fig. 15 Load distribution of a 2.5g maneuver load case on the finite element model

To obtain a robust convergence behavior, the total structural mass change threshold was set to 100 kg, which had to be fulfilled for two consecutive iterations. This approach proved to be a good compromise between numerical performance and trustworthiness of the result. As isotropic materials, aluminum 2024 and aluminum 7075 were used which are well-established materials used in industrial aircraft development.

Fig. 16 and Fig. 17 show exemplary results of thickness distributions on the fuselage and the wing. The high thickness of the wing mid spar in the center wing box coincident with the high bending moment near to the wing root and the transmission of internal loads from the wing-fuselage coupling interface. The increased thickness in outboard direction on the lower and upper shell also fits well to the general load distribution of a standard aircraft wing at this particular spanwise position. Due to small loads near to the wing tip, the resulting low thickness at this part of the wing is also plausible.

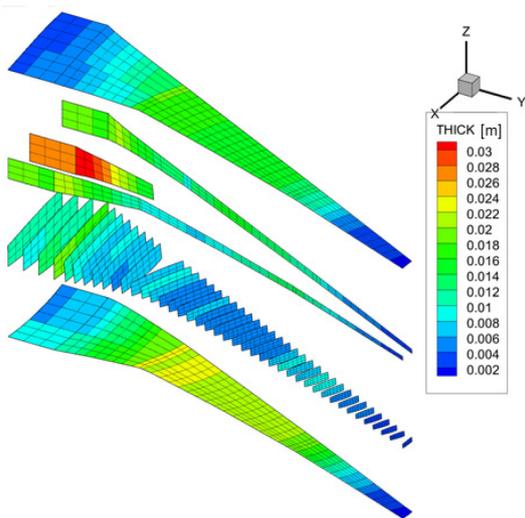


Fig. 16 Spanwise thickness distribution of the sized wing

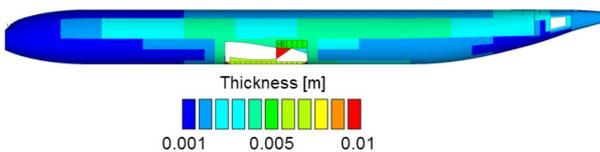


Fig. 17 Thickness distribution of the fuselage structure

In the fuselage, the highest skin panel thicknesses are found in the center fuselage area where the wing loads are transferred, resulting in very high load concentrations. In general, the fuselage part between the wing and the empennage shows higher skin panel thicknesses as the fuselage part in front of the wing. This reflects the general load distribution of a standard transport aircraft fuselage. Additionally, the skin panels at the location of the windows feature higher thickness values compared to surrounding skin panels. This is caused by a larger local stringer spacing thus requiring higher skin thicknesses to prevent local skin buckling in this region. In addition to the thickness distribution, Fig. 18 and Fig. 19 depict the critical sizing load cases for the whole fuselage structure and for the upper and lower shell of the wing. Both figures

show that only a small subset of load cases nearly size the whole wing and fuselage structure.

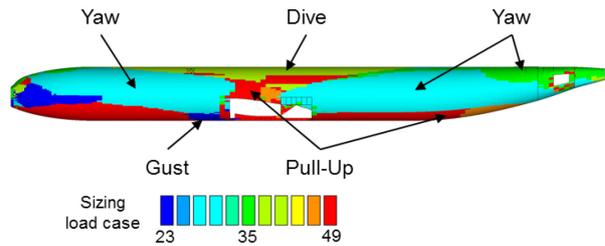


Fig. 18 Distribution of critical sizing load cases for all structural elements on the fuselage

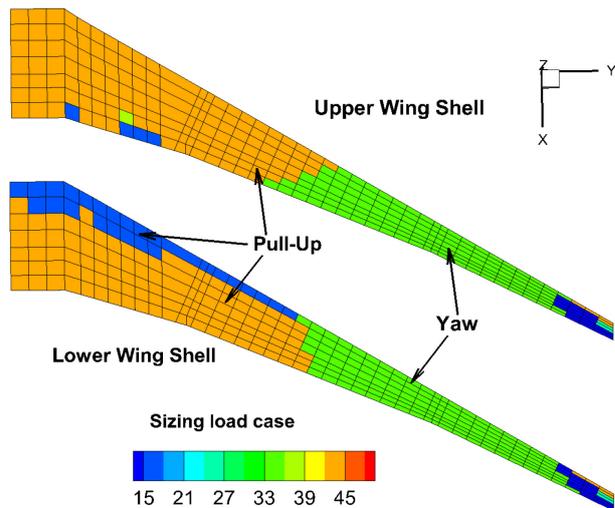


Fig. 19 Distribution of critical sizing load cases on the upper and lower wing shell

In Fig. 20 the structural response of the sized full aircraft model is illustrated for a 2.5g load case, whose global loads are shown in Fig. 15.

7. SUMMARY

In this paper, an automated sizing process of a whole aircraft structure within a MDO process is presented. The main features model generation, coupling of models, loads introduction and static structural sizing are explained in detail. The exemplary results based on a given aircraft configuration demonstrate the usability of the sizing process for isotropic materials within a global MDO process. In addition, an environment for sizing and optimization of composite aircraft structures is presented that is ready for use in future MDO applications.

During the execution of the full aircraft design optimization [7], the runtime performance of the presented sizing process was identified as non-neglectable factor when evaluating the global MDO process runtime for the given optimization task. Therefore, emphasis is put on the optimization of the performance runtime of the numerical sizing process for future developments. Besides, more advanced strength and stability criteria will be investigated and integrated in the future to further improve the quality of the numerical sizing process results.

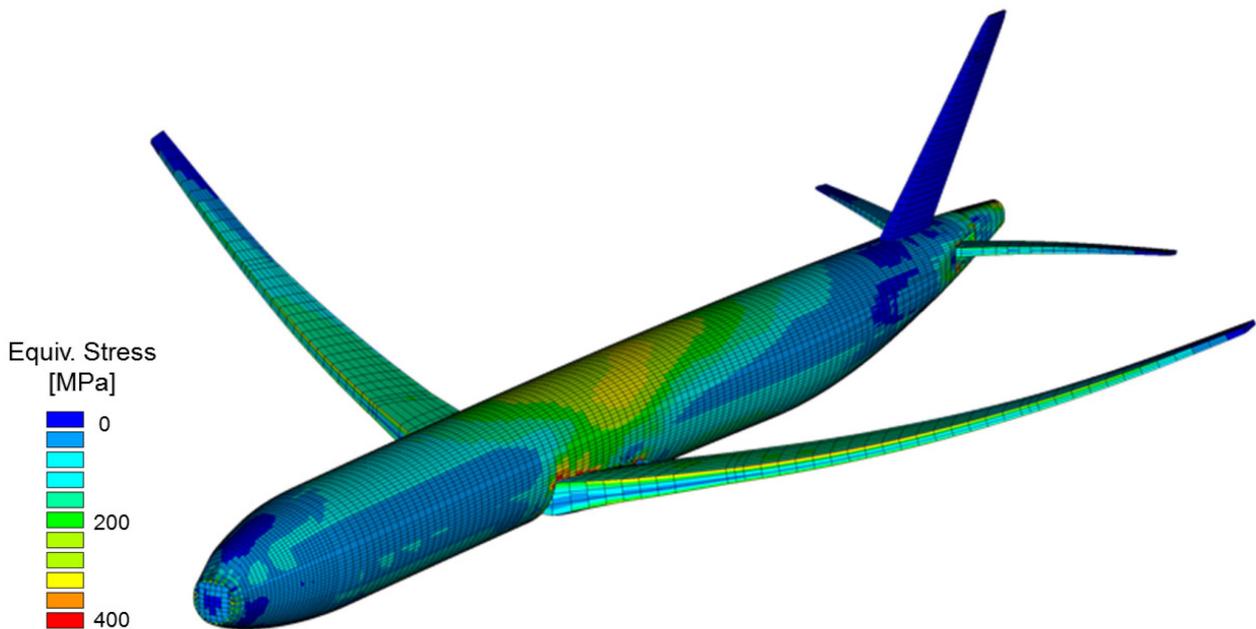


Fig. 20 Equivalent stress distribution of the sized aircraft structure for a 2.5 g pull-up maneuver

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