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Design "Feasilization" Using Knowledge-Based Engineering and Optimization Techniques

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The multidisciplinary design optimization process can be supported by partial automation of analysis and optimization steps. Design-and-engineering engines are a useful concept to structure this automation. Within the design-and-engineering engines, a product is parametrically defined using knowledge-based engineering. This parametric product model needs to be initiated before global multidisciplinary optimization can be performed. The "feasilization" is done by the initiator component in the design-and-engineering engines that simulates the heuristic methods normally used by designers to estimate the first values for the parameters and variables describing their designs. The initiation of values for structural parameters and variables is elaborated for a sample composite-stiffened panel structure. It is shown that the initiator function of the design-and-engineering-engine concept can be implemented on the basis of optimization techniques using simplification of the design requirements, simplified representations of the design options, and the class of so-called schematic models to mimic the designer's job in the preliminary sizing phase of the design. An implementation of the initiator is used in a sample design-and-engineering engine for aircraft vertical tail design.

Nomenclature

A = area
b = stiffener spacing
cg = center of gravity
D = bending stiffness of a laminate
E = modulus of elasticity
$\bar{E}$ = equivalent modulus of elasticity
h = stiffener height
I = second moment of area
L = panel length
M = moment
N = load
n = number of plies in the skin
ns = number of plies in the stiffener
p = load intensity
r = local radius
S = material maximum shear stress
t = thickness
W = weight
X = material maximum normal stress in the x direction
Y = material maximum normal stress in the y direction
ν = Poisson's ratio
ρ = material density
σ = normal stress
τ = shear stress

Subscripts

x, y, z = directional axes
xy, xz, yz = planes of action

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I. Introduction

MULTIDISCIPLINARY design optimization of aircraft is a complex process, relying heavily on the preparation and updating of various models. Currently, designers are faced with too many design options, too many tools, and too little time to find a proper set of feasible design options. Most tools depend on humans for the preparation, starting, and interpretation of the analysis, although it would be desirable for the designer to only specify and interpret the analysis. These processes can be improved considerably by automation of repetitive tasks and by enhanced parametric product and process modeling, to relieve designers from non-value-adding activities, making more time available to exploit their creativity and engineering skills.

In a previous paper [1] by the authors on this topic, it was shown that knowledge-based engineering (KBE), which is a proper combination of object-oriented programming, rule-based instantiation of objects, and a geometry engine, allows parametric modeling in the optimization sense. Using KBE, a so-called multimodel generator (MMG) can be built that is able to (re)generate views on the product for each set of product parameter values. The use of this MMG in the concept of design-and-engineering engines (DEE) [1–3] was also explained. Further, the idea of an agent-based software framework [4] for efficient communication in a multidisciplinary design environment was introduced. Finally, the paper concluded that initiators [5] must be developed for all aircraft elements of importance in the conceptual design stage. These initiators fit within the conceptual design process and are responsible for the selection and "feasilization" of the design parameters of the parametric product model, to make global optimization possible.

During the conceptual design process, the designer iteratively changes the design options in the search for a (near) feasible set. Optimization methods can support these otherwise "manual" search methods. However, current optimization theory application is restricted to a solution domain defined by a selection of design variables, although optimization theory makes a distinction between design variables and design parameters. For aircraft design problems, variables specify limited differences within an aircraft configuration (set of design option selections), and parameters relate to complex variations within a configuration and intertype differences (differences in configuration). During an optimization, parameters are normally fixed and the optimization is limited to finding a combination of values for the design variables that will minimize or maximize an objective function such as weight or range. The mathematics required to optimize at a higher level and support the choice between different design options are not available, nor are product models available that allow variation between configurations during the optimization process. Availability of such optimization possibilities would be of great help to the designer, making automation of the search within a selected design space possible. This would relieve designers from many of their non-value-adding activities (e.g., geometry manipulations), making more time available to explore the design space.

The search within a selected design space can, in theory, be automated using optimization methods via describing the design options in design variables and defining a proper set of constraints and a relevant objective function. However, a useful end result will be obtained only if the initial values of the design variables are describing a feasible solution. To address the problem with optimization techniques, the designer needs feasilization of the design options. This feasilization aims at finding a feasible set of design options and design variable values and will only be successful if it follows the way a human approaches the multilevel, multidimensional characteristics of the design problem.

This paper describes the development of a composite structure initiator that is responsible for the initial sizing of structural components that have blade-stiffened composite panels as their building block. It allows the parametric model (e.g., of composite aircraft tails based on blade-stiffened panels) to be instantiated with values that yield a feasible design. KBE is used to capture the process of extracting a structural view (structural idealization) from the complete product model. The resulting structure initiator can be used as an initiator template for other disciplines. To make the paper self-contained, a short description of the DEE and its constituents is given first.

II. Overview of the DEE Concept

A DEE is defined [1–3] as an advanced design environment that supports and accelerates the design process of complex products through the automation of noncreative and repetitive design activities. Figure 1 shows the DEE concept. The main components of the DEE are as follows:

1) An initiator is responsible for providing feasible starting parameter values for the instantiation of the (parametric) product model, addressed in this paper.

2) A MMG is responsible for instantiation of the product model and extracting different *views* on the model in the form of report files, capturing discipline specific model information [e.g., aerodynamic mesh or a finite element (FE) model] to facilitate the related expert tools.

3) Analysis (expert) tools are responsible for evaluating one or several properties of the design (e.g., structural response, aerodynamic performance, or manufacturability).

4) A converger and an evaluator are responsible for checking convergence of the design solution and compliance of the product's properties with the design requirements.

The definition of the design options is based on HLPs [1]. These are functional building blocks that allow the user of the DEE to define a product in a certain product family, which encompasses a structured set of HLPs. These functional blocks are basically sets of rules that use parameters to initiate objects that represent (part of) the product under consideration. In addition, engineering processes can be applied to the initiated objects by capability modules. The HLPs are like rubberized LEGO™ and can be individually tailored, due to their fully parametric definition. They are created in a knowledge-based engineering environment based on object-oriented programming, rule-based reasoning, and geometric modeling capabilities. The KBE environment gives access to a parametric geometric modeler, which allows the rule base to perform all geometric operations normally available in a computer-aided design program. The KBE application, based on the HLP and capability-module concept to define a product and to generate different views (models) on the product, is called the MMG. Different classes of primitives are used to instantiate a certain model, which are, in the case of an aircraft, the wing, fuselage, engines, and connection-element primitives.

This paper discusses a particular DEE component: the initiator. The focus will be on a structure initiator for a specific structural concept: namely, a blade-stiffened composite panel. For evaluation, we consider an example aircraft and assume the blade-stiffened

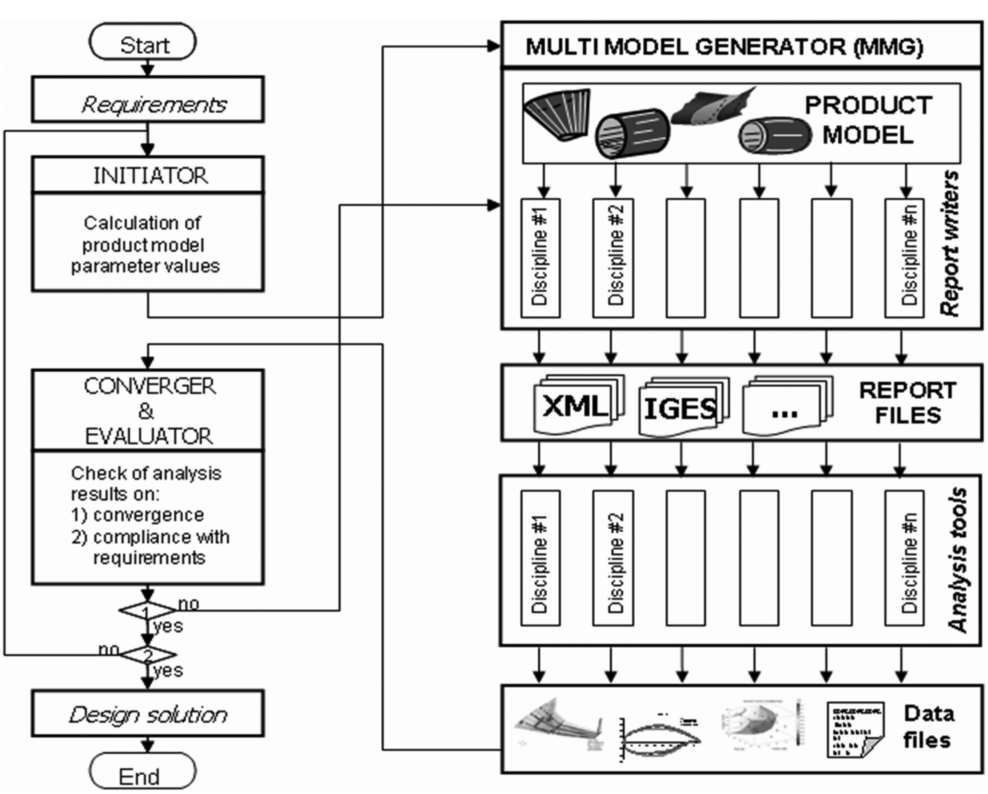


Fig. 1 DEE main design process flow (left) and multimodel generator and the analysis tools (right).

composite panels to be selected as the structural concept for the wing and empennage.

III. Initiator Concept

Any attempt to start a multidisciplinary design optimization should ideally start from an initial design vector that is (close to) a feasible solution of the optimization problem. The initiator component of the DEE is responsible for providing such an initial design vector. Although the result of the initiator does not necessarily provide a feasible solution of the actual problem, it provides the best initial design vector that can be obtained with information and engineering knowledge available at the start of the optimization problem. At the start of a design, the designer has little to no knowledge of the design. To find an initial feasible design solution (start vector), the designer normally simplifies the design problem by using a reduced set of requirements, a simplified set of design options, and schematic behavior models (see Sec. IV). Iteratively, the designer decreases the level of problem simplification to eventually find a feasible solution that encompasses all requirements, design options, and detailed behavioral models.

This iterative process captured by the initiator component of any DEE can, in fact, be modeled by one or multiple DEEs (see Fig. 2). Each DEE has a unique level of problem simplification. The number of DEEs required for a specific design problem depends on the top-level requirements, the problem complexity, and the available resources. During the design process, these DEEs are applied sequentially. The order in which they are applied depends on the individual DEE result. If a certain DEE does not find a feasible solution space, the previous DEE process must be repeated, taking into account the infeasible design space. This process continues until the final DEE process yields a feasible solution.

To solve a specific DEE design problem, because no mathematics are available at a higher level, the designer uses a heuristic but logical approach to reduce the design space through decomposition of the design problem, dividing the design problem in a set of primitive levels. The design problem is simplified per primitive level, from lowest-level primitive to highest-level primitive. The resulting simplified problems are attacked via a trial and error method (which is principally the same as applying an optimization method) to size the design variable values. An engineer uses sizing to find the optimal design variable values (e.g., in the case of a beam design to a single compression load case), whereas a designer iteratively changes the beam height and width to find a combination that meets the requirements and failure criteria while minimizing the objective value. Design variables specify difference within a solution topology (continuous variables); that is, a change in wing span or skin thickness. Design parameters specify differences in solution topology (discrete variables); that is, on a high aggregation level, one can be comparing a blended-wing configuration with a conventional wing configuration. On a low aggregation level, this may imply comparing riveting with bolting. Iteratively, the primitive levels are addressed, from low-level primitive (LLP) to high-level primitive (HLP), such that a feasible solution is found at all primitive levels.

We call this process feasilization. Next, the application of feasilization in structural initiator is discussed, guided by an example wing-type structure design case.

IV. Wing-Type Structure Initiator

As stated before, during the initiation process, the wing-type structure design problem is simplified by using 1) a reduced set of the requirements, 2) a simplified description of the conceptual design options, and 3) schematic models to allow basic sizing.

1) The requirements are divided into functional requirements, performance requirements, and constraints. The functional requirements are used by the structural designer to select design options (in this case, load path elements), the performance requirements in structural design are described in load cases, and the constraints are related to failure criteria such as structural buckling and material strength. The initiator translates the functional design in a feasible design using reduced performance and failure criteria.

2) The product design options can be hierarchically structured. Through decomposition of the product, the design options per primitive level are identified. A simplified version of the real problem is obtained by including only the driving design options in this decomposition process. The resulting LLPs have a reduced design domain, which can be described by fewer parameters than the HLPs and can be initiated with optimization techniques. Although the single HLP design problem is now replaced by multiple LLP design problems, the design optimization is assumed to require fewer resources. The tradeoff between the size of the design domain and the number of models to be analyzed is visualized in Fig. 3.

In the Airbus conventional passenger aircraft model (ACPAM) MMG, the wing and empennage are modeled as a combination of one or more wing trunks. This set of wing trunks is used to derive a

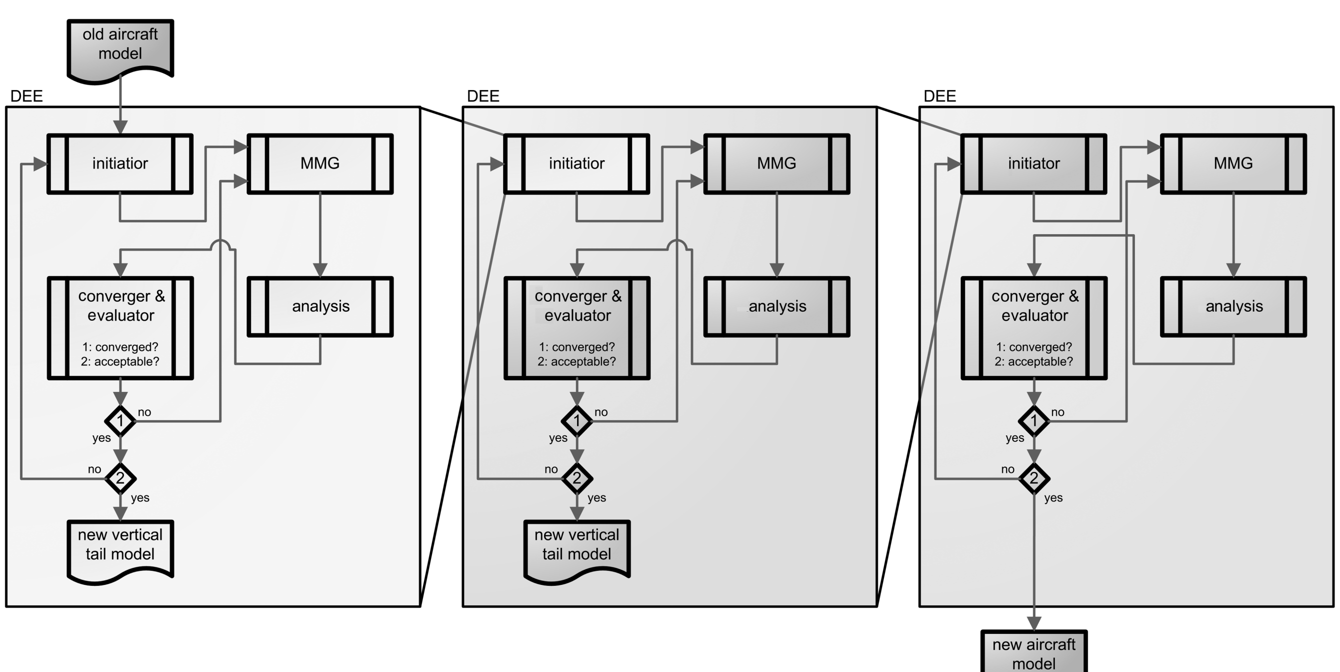


Fig. 2 Initiator component of a DEE consists of multiple DEEs of simplified problems.

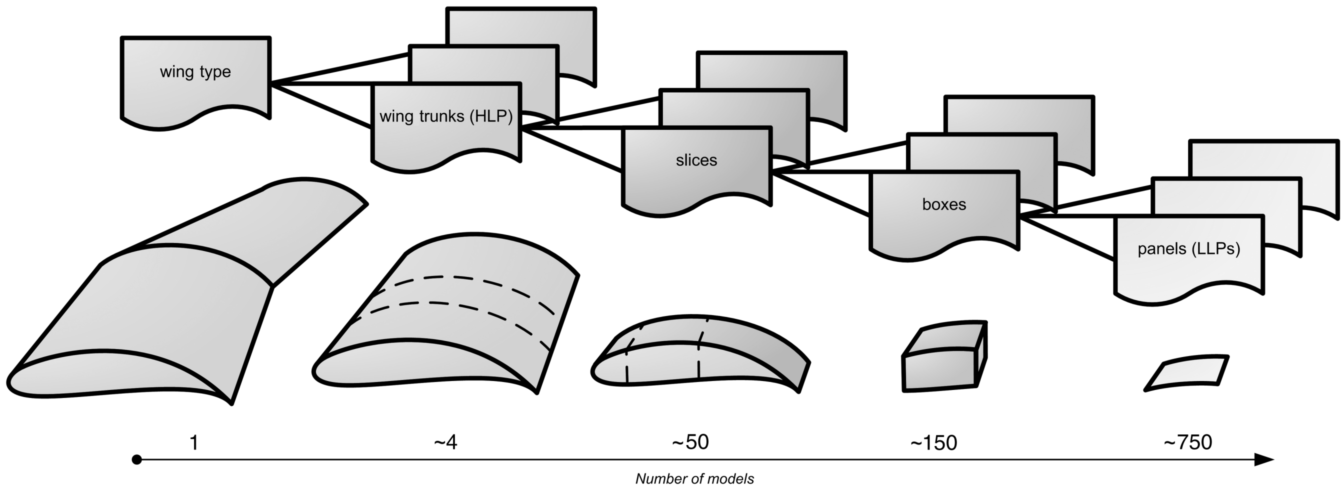


Fig. 3 Relation between the number of models and the size of the design domain.

structural view on these components that is suitable for the structure initiation process. For structural sizing, it is best to decompose the structure according to the main design requirement: the expected structural loading. Only the central box of the wing is considered, because this is assumed to be the main load path. The main load path in the central box is from tip to root. The structure is partitioned accordingly in a set of constituent box structures, illustrated in Fig. 3, using the rib positions as cutting planes. The airloads on a wing box are taken up by skin panels and transferred via the rib panels to the spar panels, which transfer them to the fuselage [6]. Therefore, the box is decomposed in a set of skin, spar, and rib panels. The panel primitives could again be decomposed, but here they are identified as the LLPs (see Sec. V). Because the loads vary within each panel, for simplicity, the panel geometrical center is taken as representative of the complete panel. Finally, the loads are translated in compression, tension, and shear load intensities for each panel.

One of the complications with the decomposition of a design problem is that it assumes that the sum of partial solutions forms a feasible overall solution; in reality, this is often not the case. Interference between the partial solutions can invalidate the applicability of the sum of partial solutions. This problem is addressed by the structure initiator: after the LLPs are designed, feasilization of higher-level primitives (the boxes) is performed, incorporating checks on failure modes associated with the combination of partial solutions (e.g., rib crushing due to bending of the wing). A method suggested by Rothwell [7] is used for rib crushing, which uses skin-panel deformations to estimate the rib compression load. The feasilization of the panels is ordered: first the skin and spar panels, then the rib panels. The implementation is presented in Sec. V.

3) The behavior analysis of the LLPs is performed by using schematic behavior models (see Sec. V.D). These models are based on approximations of system boundaries, system geometry, and system functions. The system boundaries are schematized via approximation of the polygonal edge of the panel with a rectangular shape, and interaction with surrounding structure is left out. The system geometry of the real product and the model differ (e.g., a curved skin is approximated as a flat plate). Finally, only the main system functions such as load paths are used for model definition, and functional interference with surroundings is left out (e.g., behavioral coupling between different LLPs).

This method follows the engineering practice in which handbook methods or approximate computer models are used to analyze the structural components. In this case, analytical and FE schematic models are used side by side.

The initiation of the trailing- and leading-edge structures would require another initiator that, for instance, takes into account load cases such as bird impact. This is out of the scope of this paper. Because the common LLP identified in the wing-type structure is a panel, the feasilization process of the panel structure is elaborated next.

V. Panel-Structure Initiator

In reality, multiple design options would be considered for panel design. For simplicity, however, only one panel design option is considered in the example: namely, the blade-stiffened composite plate [1]. In the case of multiple design options, each one must be elaborated separately, after which the best-performing (in this case, lowest-weight) design option is selected. In the schematic models used to size LLPs, the panels are assumed to be rectangular and flat, simplifying the problem to two dimensions. The approximation of the real panels by rectangular panels is done most conservatively by taking the smallest dimensions in width and the largest dimension in the length, illustrated in Fig. 4.

Because we are looking for an initial solution, it is assumed that the most prominent material and structural failure modes will be sufficient to obtain an initial (near) feasible solution. The failure criteria taken into account are strength and stability. No deformation constraints or design-life-constraint-related cyclic loading are taken into account. The generic panel concept is illustrated in Fig. 5.

A. Design Requirements

The design requirements used are a set of static applied load cases that specifies the minimum required *allowable* loads of the structure.

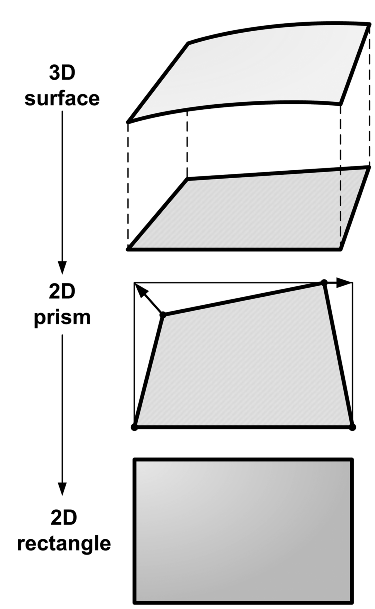


Fig. 4 Panel shape approximation.

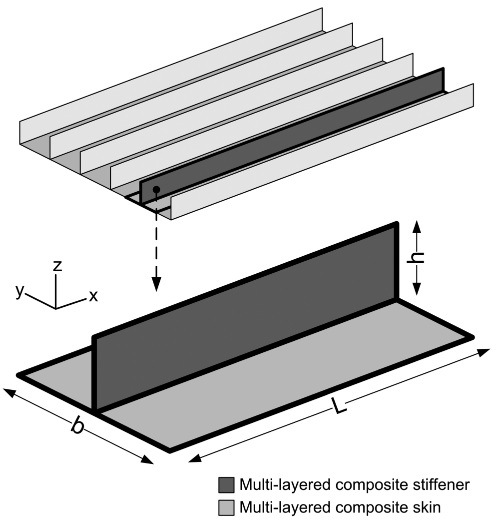


Fig. 5 Generic panel concept.

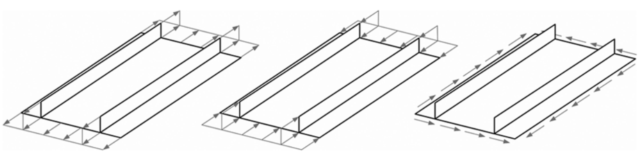


Fig. 6 Panel loading: tension, compression, and shear.

Dynamic load cases are translated to a set of static load conditions, specifying the maximum occurred loads. The loads on the structure (shear forces, bending moments, and torsional moments) are translated into load intensities (N/m) of tension, compression, shear, or a combination of those (see Fig. 6).

B. Design Options

The design options are fixed during the design process and formalize initial design decisions [e.g., higher-level primitive variables such as panel external dimensions (in this case, rib distance), panel-structural concept, stacking sequence, and material choice]. The design variables define the design space and they directly relate to the decisions normally made by the human designer in the sizing process (see Table 1). The design variable vector is a set of the design variables specified in Table 1. It is up to the designer to decide if panel characteristics such as panel length are fixed or free during the optimization.

C. Schematic Behavior Models

The behavior analysis encompasses the failure mode's material strength and structure stability. In the case of a composite blade-stiffened panel, four stability failure modes (initial buckling) are important: global panel buckling (flexural buckling), local skin buckling, local stiffener buckling, and skin shear buckling, illustrated in Fig. 7. According to Rothwell [7], in shear, only the skin will buckle, and so no shear buckling constraints are defined for the stiffener.

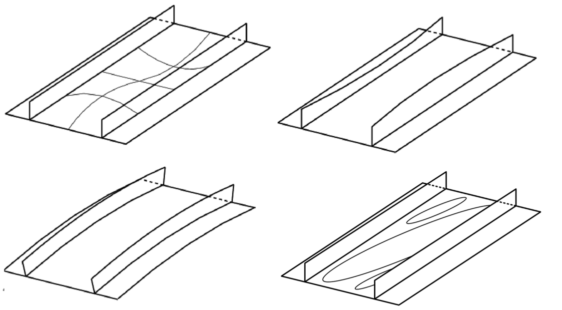


Fig. 7 Buckling modes of a blade-stiffened panel in compression, respectively, skin, stiffener, and panel buckling, and panel buckling in shear.

The ability to prepare FE models and launch an external solver is integrated in the sizing tool. This functionality is used to evaluate the constraints describing the initial stiffener buckling load in compression and the initial skin buckling load in shear. The FE program UNA [8] is used as the solver.

For results presented in Sec. V.E, the analytical schematic models are used to calculate the values for the constraints, because they require fewer resources. The FE models are used for verification of the analytical constraints. The models are handled hereafter.

1. Material-Strength Schematic Model

The strength failure modes of the composite material are determined with a schematic model based on the classical laminate theory (CLT). The stresses in the separate plies are calculated with CLT using the A matrix, the ply-stiffness matrix S, and the angle between its principal material axes and the loading axes system. The model uses average stresses and only considers material failure. Material failure is evaluated with the Tsai–Hill (TH) criterion:

$$\mathrm{TH} = \frac{\sigma_x^2}{X^2} - \frac{\sigma_x \sigma_y}{XY} + \frac{\sigma_y^2}{Y^2} + \frac{\tau_{xy}^2}{S^2} \qquad (1)$$

2. Panel Compression-Stability Schematic Model

Global panel stability is evaluated with the following [1] formula:

$$\sigma_{\text{skin buckling}} = \frac{2\pi^2}{b^2}(t_{0_{\text{skin}}} + t_{\pm 45_{\text{skin}}})(\sqrt{D_{11}D_{22}} + D_{33}) \qquad (2)$$

Table 1 Optimization variables, parameters, and bounds

			Bounds, mm	
			Lower	Upper
Design variables	b	Stiffener spacing	50	100
	h	Stiffener height	10	60
	n_0	Number of plies in the 0-deg layer of the skin	1	80
	n_{45}	Number of plies in the ±45- deg layer of the skin	1	80
	ns_0	Number of plies in the 0-deg layer of the stiffener	1	80
	ns_{45}	Number of plies in the ±45- deg layer of the stiffener	1	80
Design parameters	L	Panel length		
	W	Panel width		
	Stacking sequence	Skin	[0 deg nx, ± 45 deg nx, 0 deg nx]	
		Stiffener	[0 deg nx, ± 45 deg nx, 0 deg nx]	
	Material	Toray T700 CF		
	p.pxt	Tension load intensity in the x direction (see Fig. 5)		
	p.pxc	Compression load intensity in the x direction		
	p.pyt	Tension load intensity in the y direction		
	p.pyc	Compression load intensity in the y direction		
	p.pxy	Shear load intensity		

3. Skin Compression-Stability Schematic Model

Skin stability is evaluated with the following [1] formula:

$$\sigma_{\text{panel buckling}} = \frac{\pi^2 \overline{E} I}{L^2 A} \tag{3}$$

where

$$\bar{E} = \frac{E_{11}^{\text{skin}} b + E_{11}^{\text{stiffener}} h}{b}$$

4. Stiffener Compression-Stability Schematic Model

A formula valid for isotropic materials is adapted for composite materials and used for the calculation of the initial buckling load of a long flange:

$$N_{\text{stiffener buckling}} = \frac{2K}{(I_0/A)} \tag{4}$$

where K is the torsional stiffness ($K = 4hD_{33}$); I_0 is the polar moment of inertia about the root, the base of the stiffener ($I_0 = h^3 t/3$); and A s the flange area ($A = ht$). In the case of isotropic materials, this formula reduces to the following well-known formula:

$$\sigma_{\text{stiffener buckling}} = \frac{1}{2(1+\nu)} E \left(\frac{h}{t}\right)^2 \approx 0.385 E \left(\frac{h}{t}\right)^2 \tag{5}$$

To evaluate local stiffener buckling in compression with FE, the stiffener is modeled as a simply supported flat strip with one free edge (Fig. 8). The strip can shorten, but out-of-plane movement is constrained along the three simply supported edges.

5. Skin Shear-Stability Schematic Model

The shear buckling load of a long plate, all edges simply supported, subjected to pure shear is estimated in [9]. The plots on long plates subjected to a shear load are parameterized and included via a set of equations that describe the relation.

To evaluate skin buckling in shear with FE, a model is used that features three skin webs with four stiffeners and two ribs, illustrated in Fig. 9. The shear load is applied at all edges of the skin webs. Lateral in-plane movements are constrained in the center point of the central web, and the ribs are constrained for out-of-plane movement. The outer webs, outer stiffeners, and frames have an increased thickness (here, 1.5 times the thickness) compared with the central web and the stiffeners, to compensate for the stiffness of the left-out structure

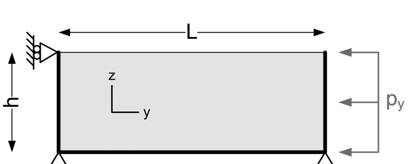


Fig. 8 FE compression model for determination of the initial stiffener buckling in compression.

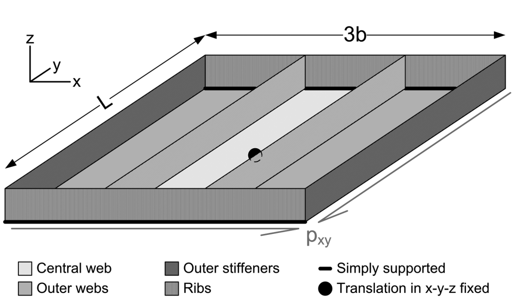


Fig. 9 FE shear model for determination of the initial skin buckling in shear.

D. Sizing Process

The sizing process is a constrained optimization for minimum weight. The used optimization algorithm is called `fmincon`, a MATLAB function from the optimization toolbox. The algorithm uses sequential quadratic programming (SQP) to find the values of the design variables that minimize an objective function subject to a set of (non)linear (in)equality constraints. The constraint evaluations are implemented in MATLAB and the finite element program UNA [8]. The design parameters are fixed during the sizing and formalize design requirements (panel-load intensities), design options (material), schematic models (analytical models), and optimization parameters (boundaries for design variables).

The panel model is defined by a set of design variables $\bar{x}$ and a set of fixed design parameters (see Table 1). The optimization can be formally described as a combination of an objective function, a set of equality constraints c_{eq} and inequality constraints c, bounds $\overline{lb}$ and $\overline{ub}$ on the design variables, and an optimization method SQP:

$$\min_{\bar{x}} f(\bar{x}) \tag{6}$$

subject to

$$c_{\text{eq}}(\bar{x}) = 0 \qquad c(\bar{x}) \leq 0 \qquad \overline{lb} < \bar{x} < \overline{ub}$$

The objective function f evaluates the weight of the panel per unit area:

$$f(x) = (W_{\text{skin}} + W_{\text{stiffener}})/bL \tag{7}$$

where $W_{\text{skin}} = \rho bLt_{\text{skin}}$ and $W_{\text{stiffener}} = \rho hLt_{\text{stiffener}}$.

Inequality constraint functions are used to evaluate the material failure criteria and the panel stability criteria. Material failures of the 0- and 45-deg material layers are evaluated with TH [see Eq. (8)]. The four stability failure modes N_{buckling} are used to validate if the applied load N_{applied} does not cause initial buckling [see Eq. (9)].

$$c_{\text{strength}} = \text{TH} - 1 \leq 0 \tag{8}$$

$$c_{\text{stability}} = \frac{N_{\text{applied}}}{N_{\text{buckling}}} - 1 \leq 0 \tag{9}$$

E. Model Verification

The results obtained with the numerical and the analytical models used for the evaluation of local stiffener buckling are compared for verification purposes. The results are shown in Fig. 10 for several values of the design variable vector [b, h, n_0, n_{45}, ns_0, ns_{45}, L]. Although the panel length is not used in the panel optimization, it is included in the verification, because individual panel lengths can differ.

The shear buckling results from the evaluation models are shown in Fig. 11 for the skins. Figure 11c shows the advantages of using FE models over analytical models. When the panel length increases, the stiffness of the stringer is not sufficient to constrain the buckling within the central web and the buckling shape will extend over multiple webs, resulting in a decreased initial buckling load. This discrete effect cannot be modeled with the analytical formula. The application of the analytical formula for stiffeners are not conservative enough, because they deviate more from the numerical calculations, which are based on a linear buckling analysis without initial deformation, which is already nonconservative.

For the current application, the results from both models correspond well enough, although the schematic models should be used with care. The stiffener modeling by analytical models deteriorates even more for decreasing panel length. The analytical schematic models for compression give a satisfying result in respect to the FE models, whereas the analytical schematic models for shear deviate from FE models, but do show a similar trend. The results

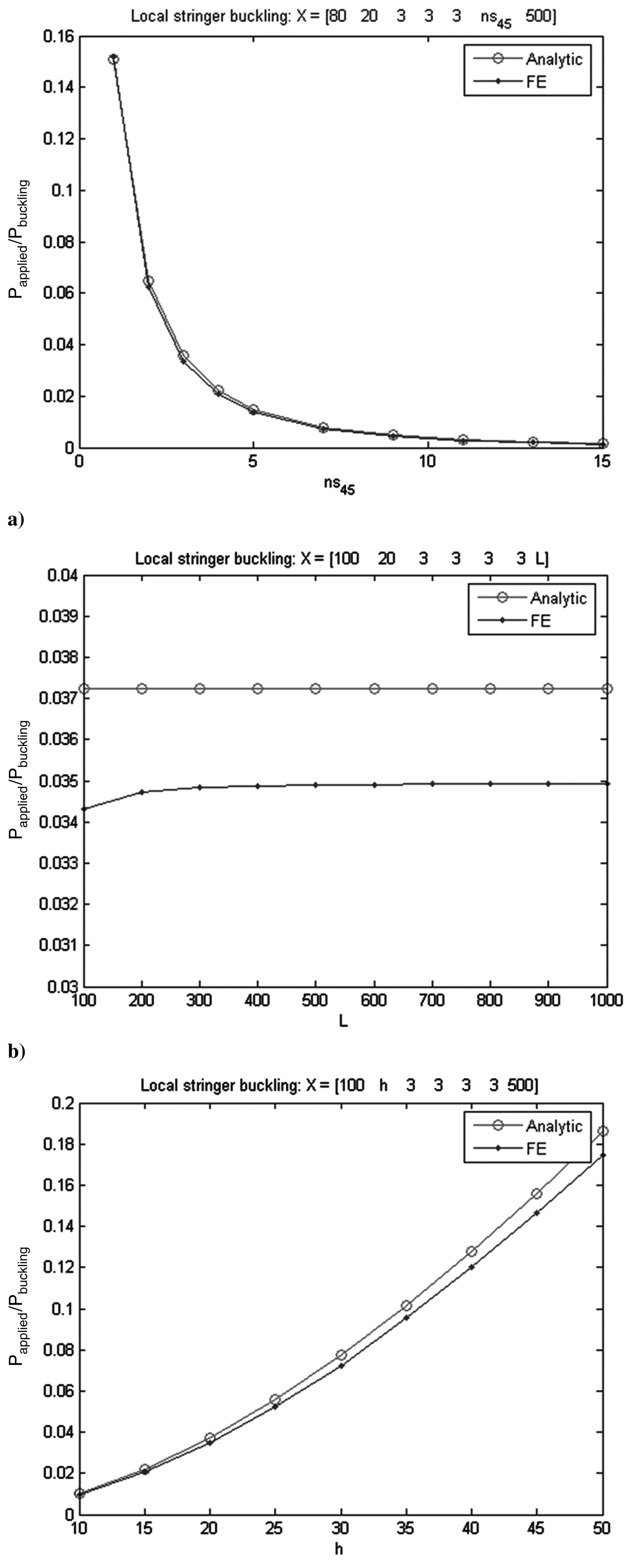


Fig. 10 Stiffener buckling in compression for different panel configurations; initial configuration changed for a) the number of ±45-deg plies in the stiffener, b) the length of the panel, and c) the height of the stiffener.

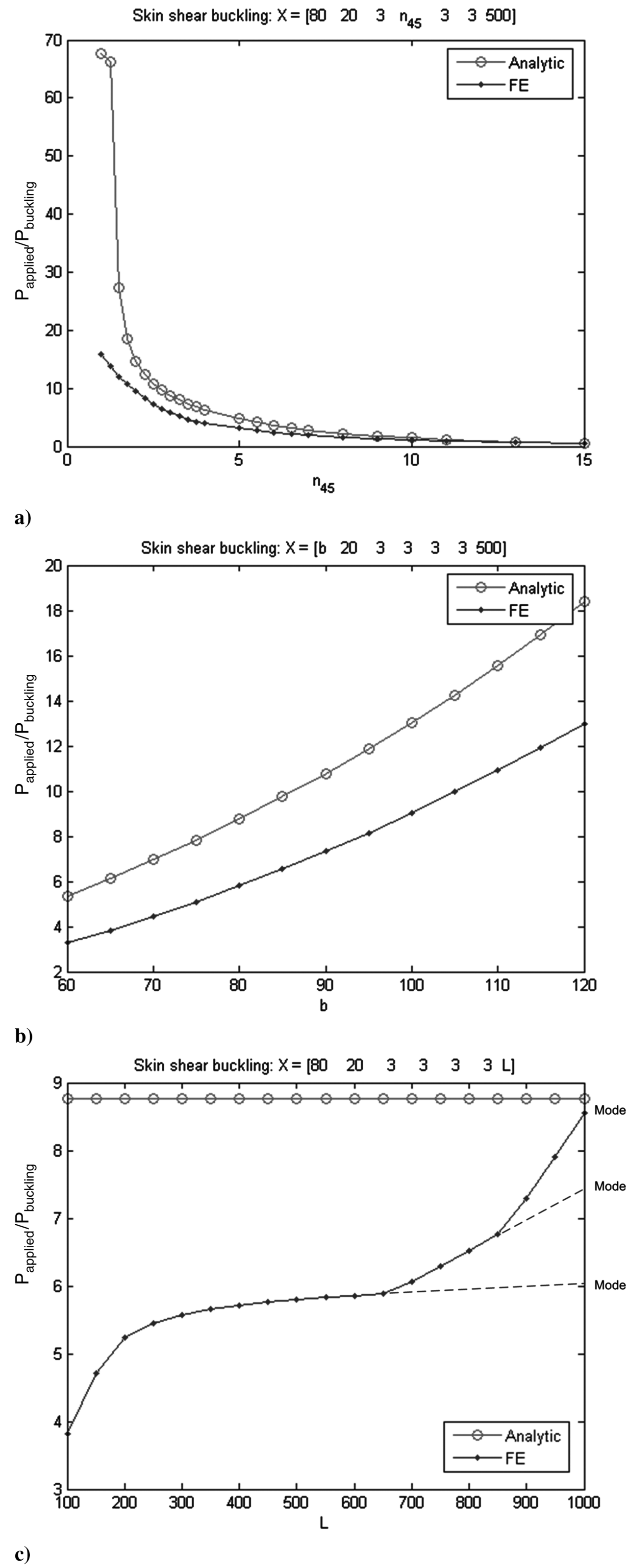
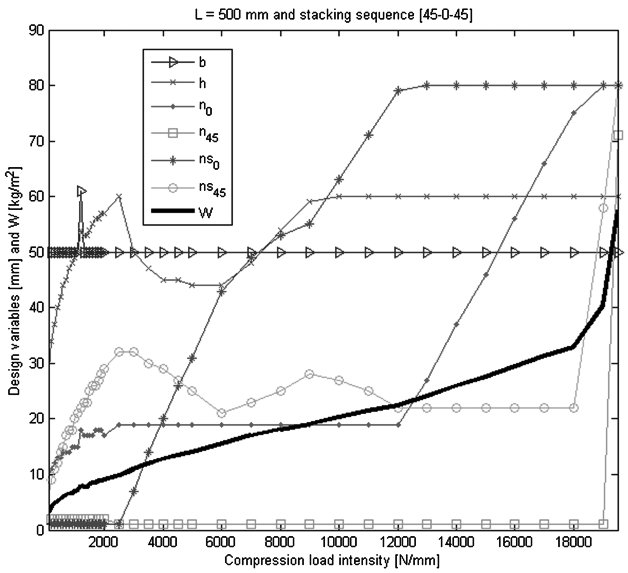


Fig. 11 Skin buckling in shear for different panel configurations; initial configuration changed for a) the number of ±45-deg plies in the stiffener, b) the length of the panel, and c) the height of the stiffener.

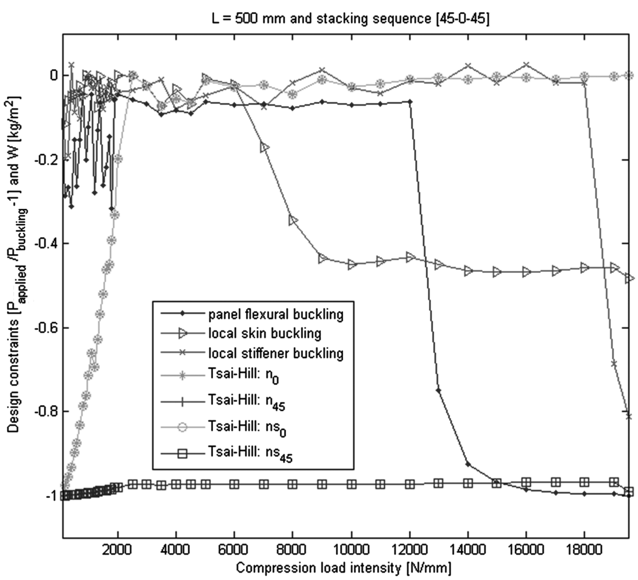
from the analytical models can be used to find the trends in panel properties to changing design requirements. The analytical shear schematic model should be improved to be able to use it for real design purposes.

F. Feasilization Process Verification

To verify the robustness of the process, a test panel is sized (i.e., optimized). Increasing compression and shear loads are put on the panel and the results are illustrated in Figs. 12 and 13, respectively. The design variable vector is $[b, h, n_0, n_{45}, ns_0, ns_{45},]$, the panel length is fixed at 500 mm, and the stacking sequence is set to [±45, 0,

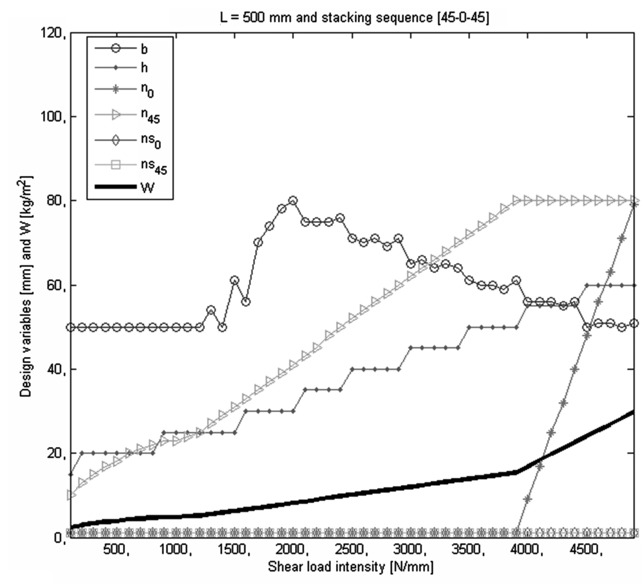


a)

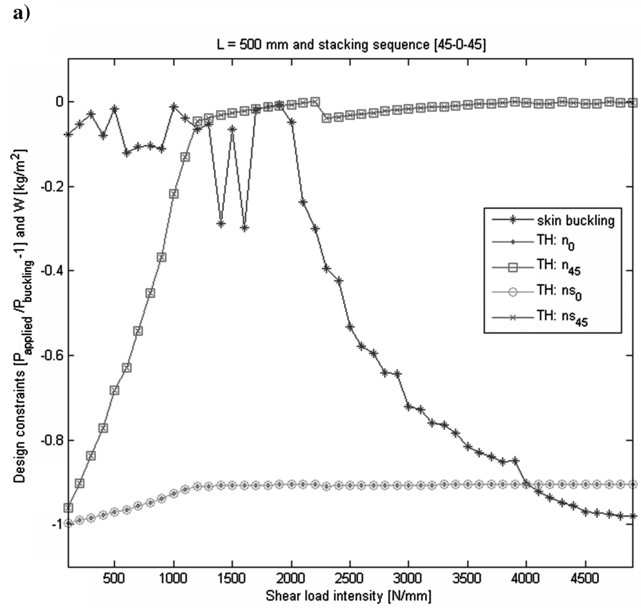


b)

Fig. 12 Panel designed for pure compression, for a range of load magnitudes: a) in the optimal design variable values and b) in the design constraint values are presented as a function of the load intensity.



a)



b)

Fig. 13 Panel designed for pure shear, for a range of load magnitudes. In the a) the optimal design variable values and in b) the design constraint values are presented as a function of the load intensity.

± 45 deg] for both the skin and stiffener. The process gives appropriate results over a large range of loads. The panel weight as a function of the load intensity is the best indicator for the validity of the results.

VI. Vertical-Tail-Plane Feasilization

The feasilization process is implemented for a specially developed DEE [10], which is based on the DEE concept introduced in Sec. II. The ACPAM [11], which is a dedicated model generator for the Airbus product family, is used as a MMG with the functionality to represent a complete family of aircraft and to produce different views on the aircraft to deliver to the analysis tools. Maneuver loads are predicted by a loads analysis tool developed by Airbus, called VarLoads [10,12]. The loads analysis tool is based on an integrated aircraft model (i.e., all flight loads can be calculated using a fully flexible aircraft model).

As a case study [10], the vertical tail of a long-range passenger aircraft is redesigned. The remaining components are represented by a validated baseline model. At first, the attempt is to reproduce the original configuration. A DEE representing the specific process is illustrated in Fig. 14.

A. Feasilization Process

The input file for ACPAM is prepared according to the baseline aircraft measures. From the MMG, the vertical-tail-plane (VTP) models are requested and exported into the PATRAN/NASTRAN environment. A Guyan reduction of the tail structure is performed in NASTRAN to obtain the reduced stiffness. The complete mass matrix of the vertical tail is then used to calculate the lumped masses. Afterward, the reduced structural and mass models of the fin are assembled with the reduced model of the rest of the aircraft, and another NASTRAN modal analysis is performed to obtain the modes of the complete structure. The results of this calculation, together with the aerodynamic 2D panels, are required as input for VarLoads. In VarLoads, a yawing maneuver is performed, and the nodal loads are determined.

The panel initiator reconstructs the vertical tail model and determines the panel-load intensities based on the nodal loads (see Sec. VI.B). The panel initiator provides a first estimation on the panel

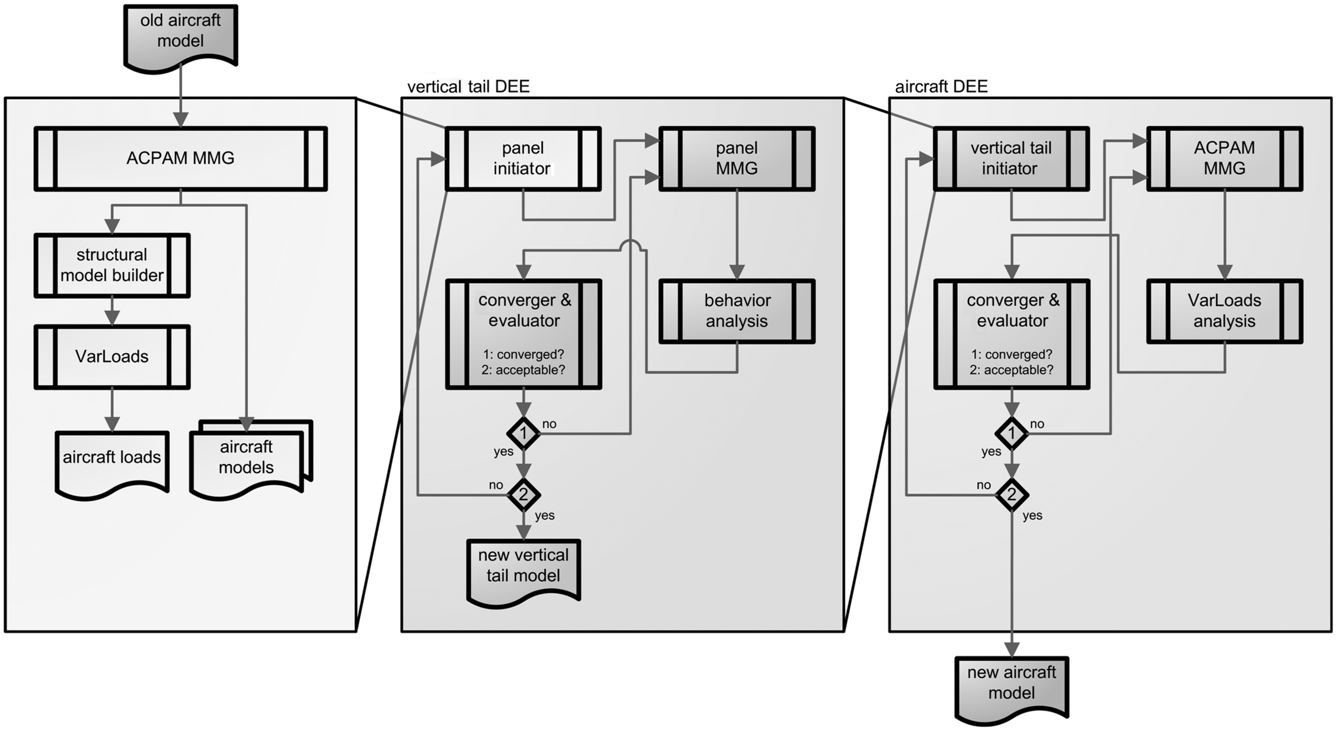


Fig. 14 DEE process for optimization and redesign of a tail.

dimensions, based on which the panel MMG generates a set of skin, spar, and rib panel models. The behavior of these models is then determined and evaluated. If the panel solution is not feasible and optimal, the panel initiator selects a new set of dimensions and feeds them to the panel MMG. This optimization process is continued until a set of feasible optimal solutions is obtained. In the first step, the skin and spar panels are sized (e.g., optimized). Second, the deformation of the tail and the resulting deformation load intensities are determined based on the sized panels, and the rib panels are sized. Finally, the panels are translated to an equivalent panel (see Sec. VI.C).

Based on the updated panels, the FE model previously generated by the ACPAM MMG is updated. The yawing maneuver is performed in VarLoads and a new load set is updated. For simplicity, the evaluator checks the differences in the tail weight, not the nodal loads, and therefore the vertical tail initiator is reexecuted and an updated tail weight is obtained based on the new nodal load set. Finally, the evaluator checks the differences in weight and stops if the design meets the requirements. All the steps are performed automatically with the help of a DEE framework [4].

B. Load Intensity Identification

VarLoads uses a structural model built up from beams and condensed masses to analyze the aircraft loads, taking into account the flexibility of the structure. The static load cases, used to size the tail, are selected for correlated loads for maximum N_y, M_x, and M_z at the root. The loads on the masses are input to the sizing process. These loads are translated into internal loads at the required locations. For skin and spar panels, the geometric center of the slice is taken as the internal load point, and for rib panels, the panel geometric center is taken. The translation from external nonintegrated loads to internal loads at a specific location is threefold. First, the external loads are integrated along the condensation line to obtain the internal load in the structure at each condensation point. Second, the resulting loads are transposed to the internal load point in the local axes system, illustrated in Fig. 15.

Finally, the internal loads are translated to load intensities, using some basic assumptions on the load paths in the structure (Fig. 16). The torsion moment M_y is introduced to both spars and skin. The force N_y is introduced in the skin and spars; combined with the moment M_x, it determines the level of tension and compression in the upper and lower skins, and combined with the moment M_z, it determines the level of tension and compression in the spars. N_x together with M_y determine the level of shear in the skins, and N_z together with M_y determine the level of shear in the spars. The rib is subjected to a crushing load due to wing deformation and to shear due to M_y. The skin, spar, and rib panels' load intensities are determined, respectively, using the following formulas:

Skin:

$$p_x = 0 \qquad p_y = \frac{F_y}{2\cdot(\text{width}+\text{height})} - \frac{M_x}{\text{width}\cdot\text{height}} \qquad p_{xy} = \frac{M_y}{2\cdot\text{width}\cdot\text{height}} + \frac{F_x}{2\cdot\text{width}} \tag{10}$$

Spar:

$$p_y = \frac{-M_z}{\text{width}\cdot\text{height}} + \frac{F_y}{2\cdot(\text{height}+\text{width})} \qquad p_z = 0 \qquad p_{yz} = \frac{M_y}{2\cdot\text{width}\cdot\text{height}} + \frac{F_z}{2\cdot\text{height}} \tag{11}$$

Rib:

$$p_x = 0 \qquad p_z = p_{\text{crush}} \qquad p_{xz} = \frac{M_y}{2\cdot\text{width}\cdot\text{height}} \tag{12}$$

Except for the crushing load, all the load intensities are calculated before the actual sizing. The crushing load is determined during sizing with Eq. (13) from Rothwell [7] (see Fig. 17) because it depends on the skin panel's properties:

$$p_{\text{crush}} = (\bar{p}_{\text{upper skin}} + \bar{p}_{\text{lower skin}})\cdot\sin\left(\frac{\alpha}{2}\right) \tag{13}$$

where

$$\alpha = \frac{\delta_{\text{upper skin}} + \delta_{\text{lower skin}}}{\text{height}} \qquad \delta = \frac{p_1 - \nu_{12}p_2}{tE_{11}}\cdot\text{length}$$

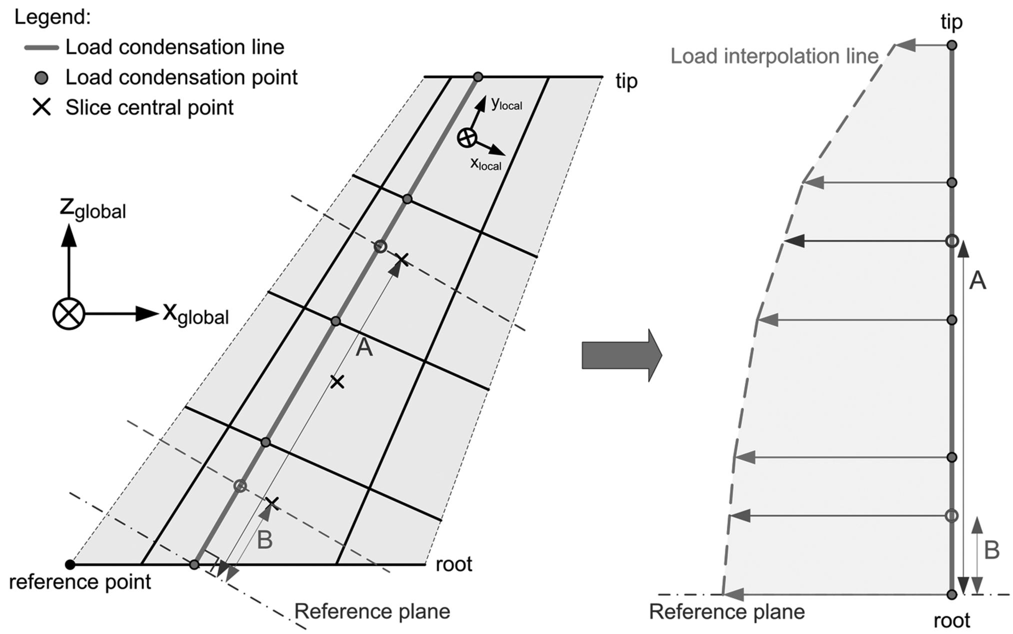


Fig. 15 Determination of internal loads at the central point of each VTP slice.

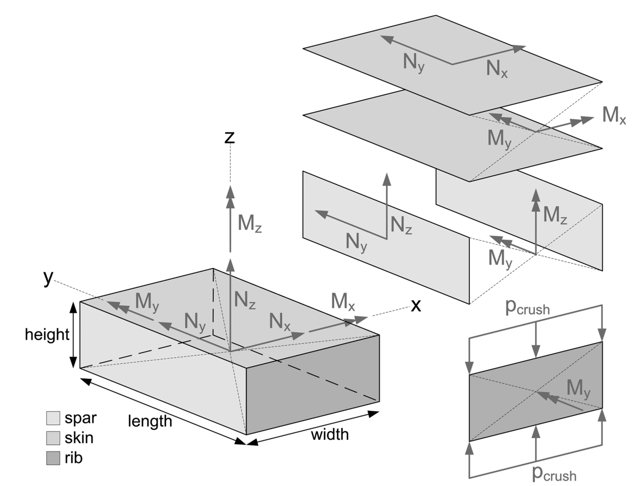


Fig. 16 Internal loads to load intensities in the panel schematic model.

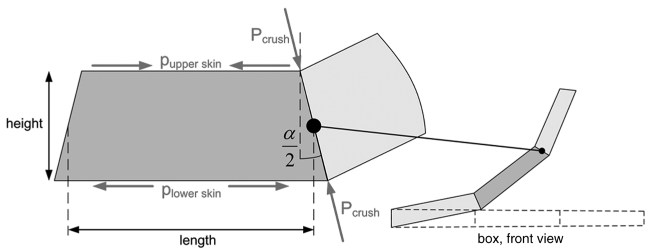


Fig. 17 Crushing load on a rib due to bending.

C. Equivalent Sandwich-Panel Model

The FE model used by PATRAN (generated by the ACPAM MMG) does not include stiffeners as separate elements, but uses a sandwich panel as the LLP. To update the original PATRAN model, the panel feasilization results are translated from a blade-stiffened panel to an equivalent sandwich panel. The equivalence is based on identical extensional stiffness. The facings are calculated for equivalent engineering constants E_{11}, E_{22}, G_{12}, and ν_{12}, with a thickness equal to half of the cross-sectional area of the blade-stiffened panel. The core is a fake material, with such a thickness that the blade-stiffened panel and the equivalent sandwich panel have an identical second moment of area in the cross section.

D. Results

The panel sizing results are presented in Fig. 18. The tools are run on a mixed architecture (Linux and Windows) with an average processor power equal to a Pentium IV 2.5 GHz. In Table 2, the

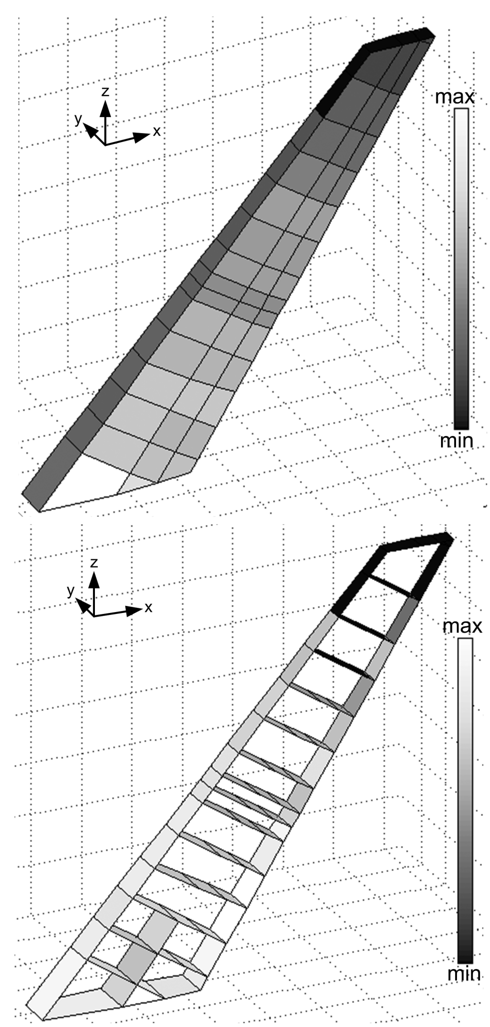


Fig. 18 Relative weight per area of the tail panels.

Table 2 Overall average calculation times; averaged and based on mixed computers with the average speed of an Intel Pentium IV 2.5-GHz processor or comparable

Process	In loop	Calculation time per configuration
ACPAM–multimodel generator	No[a]	120 s
PATRAN–mesher	Yes	180 s
NASTRAN–stiffness condensation	Yes	30 s
MATLAB–mass condensation	Yes	180 s
NASTRAN–assembly	Yes	60 s
VarLoads	Yes	420 s
Sizing tool	Yes	1620 s
Framework–overhead	NA	240 s
Total time		∼48- min first loop, ∼45- min consecutive loops

[a]ACPAM is only executed the first run of the DEE and is not included in the loop.

calculation time for each process step is depicted and the total calculation time per configuration is calculated. On average, panel feasilization requires 10 s. The complete process can be carried out automatically. One process loop requires less then 45 min, based on a tail model consisting of 171 panels. The first loop requires an extra 2 min for the instantiation of the initial aircraft model. After this loop is completed, each new loop does not need to have the ACPAM MMG geometric models created, because they are already available in the output buffer.

As validation of the process, a converger checks the differences between each loop and terminates new loops when the change in weight is less than a criterion of 1%. To reach the criterion, an average of six loops are run before the converger finds convergence and terminates the loop.

VII. Conclusions

The initiator component, part of the design-and-engineering-engine (DEE) concept, was elaborated. From the elaboration, it became clear that the initiator can consist of multiple DEEs addressing a simplified version of the real design problem to find an initial set of parameter and variable values that describe a feasible solution. The initiator concept is implemented using feasilization, which makes use of a simplified design problem by using a reduced set of the requirements, a simplification of the design options, and schematic models. The methodology is implemented for an example redesigned vertical-tail-plane case of a general passenger aircraft.

A simplification of the design options was obtained by using problem decomposition; translating the high-level primitive VTP design problem into a set of lower-level primitive skin, spar, and rib panel structures; and reducing the size of the design problem significantly. a complication with decomposition is that it assumes that the sum of partial solutions forms a feasible overall solution; this is taken into account by incorporating checks on failure modes associated with the combination of partial solutions of the LLPs.

The schematic models proved useful in describing the behavior of the panels. Using analytical models, a quick and proper estimation of the behavior could be obtained, due to their highly simplified character relative to the more resource-intensive finite element models. Although the results showed that the schematic models should be used with care, because all the models were constrained to a certain design space and one of the models deviated too much from the behavior calculated with the FE models to be used in an actual design case.

The feasilization process is automated using optimization methods. However, the implementation was done for a single design option; an automated concept selection and tradeoff process is not yet implemented. The automation of the sizing process is a first step in automation of the conceptual design process, relieving the designer from the otherwise manual sizing process. Future development steps will focus on automation of the concept selection and tradeoff processes.

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