AUTOMATED SIZING OF A COMPOSITE WING FOR THE USAGE WITHIN A MULTIDISCIPLINARY DESIGN PROCESS

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Abstract

In times of reducing product development cycles and greater economical, safety and ecological requirements on aircraft structures innovative aircraft configurations with new materials and new structural design concepts become more and more important. Although their positive impact on the conventional pre-design methods come to their limits and to an appropriate tools for design and evaluation the need for Multidisciplinary Design Optimization increases. .

In this paper a structural design and optimization module developed for the application in such an optimization process is presented. The module is coupled to the software environment DELiS where structural models mathematically described by finite elements can be automatically created, based on a parametric description. The outer loads, coming from CFD or aeroelastic calculations, are applied to the model and inner loads are calculated utilizing a linear static finite element analysis.

To evaluate the results information about inner loads, displacements, properties and geometry are passed to the commercial software tool HyperSizer. Material parameters like thickness and stacking sequence and also cross sectional stringer parameters are optimized for the given loads on panel level. For sizing and optimization a set of failure criteria is used; analytical equations are available for a reduced calculation time. The capabilities of the module and the sizing approach are demonstrated on a composite wing of a long range transport aircraft.

Keywords Composite wings, Sizing, Design optimization, MDO.

1. Introduction

In order to obtain efficient designs for future aircrafts with respect to requirements like aerodynamics, structure and cost performance, Multidisciplinary Design Optimization (MDO) is a key factor (Sobieszczanski-Sobieski and Haftka, 1997) due to several reasons.

In the current design processes, an optimization of the aircraft configuration is performed with low-fidelity methods based on experience and statistical data. On the other hand new materials like composites are more and more used for primary structures in aerospace industry, i.e. for the A350XWB and the B787. Due to the lack of statistical data for aircrafts with composite wings or fuselages assumptions must be made not taking the potential of CFRP and other composites into account. The same statement is valid for innovative configurations, such as a forward swept composite wing which shows a lot of advantages in fuel consumption compared to conventional designs (Seitz et al, 2011). In addition to that, the structural design concept, which has a significant influence on the mass and stiffness of the wing (shown i.e. by Bach et al, 2014) is neglected in the early design phase. Furthermore, the description of the complex interaction between disciplines like aerodynamic and structural mechanics has its limits utilizing low-fidelity methods. This leads to wrong distributions of loads.

To show the real potential of innovative configurations and new materials in the aircraft pre-design a MDO process with higher fidelity methods is applied.

In order to consider the described challenges in recent years several MDO processes have been implemented in industry and research, i.e. DeBlois, A., Abdo, M. (2010), Schuhmacher, G. et al (2012) or Kennedy et al (2014). To obtain results in an acceptable amount of time a computationally efficient process is required including a well-defined framework. These requirements have to align while using higher fidelity models capable of representing the complexity of its discipline. In structure mechanics finite element (FE) models are commonly used to calculate deformations and inner loads. These results are used as input for analytical equations to determine the Margin of Safeties (MoS) of a pre-defined region and to size and optimize the structure and the design concept. Analytical equations are commonly used due to their low computationally effort compared to complex numerical analysis.

The paper is structured as following. In section 2 the structural sizing and optimization module is explained. The process in a global and local context is illustrated. Furthermore, the relevant failure criteria and the sizing concept are explained. In the third section, the illustrated theoretical process of section 2 is utilized to a wing of a long range transport aircraft. The results are discussed and tested to plausibility. In the fourth section the paper is concluded.

2. Structural Sizing and Optimization Module

2.1. Embedding of Structural Optimization Module in global context

Within the DLR research project Digital-X an automated process for design and optimization of an aircraft configuration including high-fidelity MDO is set up. In the high-fidelity part of the MDO process a more detailed structural sizing of the aircraft wing is performed, and selected load cases are recalculated with CFD analyses to obtain more accurate loads. Instead of SMT loads ("Shear, Moment, Torsion", i.e. described in Hürlimann, 2012), pressure distributions are used, in order to have a more realistic load distribution on the wing outer surfaces. The flow chart of the MDO process is shown in Fig. 1 (Kroll et al, 2014). The model generator within DELiS, which is described in Freund et al (2014), is integrated within the high-fidelity part of the MDO process in. In DELiS a FE -model of the aircraft wing is automatically generated based on the parametric aircraft description CPACS, (Common Parametric Aircraft Configuration Scheme) which has been developed by DLR (Liersch and Hepperle, 2011) to create a consistent database for the multi-disciplinary aircraft design process. In addition to the model generation the external loads, boundary conditions and secondary masses, i.e. high-lift systems or fuel, which are calculated in a previous part of the process, are applied to the FE-model and linear-static calculations are performed to obtain the structural response in form of displacements and internal element loads. The information about the wing geometry, initial properties and loads are then passed to the sizing and optimization module.

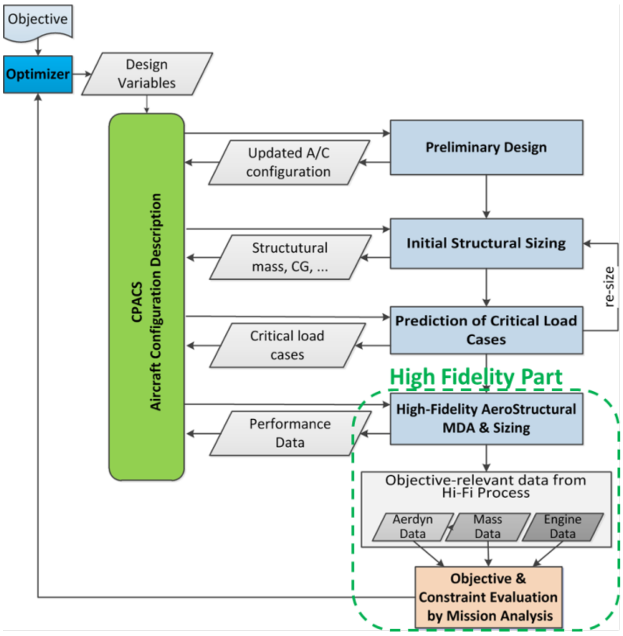

Fig. 1 MDO process in Digital-X (Kroll et al, 2014)

2.2. Sequence of sizing and optimization process

In Fig. 2 the process flow of the sizing and optimization module with its interfaces to DELiS, as well as the process is shown. For the main part of the module, the component sizing, the commercially available software HyperSizer (HyperSizer, 2014) is used. HyperSizer itself is steered by self-developed routines delivering and receiving information about geometry, external loads, results, etc. The different process steps are described in the following subsections.

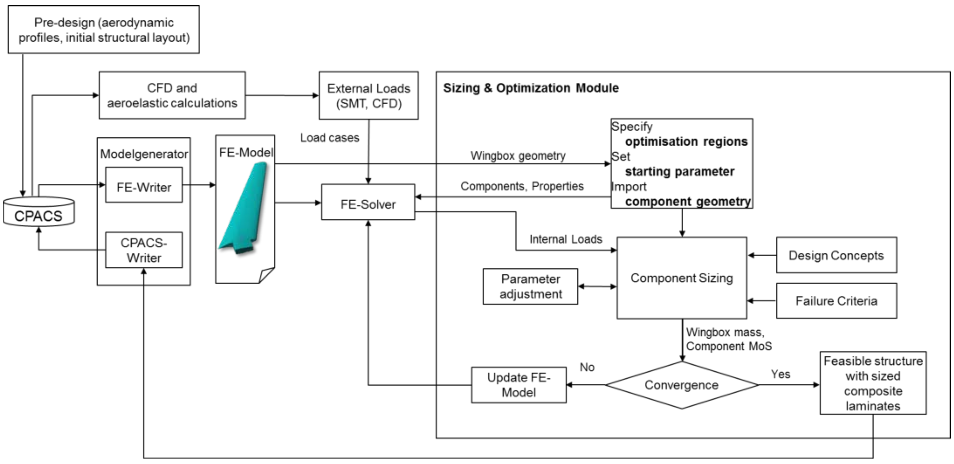

Fig. 2. Process flow of sizing & optimization module (Bach et al, 2014)

2.2.1. Model import and optimization regions

At the beginning of each optimization run the FE-model of the wing is imported into HyperSizer. The model must be build up using specific finite elements which are described in Collier Research (2014) for the supported FE solvers like MSC Nastran or Ansys. With the model the FE properties, the coordinate systems and also the element coordinate systems are imported. Furthermore, FE results in form of displacements and element forces

for each load case which is present in the FE file are imported. These results are used to determine the loads of the design concept objects (i.e. skin field, stringer web, etc.). The methodology is described in section 2.2.4.
During the model import components and assemblies are created in HyperSizer. By default each property defines a component. A component can be defined as a 2D panel or a 1D beam and also is used to span the respective optimization region. In DELiS the wing cover component boundaries are the spars and the ribs. As an example, a wing box with a front spar, a rear spar and 30 ribs, the upper and lower cover is divided into 29 components (optimization regions). All components are sized individually. The geometry of the component (i.e. length, width and curvature) is also calculated during the model import. In addition to that assemblies are created following routines defined by the user. Assemblies usually define a large part, i.e. a wing cover with its components, or set of similar parts like the ribs within the wing box. Assemblies are used to define manufacturing criteria for composite laminates which are considered during optimization. If design parameters should be kept constant for a large number of components, this can also be done on assembly level.

2.2.2. Structural Design Concepts

With HyperSizer it is possible to consider a large number of design concepts for panels and beams. For 2D panels, design concepts like unstiffened, stringer stiffened or sandwich panels can be chosen. For that, different stringer profiles (i.e. T, I, J, …) or sandwich cores (i.e. honeycomb, foam, …) are available. For 1D structures modelled by finite beam or truss elements like beams or discretely modelled stringers different profiles or a so called cap beam, which only represents a longitudinal stiffness, can be used. In general, all geometrical parameters can be modified by the user or they are calculated during sizing and optimization. Usually one design concept is assigned per component. Nevertheless, it is possible to assign two or more concepts and let HyperSizer calculate the lightest design concept for the given loads.
In HyperSizer the longitudinal direction of stringers and the 0° fiber direction are identical and they are oriented along the x-axis of the finite element property. The finite element z-axis is used to determine the laminate stacking direction and the position of the stringers on the panel. Therefore it is necessary to take these conventions into account during the FE model generation in DELiS.
If stringer stiffened design concepts are considered during the optimization it is not necessary that stringers are modelled discretely with finite beam or shell elements. HyperSizer has an internal object model that distributes the component loads between skin and stringer. This allows a simplified modelling of the wing box while the stringer influence on mass and stiffness is considered. Therefore, the same model can be used to investigate different structural design concepts. This also reduces the overall process time because only one FE model has to be created. The described functionality is useable by calling Python objects. An interface between DELiS and HyperSizer has been developed calling the HyperSizer object model with Python scripts. Thus, the tool and its functionality can be implemented in a MDO tool chain,

2.2.3. Failure and Manufacturing Criteria

For structural sizing and optimization a variety of Failure Criteria exists. In addition to global stability and laminate based strength, ply based strength, local buckling, column buckling or mixed mode failures like crippling or Johnson-Euler buckling can be taken into account. For all implemented criteria analytic equations are available, taken from standard handbooks like the HSB (Handbuch für Strukturberechnung), NASA reports or the MIL-HDBK, so a fast MoS calculation in HyperSizer is possible. An interface is present in HyperSizer so the user can extend the available set of failure criteria with self-programmed evaluation routines. Each criterion is evaluated for a load level (usually Limit Load (LL) or Ultimate Load (UL)), which is determined by the user. In addition a required MoS for this load level can be specified by the user, which is $MoS_{required} = 0$ by default. If a failure is allowed to occur before the specified load level is achieved a negative required MoS must be entered. The default settings are shown in Table I. The settings and criteria are only defaults, used to deliver a set of robust settings for a MDO process. For all categories more criteria are available in the program and can be activated by the user, i.e. strength criteria like Tsai-Wu or Hashin.

Table I. Default Failure Criteria Settings

Failure Category	Failure Criterion	Load Level	Applied on
Global Stability	Panel Buckling, uniaxial, biaxial, shear and combined load	Ultimate Load	Panel
	Column Buckling	Ultimate Load	Stringer
Local Stability	Local Skin Buckling	Limit Load	Skin
	Local Stringer Buckling	Ultimate Load	Stringer Web + Flanges
	Composite Crippling (MIL-HDBK-17)	Ultimate Load	Stringer
	Johnson-Euler-Buckling	Ultimate Load	Stringer
Strength	Max. Strain 1 direction	Ultimate Load	Skin, Stringer, Caps
	Max. Strain 2 direction	Ultimate Load	Skin, Stringer, Caps
	Max. Strain 12 direction	Ultimate Load	Skin, Stringer, Caps
Laminate Strength	CAI	Ultimate Load	Skin, Stringer, Caps
	TAI	Ultimate Load	Skin, Stringer, Caps
	SAI	Ultimate Load	Skin, Stringer, Caps

In order to consider complex failure modes in composite laminates it is usual to define design guidelines and manufacturing constraints. Within the optimization process it is possible to consider various criteria. Whenever composite materials are used (not dependent if smeared properties or discrete layups are used) the following rules are applied:

- A minimum of 10% plies of 0° and 90° and 20% of 45° plies in each component
- A maximum value of 70% 0° and 90° plies, 80% of 45° plies in each component while considering a minimum ply share in each direction
- Laminates are symmetric and balanced

If a stacking is considered, additional rules corresponding to best practices for composite laminates become active:

- A maximum of four adjacent plies of the same orientation
- A 45° ply is added every 6th ply (while the laminate remains symmetric and balanced)
- The first and last plies of a skin filed are +45°/90°/-45°
- The first and last plies of the stiffener web are +45°/-45°

By using these optional rules the result of the optimization is restricted before the optimization even started, but the result is closer to a fully designed wing, where additional aspects, i.e. operations and repair are considered. Therefore, an improved statement about the final mass is possible in pre-design.

2.2.4. Component Sizing

In addition to the previous described settings it is necessary to define values for the design variables of a component. In HyperSizer all calculations are performed on a super-stringer object. In Fig. 4 such object is shown for a stiffened panel with I-stringer including all design parameters.

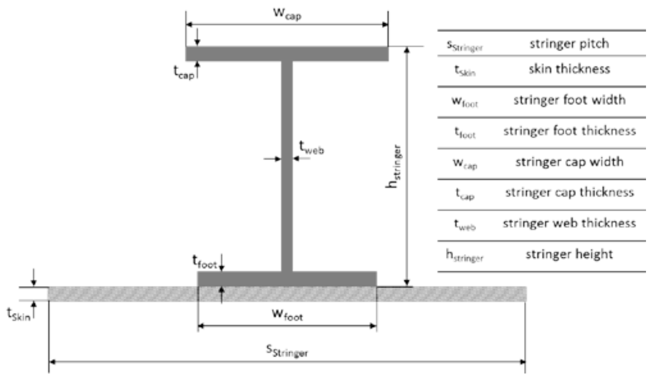

Fig. 3. Superstringer-Object in HyperSizer with I-stringer design variables, no curvature shown

For all parameters a minimum value, maximum value and a number of steps between them must be defined so each parameter has a set of discrete design points. Also all allowed materials for the skin and the stringers have to be defined. If composite materials are used all thicknesses are related to the laminate stacking so the final thickness is an integral multiple of one layer thickness. The number of design candidates for each component is calculated by summing up the number of permutations of the design points. Afterwards, the design candidates are arranged according to the structural weight, starting with the lightest solution. For each design candidate allowables are recalculated if necessary, i.e. the local buckling allowable of the stringer web for different web heights.

The composite layers are determined in a multi-step process. In a first step the ply distribution in 0°, 90° and ±45° direction is calculated, including an overall thickness. Afterwards, a set of discrete layups is created using the previously calculated values as input parameters. Hereby, the stacking is varied and the component is recalculated. This allows a more accurate representation of the laminates bending stiffness for stability analysis because the bending stiffness depends on the stacking sequence. In both steps the design rules explained in section 2.2.3 are taken into account.

As mentioned before the stresses of the individual objects of the stiffened panel are calculated using the results of the linear static calculations. In the FE model the property of a component represents the overall panel stiffness and, therefore, the results represent the total panel stress and strain. Using the internal element loads N_x and the panel stiffness A_{11} the total panel strain ε_x is calculated. Afterwards, the individual object stresses σ_i are calculated using the stress-strain relationship and the individual object stiffnessesE_i, see in Equations (1) and (2).

$$\varepsilon_x = A_{11}^{-1} N_x \quad (1)$$

$$\sigma_i = E_i \varepsilon_x \quad (2)$$

The resulting object loads $N_{x,i}$ are then calculated by multiplying the stress with the respective thickness t_i.

$$N_{x,i} = \sigma_i t_i \quad (3)$$

This methodology is also applied to calculate the shear loads and the tension / compression loads in lateral direction. The calculation is performed for each design candidate and therefore, skin and stringer are not optimized separately. With this approach an optimum skin-stringer combination is calculated for the given internal loads for each component and the effects of shifting stiffness between skin and stringer become visible. The object loads calculated with Equation (3) are the input loads for the failure criteria.

All components are calculated individually and for each component the lightest design candidate where all MoS are equal or higher the required MoS is determined. The properties in the FE model are updated after the optimization. Only the properties of components imported into HyperSizer are updated. If a stiffened panel design concept is used the stiffness matrix of the whole panel is written into the FE file so the combination panel skin and the stringer stiffness are considered. For beams cross sectional properties and the material constants are updated. A new FE calculation is performed and the new internal element loads are imported into HyperSizer. This is done until convergence is achieved.

The convergence criteria are shown in Equation (4). As main convergence criterion the change in mass Δm between two iterations is used. The value can be specified by the user. Nevertheless, as constraint all MoS checks must be positive.

$$\begin{aligned} &\Delta m = \frac{m_i - m_{i-1}}{m_{i-1}} \\ &\text{all } MoS > MoS_{required} \end{aligned} \quad (4)$$

When convergence is achieved, the CPACS file is updated using the new properties and the wing deflection is given back to the aero-structure coupled process so new external loads can be calculated.

2.3. Module Capabilities

With the described set up of the structural sizing and optimization module a broad variety of tasks and investigations can be performed. In addition to an analysis of an existing structural design, a sizing and a panel

optimization with a fixed structural design (i.e. fixed stringer type and spacing) is possible and is considered as a standard application for the module. An exemplary application is presented in the next chapter.
Due to the extensive number of available design concepts and the computational costs, the structural optimization module can be used within design studies to compare different structural designs for a wing. Bach et al (2014) have performed a design study where a forward swept laminar wing has been optimized using two structural designs (T and I stringers) of the wing covers. The results have shown that with almost no difference in structural mass the stiffness significantly changed between the designs. Another application used by Dähne et al (2014) is the effect of failure criteria application and their safety factors on the wing mass. The used configuration was the forward swept composite wing where the main stiffness direction and ply distribution was optimized in a previous project with a simplified approach for the structural design concept: stringer where smeared into the composite properties of the skin using the “Bending Moment of Inertia factor” of the MSC Nastran PSHELL card (MSC Software, 2013). A more accurate representation of the design concept showed a good correlation between the calculated deformation and the previously determined deflection and twist. One the other hand the application of a more realistic set of failure criteria and a more accurate representation of the design concept lead to an increase in wing box mass of approximately 25%, which is usually an input parameter for performance analysis of an aircraft.

3. Exemplary Application on a composite wing for a long range aircraft

To show the capabilities of the optimization process, a composite wing box of a long range transport aircraft is optimized. As aircraft configuration the Airbus XRF1 configuration is utilized, which is the use cases for the DLR project Digital-X (Kroll et al, 2014). The configuration is designed to carry ~350 passengers with a maximum range of 8000 nm. A CAD visualization of the XRF1 configuration is shown in Fig. 4. The structural layout including number, position and orientation of ribs and spars was determined by Airbus.

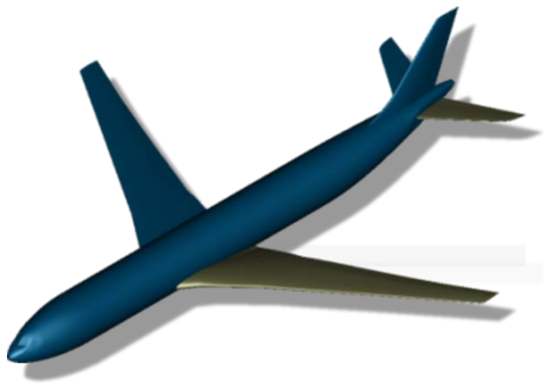

Fig. 4. Airbus XRF1 Configuration (Führer et al, 2014)

In the following part the definition of the design for the present optimization is specified. Stiffened panels are defined for the wing upper and lower cover and the ribs. The T-stringer match the design concept of the A350XWB (Aero News, 2011), which is a similar aircraft configuration. The ribs are also stiffened with T-stringers. A constant stringer pitch of $\Delta s = 210$ mm is assumed for the present optimization. The spars are evaluated as unstiffened laminate design concept. As material state-of-the-art CRFP aerospace prepreg is used. The default failure criteria settings shown in Table I are used as optimization constraints, whereas post-buckling and failure modes from joints are not considered. The optimization is performed using 17 load cases (maneuver and gust cases) which are provided by another sub-process in Digital-X (see Fig. 1, prediction of critical load cases). In the SMT loads the aerodynamic forces are considered as well as loads coming from engine, fuel, landing gear, secondary masses (i.e. high-lift systems) and their inertia loads. For the optimization process a FE-model of the complete XRF1 aircraft configuration is used, which considers the interface loads between fuselage and wing box. Overall the wing box consists of 241 optimization regions.

3.1. Mass results

In Fig. 5 and Fig. 6 the convergence plot and the mass share of the wing box assemblies are shown. The wing box mass at the last iteration is $m_{wingbox,final} = 6361{,}37$ kg.

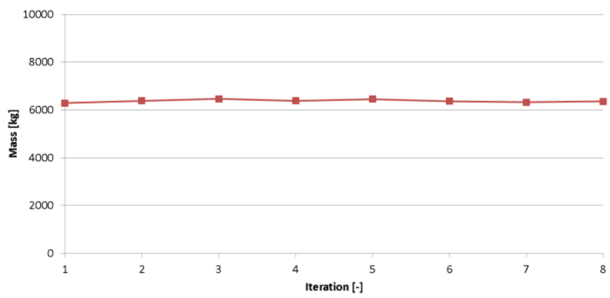

Fig. 5. Mass Convergence of XRF1 Wingbox

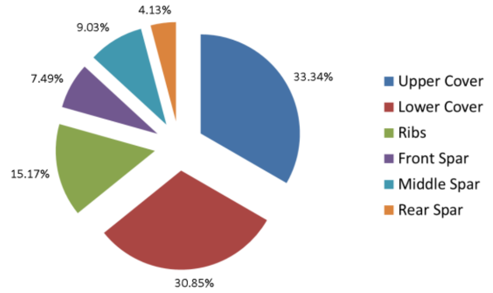

Fig. 6 Share of Wingbox Assemblies on Total Wingbox Mass [%], last iteration

As expected the wing upper and lower cover are the heaviest assemblies, followed by the ribs and the middle spar. The huge mass of the middle spar was not expected, due to the fact that it ends after the first third of the wing box span, Fig. 7, (counted from wing symmetry plane). Further detailed investigations are therefore necessary to investigate this result.

3.2. Thicknesses

In Fig. 7 the thickness distribution of the wing box is shown. The numbers in the color boxes represented the number of components with these thickness values. Corresponding to the relatively high mass of the middle spar, the greatest thickness is present at the middle spar in the center wing box. For all spars the thickness decreases from center wing to wingtip. Near the wingtip the minimum skin thickness of $t_{min} = 1.778$ mm is achieved for some components. The minimum skin thickness can be calculated using the composite design rules described in section 2.2.3 (three default first and last plies, at least 10% share in each direction) in combination with the layer thickness of the used material, which is $t_{layer} = 0.254$ mm. Therefore, at least seven layers are present in each skin field.

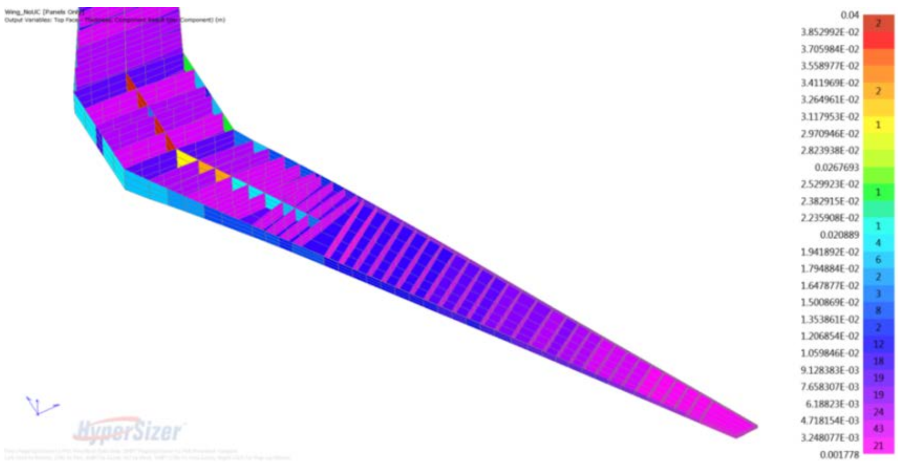

Fig. 7. Skin Thickness Distribution in Wingbox, no upper cover shown, thickness in [m]

3.3. Critical Failure Criteria

In Table II an overview is given about which failure criteria is critical in the respective Assembly.

Table II. Number of Components dimensioned by Failure Criteria

Criterion	Local Buckling	Global Buckling	Ply Strength	Laminate Strength	Crippling	Johnson-Euler
Nr. Of Components	36	91	65	0	16	33
Upper Cover	3	0	1	0	16	33
Lower Cover	4	1	48	0	0	0
Ribs	29	12	0	0	0	0
Front Spar	0	40	1	0	0	0
Middle Spar	0	12	0	0	0	0
Rear Spar	0	26	15	0	0	0

The results shown in this table correspond very well with the present loads and the used structural design concepts. The design of the upper cover is mainly influenced by shear and compression loads. Global buckling is prohibited in the upper cover by the usage of stringers but local buckling failure and failure modes resulting from local stability failure (Crippling, Johnson-Euler buckling) are dominating the upper cover. In the lower cover more components are sized by ply strength failure due to high tension forces present in the load cases with high positive acceleration. The unstiffened spars are primarily dimensioned by global buckling, especially by shear buckling. It needs to be mentioned that the local buckling MoS is only calculated if stringers are presented in a component.

3.4. Skin-Stringer coupled design optimization

In Fig. 8 the internal loads in 0° fiber direction for load case 6, which is mainly critical for the upper cover, are shown.

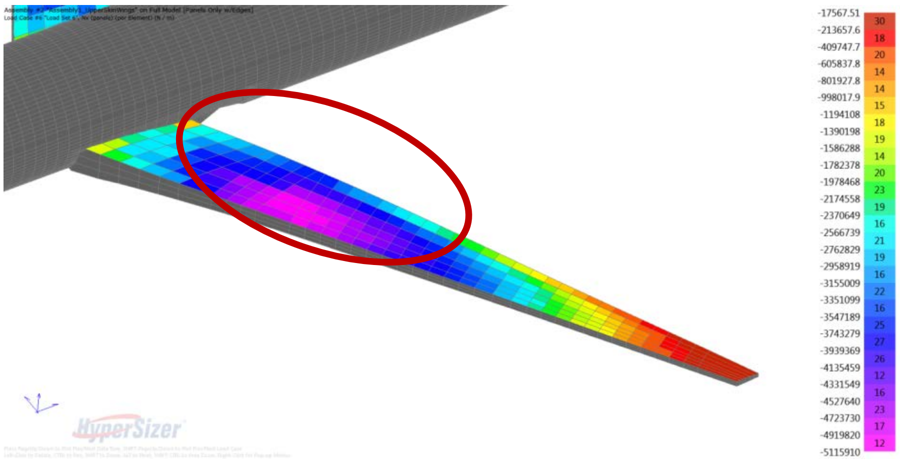

Fig. 8. Element Forces Nx [N/m], Loadcase 6

The highest compressive forces (a magnitude above the shear loads) are near the end of the middle spar, the area is marked.

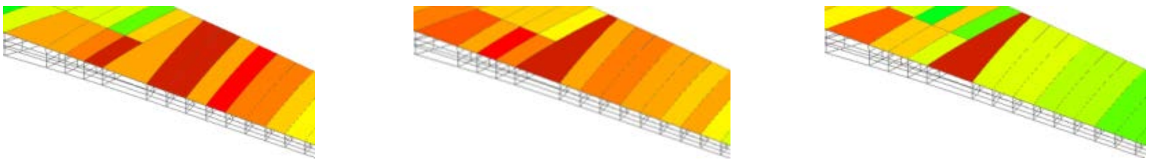

Fig. 9. Skin thickness (left), Stringer web thickness (middle) and stringer height (right) of wing upper cover

In Fig. 9 the skin thickness, stringer web thickness and the stringer height are shown for the area near the end of the middle spar. The spectrum is left out for visualization purposes. In general green fields mark a low value of forces and red fields a high one. The elements with the highest compression force are in the trapezoid one in the middle of Fig. 9. Nevertheless the skin thickness is greater in the adjacent components. In order to reduce the loads in the skin the stringer web is increased in order to reduce the number of skin plies, which results in a lighter panel. This effect is not visible if skin and stringer are optimized separately.

Conclusion

In this paper the set up and capabilities of a structural sizing and optimization module for the usage in MDO process chains is described. The module is capable of sizing and optimizing aircraft structural components using composite or metallic materials with respect to a broad variety of failure criteria which are chosen by the user. The calculations are performed on panel level taking the design concept, i.e. stiffened or unstiffened panels, into account. An optimization of the XRF1 wing with composite materials was performed showing the capabilities. In addition to the general functionality it is necessary to consider more dimensioning failure criteria like Crippling or Johnson-Euler buckling. This is visible when evaluating the critical component's failure criteria. Furthermore, the positive effect of having a combined skin-stringer optimization was discussed.
All calculations were performed without changing the external loads. In future, the sizing and optimization module, including the DELiS environment, will be implemented in the Digital-X MDO chain and aero-structure coupled calculations will be performed during the optimization.

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