A NEW DASSAULT INDUSTRIAL APPROACH FOR AERO-STRUCTURAL OPTIMIZATION OF COMPOSITE STRUCTURES WITH STACKING TABLE CONSTRAINTS

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Abstract: During the initial design phase the evaluation of the aircraft's structural mass for various plan forms is still a challenge, even more so when using composite materials. The structural mass results from all the structural parts which insure both the structural integrity as well as the static and dynamic stability of the aircraft. In order to increase aircraft performance this mass should be minimized.

The aero-structural optimization performs this minimization while respecting the constraints on the structure: sustain loads and satisfy various aeroelastic constraints. These constraints could be avoiding flutter phenomena in the flight envelope or guarantying static aeroelastic characteristics such as aileron efficiency. With a composite structure numerous technological constraints are also required to ensure the final structure will respect state of the art drawing rules and can be easily manufactured. One important technological constraint for a composite structure is to use a single stacking table. These new constraints introduce many new complexities in the optimization process.

This paper describes an industrial method developed at Dassault Aviation to perform this aero-structural optimization on composite structures. This fully automatic process can compute the optimized structural weights of several configurations. The resulting structure uses a unique stacking table that is optimized by the process, thereby ensuring the structural mass is minimal.

Combining this with aerodynamic performance data for each configuration, it becomes a powerful tool to drive the design of new aircraft.

First the global optimization process will be briefly presented. Then the method used for optimizing a composite structure with a stacking constraint will be described. Finally an application of this process on a generic Falcon jet with a composite wing will be shown.

INTRODUCTION

In order to draw an optimal plan form and external shape of a new aircraft, it is essential to be able to evaluate performances with respect to various geometrical parameters. For example aerodynamic performance is routinely integrated in the preliminary design phase and can be evaluated rapidly for many different shapes. Another crucial performance to evaluate is the structural mass of the aircraft.

In [1] a methodology is described to perform this mass evaluation for a metallic structure. The aero-structural optimization process relies on a number of modular and parameterized blocks. Each block performs different tasks such as the creation of the reduced order models, aerodynamic load computations, aeroelastic computations or structural sizing. Using a series of these blocks the aero-structural optimization iterations are done automatically. An application on a generic Falcon jet with a metallic wing has demonstrated the relevance of this process when studying various aircraft configurations.

This paper presents the recent developments to modify the existing process to cope with composite structures. When using composite structures many new challenges arise, the most constraining for aero-structural optimization being the new technological constraints needed to respect numerous state-of-the-art drawing rules which enable the easy manufacture of the structure. The discrete nature of composite layups also adds new complexities to the process.

With composite structures new variables are available to put material not only where it is needed but in the direction it is needed. However these new variables come with new constraints to ensure the structure is viable. For example, to ensure the structure can be manufactured, each separate panel must use a single stacking table. This is a global constraint which must be applied to the whole structure.

The classical method for designing with composite materials is to perform an initial optimization and then choose, with engineering judgment, a unique stacking table by analysing the result of the initial optimization. In addition to being time consuming, this deteriorates the optimized solution and as a result the final structural mass is heavier than the optimized one. Therefore a new method which doesn’t rely on engineering judgement is developed. This method is better suited for automation and returns a more optimal solution.

Based on the initial process for metallic structures, a new global optimization process for composite structure is developed. With this new method the stacking table constraint is introduced directly into the optimization of the structure. This gives a better solution and eliminates any sub-optimal adjustments of the solution. The global optimization is used to perform an entirely coupled sizing and aeroelastic optimization.

In this paper, first the global optimization process will be presented and the main differences between the metallic and composite case will be exposed. Then the structural sizing and aeroelastic optimization for a composite structure will be fully described. Finally an application of this process on a generic Falcon jet with a composite wing will be presented. Lastly the conclusions of this study will be drawn.

1 GENERAL OPTIMIZATION PROCESS

The general process is based on the method in [1]. The objective is to minimize the structural weight of the aircraft while guarantying its structural integrity and satisfying the various aeroelastic constraints such as minimal flutter speeds.

Figure 1 shows the optimization process. The global layout is identical to the process for metallic structures. Using parametrized geometry the structural finite element model (FEM) is built and the aerodynamic computations are performed. Then reduced models for load computations and aeroelastic analysis are created. These models are used to compute sizing loads and aeroelastic constraints. Using a specific optimization method the optimal properties found respect both the aeroelastic constraints and the sizing criteria. These new properties are used to generate a new FEM. The previous steps are then repeated until the process converges.

Most of these blocks are done with the in-house computational tools Elfini©. This tool is already capable of analysing composite structures, including dynamic and static computations, model reductions, flutter derivative estimation, etc. The mass evaluation is also modified to accurately predict the mass of the manufactured composite structure.

The main difference between the metallic and composite processes however lies in the strategy for a coupled aeroelastic optimization and structural sizing (red block in Figure 1). This block is based on a completely new method for aero-structural optimization which is detailed in the following paragraph.

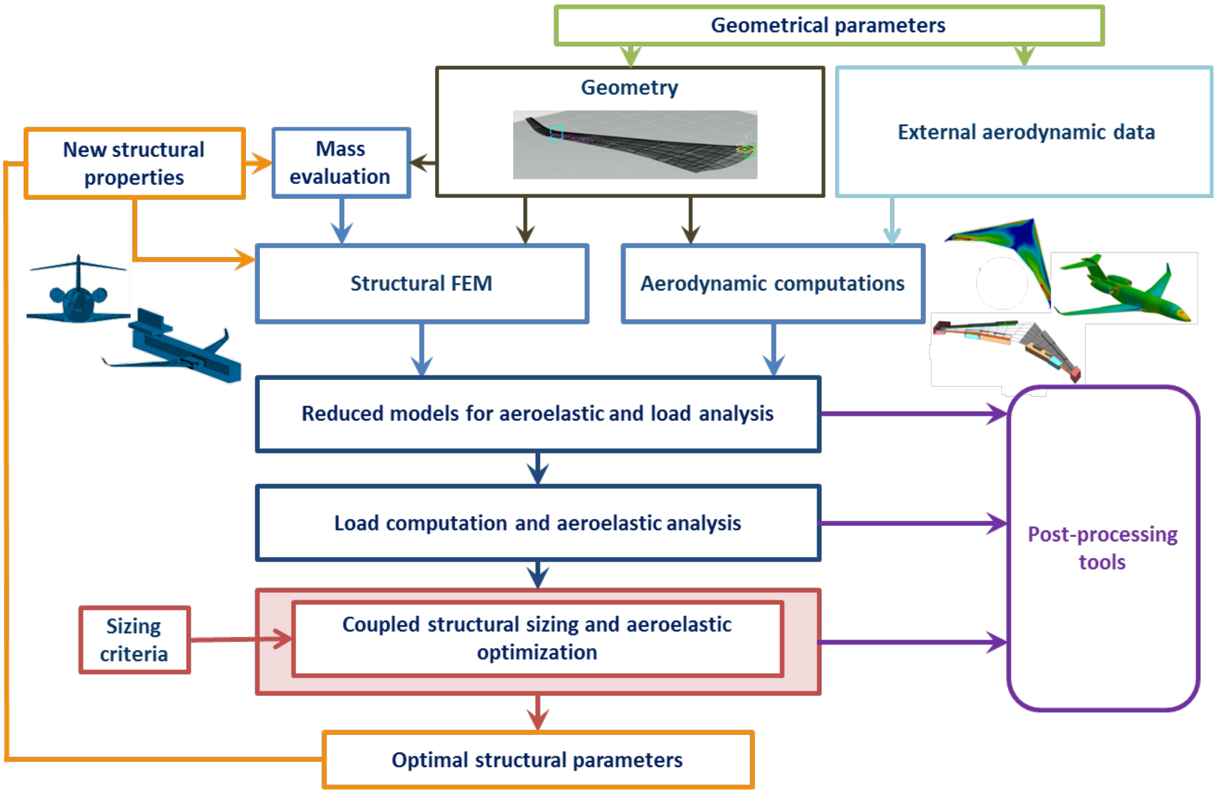

Figure 1 : General optimization process

2 COMPOSITE SIZING AND AEROELASTIC OPTIMIZATION

As for metallic structures, the composite structure is divided into “super-stiffeners” (or cells). Each cell is made up of a stiffener and the skin on each side. Figure 2 shows a chord-wise cut of a cell. On each of these cells the sizing loads are computed. Then the optimal properties are calculated while respecting numerous sizing criteria.

Figure 2 : Super-stiffener

The use of global constraints (flutter, technological …) implies the use of a global optimization, as opposed to a cell by cell method. Since a composite structure is made of a sequence of layers, the optimization must return the number of layers in each direction. This number must be discrete since the composite layers are undividable: it is not possible to add a fraction of layer.

2.1 Technological constraints for composite structures

In a composite structure two plies with different directions must never come end to end. This constraint is respected by imposing a unique stacking table on each panel of the structure. This table gives the complete layup of a cell as a function of that cell’s thickness. An example is given Figure 3. In this example the plies are set in three different directions (typically 0°, +/- 45° and 90°).

In the stacking table the plies for a given thickness must always be included in the layup for a superior thickness. As a result, starting at the maximal thickness, only one ply is removed (and none can be added) from a current thickness to the next inferior thickness in the sequence. The stacking table also defines the order in which the plies are stacked when manufacturing the structure.

For aero-structural optimization the stacking order has very little influence on the material properties obtained and therefore on various constraints used. So, instead of the complete table a simpler representation is used: the degression law. It gives only the number of plies in each direction but not the order in which they are stacked. The degression law matching the previous stacking table is given Figure 4.

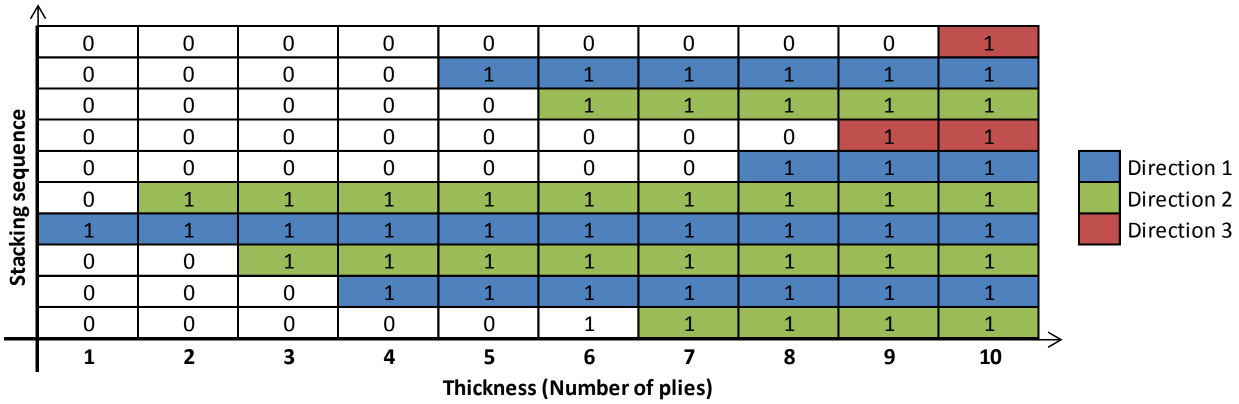

Figure 3 : Example of a stacking table

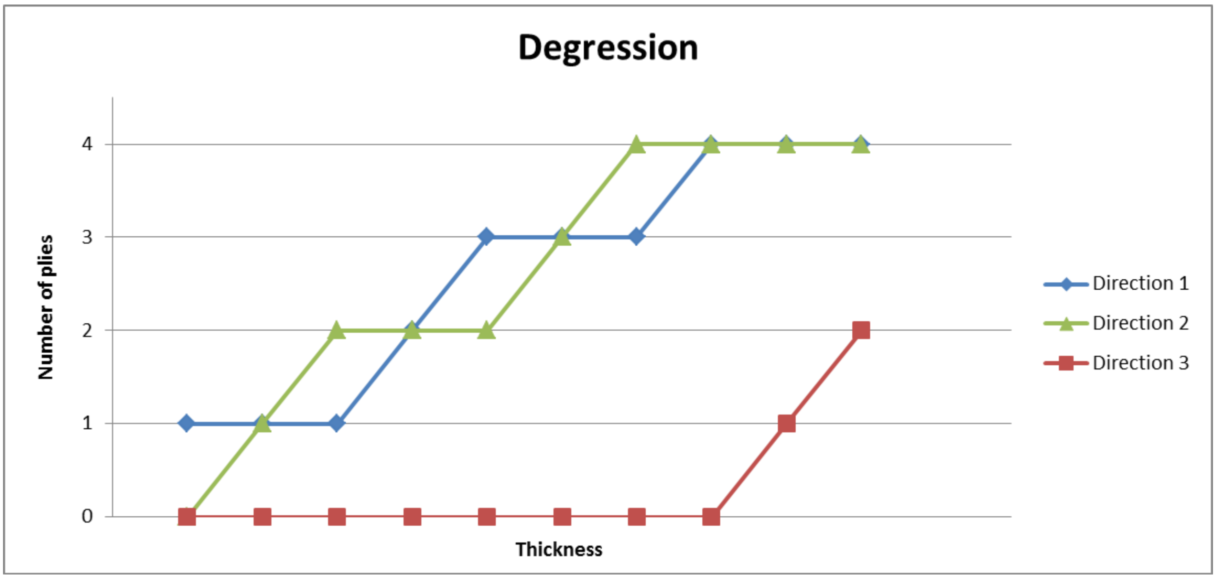

Figure 4 : Example of a degression law

It is possible to create a valid stacking table from a degression law as long as each curve of the degression law is an increasing function of the thickness.

In the following method the applied constraint will be to use a single degression law for each separate panel of the structure. A minimum and maximum percentage of plies in each direction should also be imposed on the degression law.

2.2 Composite sizing

The introduction of new global technological constraints for composite structures imposes the use of a global optimization approach. A new global optimization problem taking into account each composite panel as a whole is defined. Its cost function is still the structural mass. The method is detailed below.

2.2.1 Constraints

Various constraints are applied on each cell of the composite structure; there are both sizing criteria such as buckling, compression after impact, maximal tension / compression near holes, etc. and technological constraints for example geometrical shape factors, minimal thickness, etc. These constraints are defined for each cell and depend only on the properties of that cell. In addition, the global constraint to respect a degression law (and therefore a unique stacking table) is applied.

2.2.2 Variables

The possible fibre directions in the skin are chosen, so the basic variables which define the skin are the number of plies in each direction. When taking into account the degression law only one variable is needed for the skin since the number of plies in each direction is a result of the total thickness of the cell. The percentages of plies in each direction of the stiffener are fixed, so the stiffener is defined by its total number of plies and height. That leaves 3 variables for each cell of the structure: skin and stiffener thickness and stiffener height.

The degression law used is not a priori chosen but optimized during the process. The maximal number of plies in each direction must first be chosen. Then, since for each inferior thickness only one ply is removed, the degression law is defined by the order in which the plies are removed. By directly optimizing the degression a better optimum can be found than if the degression was chosen after the optimization. This also greatly reduces the work and additional mass needed to modify the optimized result so the structure can be manufactured.

2.2.3 Sizing method

The sizing is done in 4 steps. The first step is to choose the maximal number of plies in each direction for the degression law. This is done by performing a cell by cell optimisation of the structure without any global constraints. The maximal number of plies used for the degression law is directly the maximum number of plies of the cell by cell solution plus a margin.

The next step is a global optimization of the structure with a degression law constraint. This optimization is done in Matlab© using the so-called “Interior Point algorithm” (cf. [5]). In order to find the global minimum this optimization is done with multiple starting points. For each point a random initial degression law is generated. Most of these optimizations converge toward the same minimal solution which is therefore considered to be the optimal solution. The result is continuous (as opposed to discrete) and gives the optimal degression law and the properties of each cell.

Then a discrete degression law is created based on the best result from the previous optimizations. The continuous optimized degression curve is approached by a discrete curve. This is taken as the optimal degression law for the structure.

The final step is a hybrid cell by cell optimization of the structure using the optimal degression law. The number of plies must be discrete; however the stiffener’s height is a continuous variable. To find the optimal solution the height is optimized for all the combinations of skin and stiffener thickness. This method guaranties that the optimal solution is found where a simple rounding of the number of plies given by the continuous optimization doesn’t even guaranty a feasible solution.

2.3 Aeroelastic optimization model

The aeroelastic constraints are formulated with linearized models. Using the state of the art computational tools provided in Elfini©, the variation of the constraints $Const$ with respect to the variables x is provided as the $n_{Const} \times n_x$ matrix:

$$\frac{\partial Const}{\partial x} \tag{1}$$

Using these linearized models, the constraint to satisfy is:

$$Const_0 + \frac{\partial Const}{\partial x} x \leq Cont_{Obj} \tag{2}$$

With the new global sizing method these aeroelastic constraints can be added directly in the structural sizing, making it a truly coupled optimization.

2.4 Aero-structural optimization strategy

At each iteration of the global optimization process two different optimizations can be performed. The first is the structural sizing without aeroelastic constraints. This is done using multiple random initial points in order to find the global optimum. The second is the coupled optimization with structural sizing and aeroelastic constraints. Since it uses a linearized model it must be restricted to a region around the initial linearization point. Under this constraint multiple initial points cannot be used.

In order to find the best solution, structural sizing is first performed without aeroelastic constraints, enabling the use of multiple random starting points. Several global iterations are done until the solution is converged. Then the aeroelastic constraints are slowly added. By adding them slowly the solution is modified to satisfy the constraints while staying in the validity region of the linearized constraints.

3 APPLICATION

The goal of this application is to study the effect of various winglet shapes on the structural mass of an aircraft with a composite wing. The winglet shape has a high impact on the wings mass. It can modify the aerodynamic load distribution on the wing which then affects sizing loads. The winglet can also greatly impact aeroelastic phenomena by, for example, modifying the wing torsion mode's frequency, shape and aerodynamics.

Taking into account the aerodynamic performance of each winglet shape it is possible to determine the optimal winglet for the aircraft. The result with the first winglet is presented here. More results are presented at the 2017 IFASD conference.

3.1 Model

A preliminary model of a generic Falcon jet with a composite wing is used in this study. The finite element model is given Figure 4. Since it is a preliminary model the fuselage is not modelled. A "dummy" rigid structure is created as a mass support and to apply the aerodynamic effects of the horizontal tail and the elevators. The stiffness of the fuselage is adjusted so the fuselage modes do not interact with the wing modes.

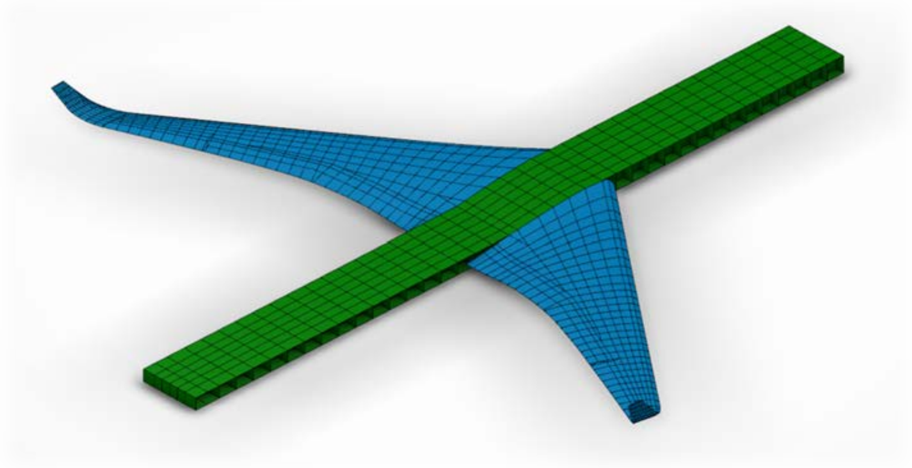

Figure 5 : Finite element model used

The variables in this study are the wing panel properties. The initial properties are arbitrary but chosen such that the constraints are not met, the initial panels are much to light. There are 222 super-stiffeners on each wing. The model is assumed to be symmetrical. On these panels the composite layers can be placed following three different directions. Each panel (upper skin and lower skin) must respect a unique degression law (one for each panel).

3.2 Sizing manoeuvres

To certify an aircraft a huge number of load and turbulence cases must be considered. In order to reduce the computational time of the aero-structural optimization process, a subset of two manoeuvres were chosen. These manoeuvres are sufficient to approach the sizing load of the structure with the required accuracy to precisely asses the structural mass.

To greatly improve the accuracy of these load computations, aerodynamic Reynolds-averaged Navier-Stokes (RANS) computations are used. For this application two effects have been used: the initial field around the aircraft (“zero effect”) and the effect due to the angle of attack.

3.3 Aeroelastic constraints

For this application a minimal flutter speed is imposed at the critical altitude. Two different mass cases are studied, one empty fuel weight and one full fuel weight. The unsteady aerodynamic computations used are based on linear Navier-Stokes results (cf. [6]).

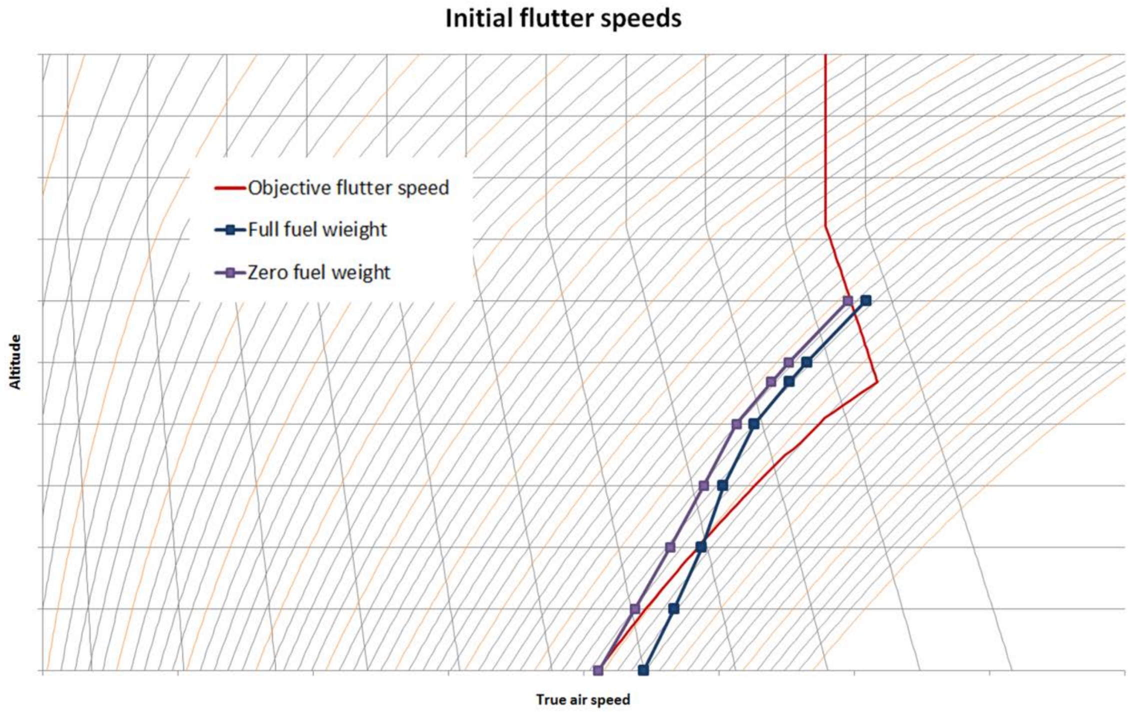

Figure 6 : Initial and objective flutter speeds

On Figure 6 is shown the flight domain with the initial flutter speeds and the objective.

3.4 Results

The optimization was done in 17 iterations. The first 10 iterations were with sizing and technological constraint only. No aeroelastic constraints were applied. 10 random initial points were used. The following 4 iterations were needed to add the minimal flutter speed constraint. Then 3 more additional iterations were done to reach the convergence. Figure 7 shows the evolution of the mass during the 17 iterations. After the 10 first sizing iterations the structural mass is almost stabilized. After the structural sizing iterations the mass gradually increases to change the minimal flutter speed. Iteration 14 shows an overshoot, the minimal flutter speed after the iteration is higher than required. This is automatically corrected by the algorithm during the following iteration. After iteration 15 the process is converged, as demonstrated by the next two iterations. The final mass variation is less than 0.05%. The final solution respects all the imposed constraints (sizing criteria, minimal flutter speed and unique stacking sequence).

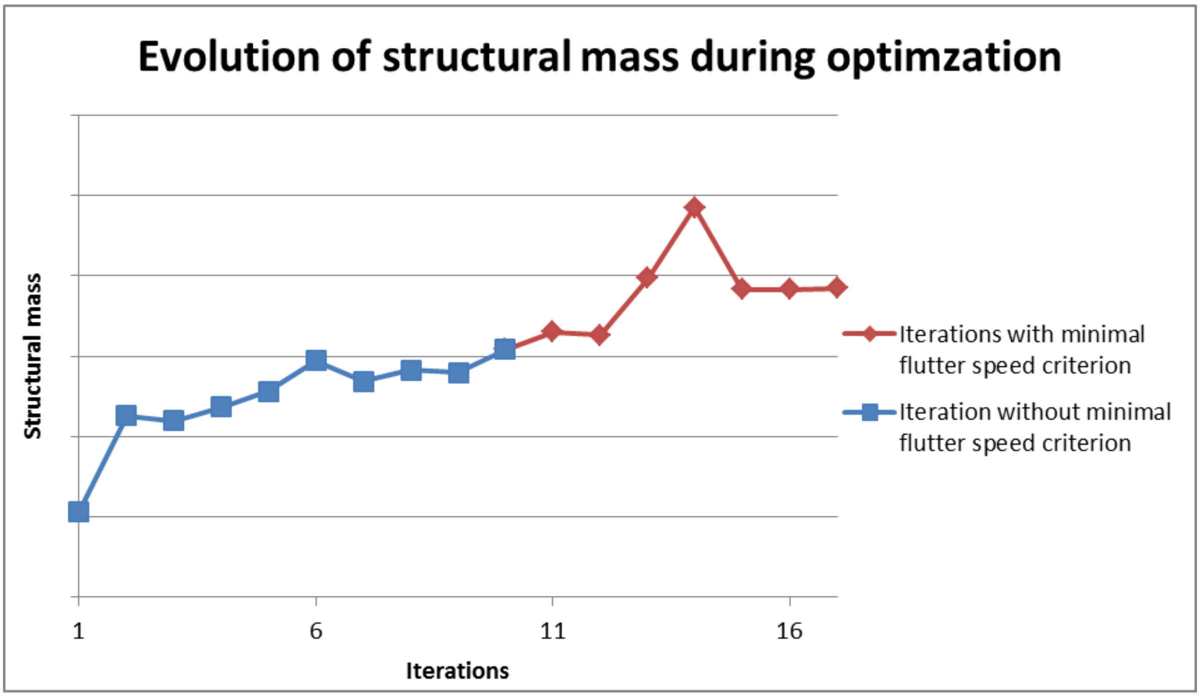

Figure 7 : Evolution of structural mass during the 17 iterations of the aero-structural optimization

Figure 8 illustrates the variation on the structural mass as a function of the minimal flutter speed during the aero-structural optimization. The 10 initial sizing iterations (blue points) have little influence on the flutter speeds. Once the aeroelastic constraints are introduced from iterations 11 (red points) the minimal flutter speed increases rapidly.

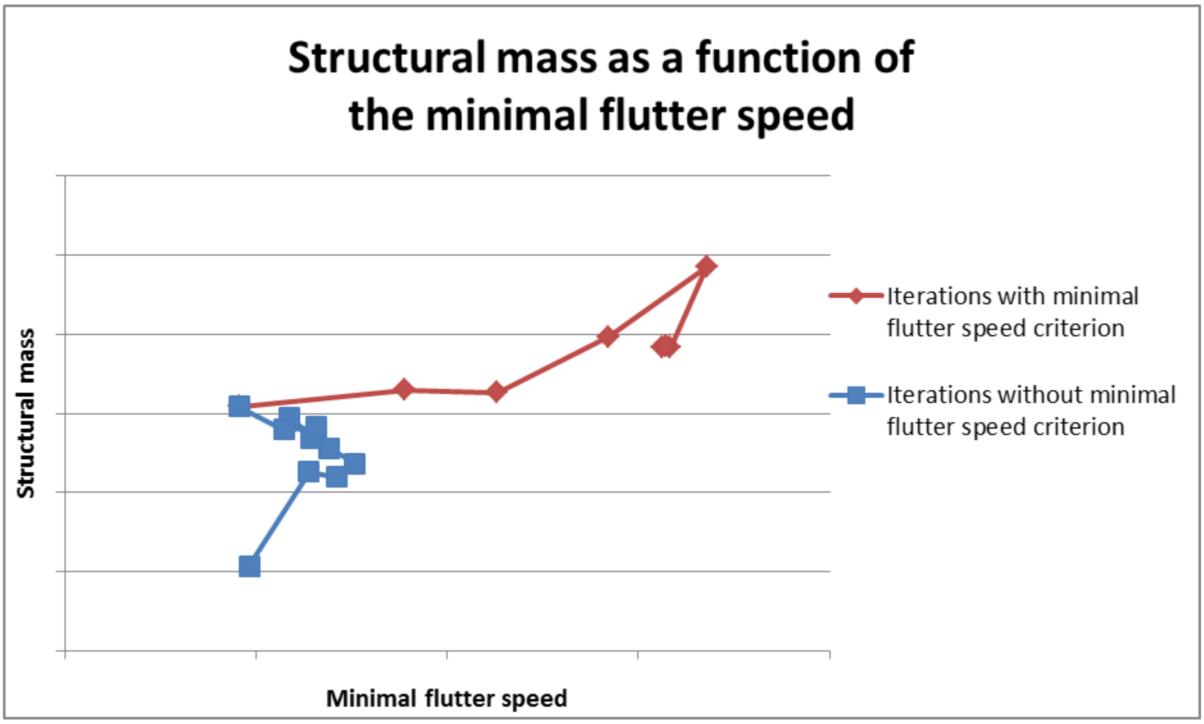

Figure 8 : Minimal flutter speed as a function of the structural mass during the optimization process

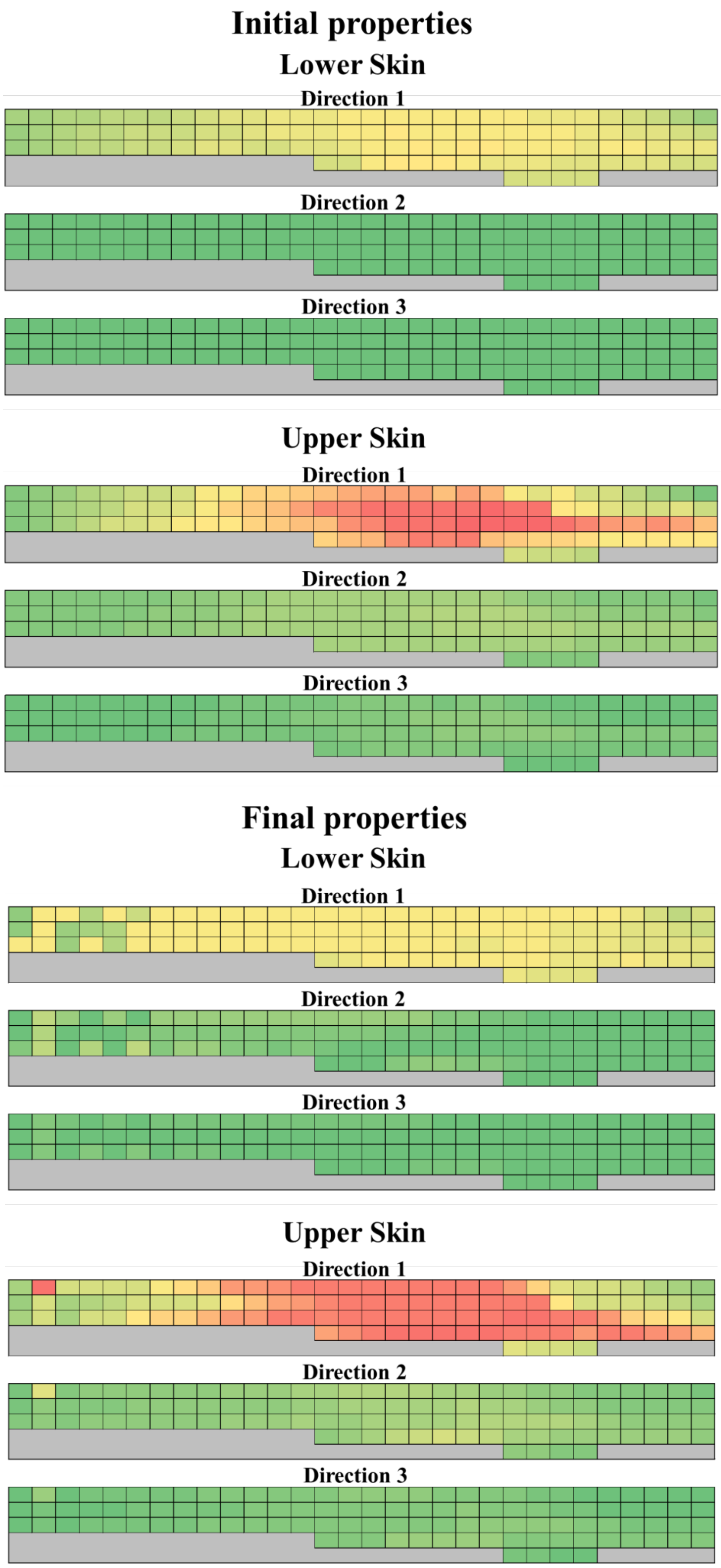

Figure 9 : Initial and final wing panel properties for each composite fibre direction

The initial and final properties are presented in Figure 9. The number of plies in each direction for the lower and upper skin is plotted.

The execution time of the process is about 3 hours per iteration (iteration as shown Figure 1 including FEM update, load calculation and aeroelastic analysis and property optimization). That is compatible with rapid sizing loops for preliminary configuration studies.

4 CONCLUSION

New methods for aero-structural optimization of composite structures have been developed at Dassault Aviation. Like the initial process in [1], this new method is based on the state-of-the-art computational tools developed in-house in Elfini© and Catia V6© as well as the powerful optimization tools in Matlab©. The mastery of all the tools involved has allowed us to create a fully automatic and parameterized procedure which can be applied to study many different variations during the development of a new aircraft.

By taking into account numerous technological constraints, the optimized composite structures are realistic and can be easily manufactured since they use a common stacking table. This is achieved by directly optimizing the degression law. This has mainly two advantages; the effort needed to modify the structure before manufacturing is significantly reduced and the solution found is better since all the constraints are taken into account from the beginning. Using a global optimization enables the integration of the aeroelastic constraints resulting in a fully coupled sizing and aeroelastic optimization. This results in a better solution since the sizing can directly take advantage of the aeroelastic reinforcements.

An application of the process on a generic Falcon jet with a composite wing demonstrated its potential for optimizing composite structures. With tough aeroelastic constraint the process was able to find an optimal solution which respects both the aeroelastic and structural sizing constraints.

The optimized structural mass obtained with other winglet shapes will be presented. This will show that the process can be used to carry out preliminary configuration studies and is able to determine the optimal winglet shape for a future aircraft by establishing the best compromise between aerodynamic efficiency and added structural mass on the wing.

5 ACKNOWLEDGEMENTS

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