OPTIMAL DESIGN OF FUSELAGE STRUCTURES

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Abstract

There is an industry-wide effort to produce aircraft designs that are more competitive in terms of performance and cost, and to do so with a decrease in development time. To achieve these goals for "Faster, Better, and Cheaper" solutions, the preliminary stage of aircraft design is evolving to include greater detail, and greater consideration for the interaction of multiple disciplines. The dream of fully integrated processes that allow for the creation of optimal designs is slowly, but most assuredly, becoming a reality.

This paper introduces a methodology for the optimal design of fuselage structures for use in an MDO environment. The process integrates a finite element package (NASTRAN or ASAS), in-house pre- and post- processors (UPDATEPROPS and RESERVE), and visualisation (PATRAN) techniques to automate the sizing of the fuselage skins and stringers.

1 Introduction

The constraints on the design of aircraft fuselage structures include limits on the material stresses, buckling loads in the stringers and the skin panels, and post-buckling effects on the skin-stringer combination. The post-buckling behaviour is a nonlinear phenomenon, but its contribution to the design is determined from the results of a linear analysis using design curves derived from empirical data. This technique is used for its computational efficiency as compared to a full non-linear FE analysis for each load case.

For certification by the civil aviation authorities, several hundred load cases are analysed on a finite element (FE) model of the aircraft, and processed through a post-processor (RESERVE) to verify that the design constraints are satisfied. In the past, this process has been primarily used to verify the integrity of the structure following the design – normally resulting in a conservative design.

The aim of this work is to use the analytical capabilities of the FE-RESERVE process to optimally design the fuselage structure in a semi-automated environment. This is performed with consideration for all of the critical load cases for the design.

The platform for this study is the Dash-8 Q400 aircraft manufactured by Bombardier Aerospace at its Toronto site.

2 Finite Element Model

The structural design is driven by the internal loads derived from a Finite Element (FE) analysis of the aircraft. The finite element model comprises the entire aircraft, as illustrated in Fig. 1. This type of model is sometimes referred to as a "loads" or "stiffness" model. The philosophy is to model the structure with sufficient detail to evaluate the internal loads necessary to size the skin panels and stringers. In general, the fuselage is discretised to a single element between each stringer and each frame. This is convenient for the optimisation process because the number of design variables is determined by the number of elements under consideration. For example, just

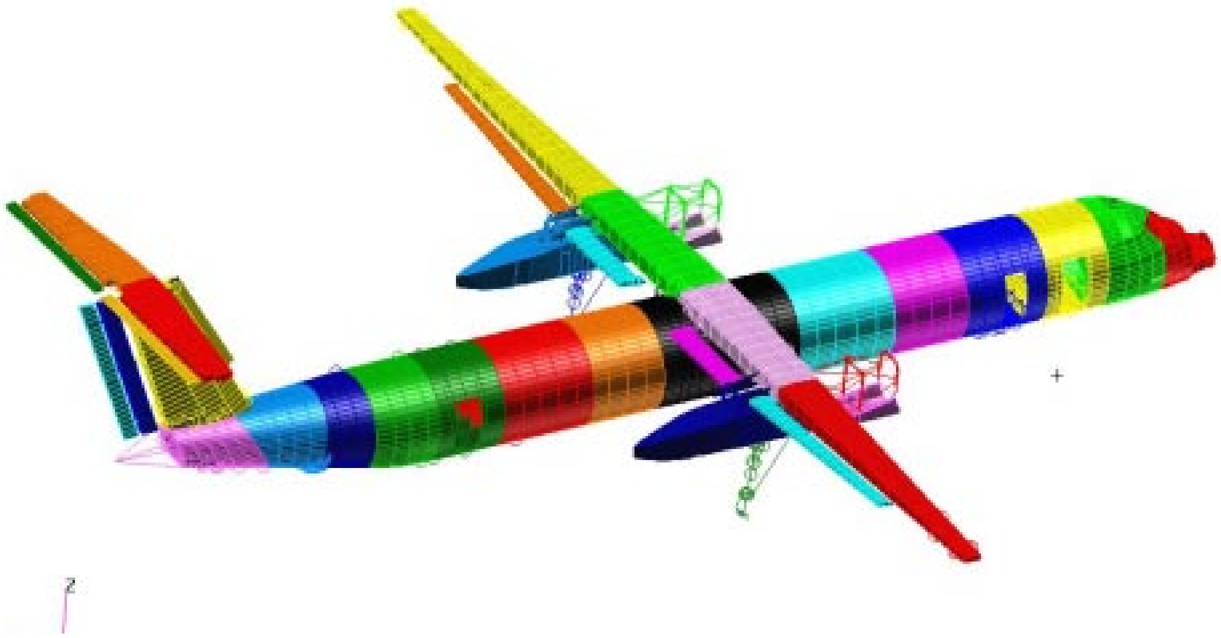

Fig. 1. FEM model of Dash8-Q400 aircraft illustrating substructures.

one design variable is used to describe the thickness of a panel between adjacent stringers.

For the example presented here, the FE code ASAS (from Century Dynamics) was employed, although the methodology works equally well with MSC/NASTRAN, or other FE software. ASAS has advantages for this type of design methodology because of its sub-structuring capability. The aircraft model contains a large number of degrees of freedom (>10,000) many of which are remote from the area of design (e.g. wing, nacelle) but contribute to the design through their associated loads. Through the use of sub-structuring, the iterative process is made more efficient by creating substructures for these components. Doing so increases the computational efficiency because the equations representing these components need only be reduced once for the entire design process. Moreover, the internal loads are recovered only on those components being designed.

Structural loading is achieved using sets of "unit" load cases which are distributed throughout the aircraft structure. These cases are combined using superposition to yield a consistent case including inertia effects and external loads. Each load case represents a consistent set of loads, rather than an envelope case which is typically formed to represent several loading conditions. Using consistent cases allows individual locations to be designed by particular load levels and will, in general, allow for a more optimal design than could be achieved using envelope cases.

The total number of cases considered is equal to 723, yet not all cases are critical to the design for each section of the fuselage. Only those load cases critical for that particular section of the fuselage are recovered and processed. This filtering of load cases is performed from the integrated loads envelope of the aircraft structure. As an example, a manoeuvre case which produces high vertical bending moment on the rear fuselage, may not be relevant for the front fuselage because it is overwhelmed by the magnitude of other loading conditions. Applying this methodology reduces the number of load cases to as few as 104 and at most 387, for any particular fuselage section.

This results in a distinct improvement in the efficiency of the design process. Since it is iterative, eliminating that which is not required can have a substantial effect on the elapsed time between iterations.

3 RESERVE Post-Processor

Two related buckling phenomena important in aircraft design are panel buckling and stringer buckling. During panel buckling under shear loading, the deformation may produce diagonal buckles which have the capability to support tensile loads. This results in a complex, yet efficient, transfer of shear load from the panel to the stringer, commonly known as *"tension field beams"*. The original work by Wagner [1] is still used as a reference. The stringer buckling is attributable to primary compressive loads as well as to secondary stresses caused by the diagonal tension effect.

A standard procedure for finding panel allowables, stringer allowables, secondary loads and secondary moments is given by Bruhn [2]. Because of the extensive computational effort, this is a very tedious procedure (especially for large structures).

The RESERVE program is an automated method for calculating reserve factors for typical skin/stringer aircraft structures. Selected standard methods of structural analysis were utilized to perform this task.

The primary internal loads in the structure are calculated by the finite element analysis. RESERVE extracts these loads from database generated by the FEA along with the available geometric and material properties of the idealized structural elements. Given an input list of stringer element numbers, algorithms automatically determine stringer, skin and frame element connectivity, local coordinate frames, and applicable material and geometric properties.

Subsequently, the program calculates the interactions between the skin and stringer due to the nonlinear effects of skin/stringer buckling. The influence of the load redistribution and secondary stresses due to diagonal tension effects are included in the calculation. The NACA TN 2661 [3] method of diagonal tension analysis was used for this purpose, with modifications presented by Mello [4], Tsongas [5] and Mohaghegh [6]. As a final step, the program combines the primary and secondary stresses and calculates a reserve factor for each element under consideration.

3.1 Reserve Factors

The reserve factors for various skin and stringer failure modes are output for the critical design load cases. The stringer failure modes include:

- Tension failure,
- Forced crippling, and
- Secondary bending.

The following failure modes are included in the panel analysis:

- Gross panel failure,
- Von-Mises stress (tension),
- Principal stress (compression), and
- Permanent buckling (limit load).

The final RF value assigned to a particular element is the minimum for all of the failure conditions considered.

4 Design Constraints

The RESERVE post-processor is the tool for evaluating the design constraints for the problem by returning a figure-of-merit based upon the various calculations. This "reserve factor", RF, represents the most critical of all the analysed conditions.

Additional constraints to the problem are represented by manufacturing and geometric tolerances. For example, minimum skin gauge, maximum stringer height, and minimum web thickness act as side constraints to the design process.

5 Design Variables

The design variables comprise the sizing of the skin and stringers of the fuselage - the thickness of the skin panel, and the cross-sectional geometry of the stringer. The frames are currently not designed in this process.

For the methodology to be effective as a preliminary design tool, it is important to have the capability to account for the dependence of the manufacturing processes on the outcome of the design. Thus, a number of different strategies have been implemented to establish how these parameters are updated from the results of the FE analysis.

5.1 Skin Design

Three techniques for design of the fuselage skin have been incorporated into the methodology.

5.1.1 Variable Skin Thickness

The first method allows for complete freedom in specifying the thickness of the skin. With the possible exception of some side constraints specifying the minimum and maximum gauge, the thickness of each skin panel may be driven to any value by the internal loads and the design constraints. This effectively represents the manufacturing processes of chem-milling and machining.

5.1.2 Discrete Skin Thicknesses

Aluminum sheet is normally supplied from their rolling mill to the airframer in particular thickness gauges (e.g. 0.012, 0.016, 0.020, 0.025 inches). To allow for this type of construction in the absence of machining, the design variables are updated in discrete values. Using this method, the “optimal” value for the skin thickness may actually exist somewhere between the discrete values supplied by the designer. This presents a problem for convergence as the solution can flip-flop

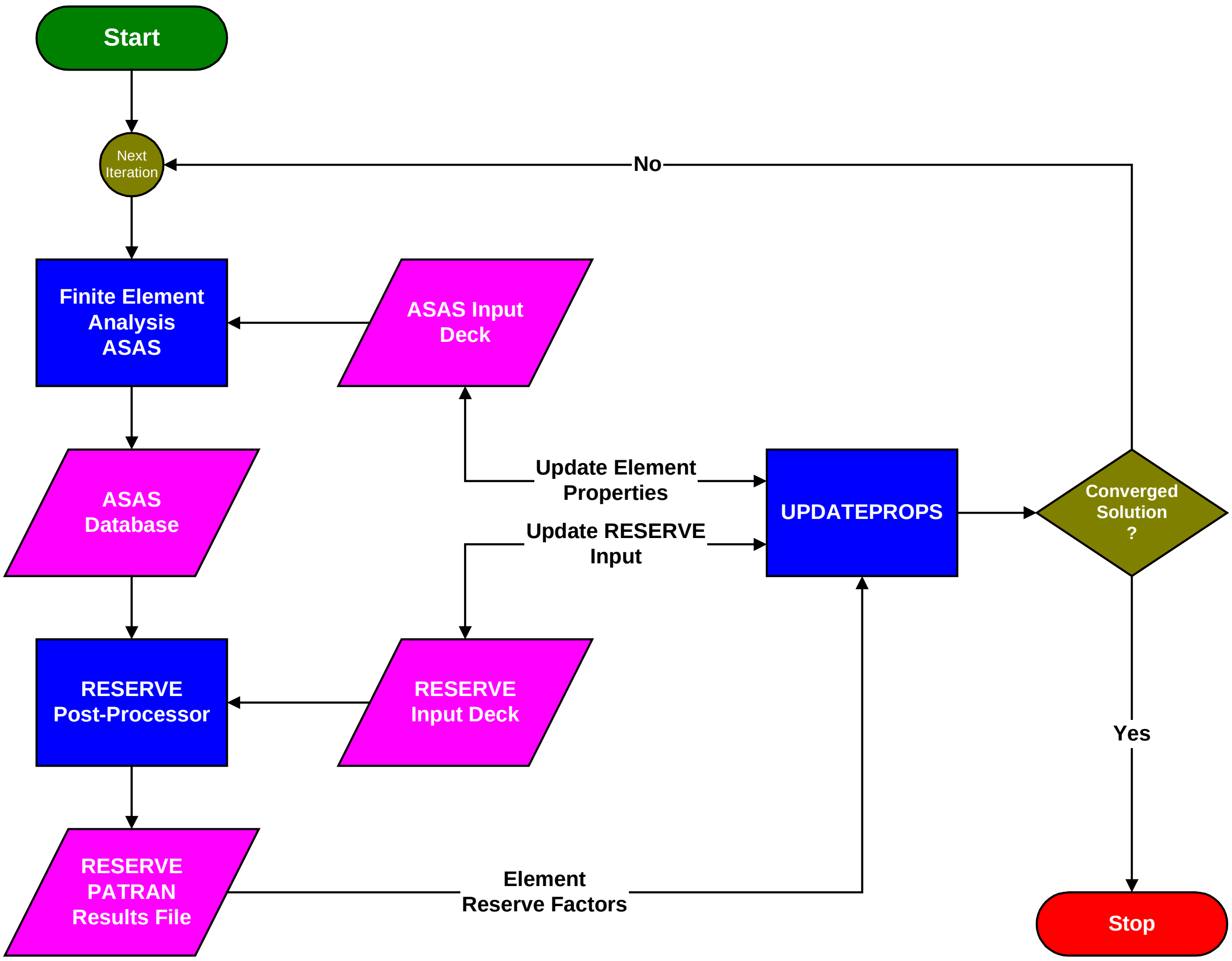

Fig. 2. Fuselage design methodology.

between two values from one iteration to the next. Such issues are eliminated by tracking the solution, removing the offending element from the design process while constraining its design variable at the appropriate value.

5.1.3 Specified Skin Laminations

Using a methodology similar to that presented above, the capability to allow the design to consider specific laminations of a built-up material has been implemented. In this instance, the purpose is not to vary the orientation of the material, or its constituents. Instead, the assumption is that “stock” lamination sequences are to be used for the design of the structure. This is certainly the case for fibre-metal laminates (FMLs) where the material is manufactured with specific layups, and can also apply to many composite material configurations.

This manufacturing option has the added complexity that the material properties are changing in addition to the material thickness. However, the material properties are not design variables themselves because they are dependent upon the thickness of the material.

5.2 Stringer Design

Three methods for stringer design have been devised.

5.2.1 Stringer Scaling

As an analogous method to the variable skin thickness methodology is the capability to design the stringers by allowing the geometry to vary continuously. Continuous scaling of the original beam cross section is supported by updating the geometry of the individual stringer ligaments.

5.2.2 Discrete Stringer Selection

Structural designers often use a set of specific beam sections for the design of the structure. This method allows for automated selection from a list of specified beams.

5.2.3 Stringer Profiling

Often, a particular extrusion is selected for a design because it is efficient to machine the details in a stringer from the larger extrusion. By allowing the geometry of particular ligaments on the stringer to be iteratively updated, a “coarse grained” profiling may be modelled.

5.3 Design Exclusion

Any portion of the fuselage barrel may be included, or removed from the optimisation process as determined by the user. This allows the designer to impose constraints on the design process (e.g. due to manufacturing or analytical limitations) by specifying the skin thickness or stringer geometry in particular areas.

An example of this restriction is the skin thickness in the belt along the fuselage side containing the windows. The design in this region is influenced by the stress concentration at the cut-out and the supporting structure which surrounds the periphery of the cut-out. Such details in the design of the structure require information not yet available to this process, so the property values are specified, and are not designed.

Proper consideration of the regions being designed must be given by the designer. Firstly, by removing a region from the design process, the designer has fixed its properties. It is possible that the redistributed load could cause the load levels in this non-designed area to exceed tolerable values, so these regions must still be verified for acceptability following the optimisation process.

6 Updating Design Variables

The core of the design process is the software programme which updates the design variables based upon the RFs. This activity performs several tasks:

- Calculate new properties (update the design variables),
- Modify the Finite Element model,
- Update the RESERVE inputs, and
- Save results for visualisation with PATRAN.

The design process described in this paper falls into the category of “fully stressed design”. But because this process does not involve the calculation of gradients, a method was devised to prevent the solution from overshooting and oscillating between iterations. Through the application of a relaxation coefficient, the new properties are updated at a reduced value. Considering the design variable of a panel at the n^{th} iteration (V_n), the value at the subsequent iteration is:

$$V_{n+1} = V_n / RF^Q \qquad (1)$$

where Q is a specified relaxation factor.

Testing of the methodology has shown that the stringer sizing is more likely to suffer from overshoot than the panel sizing. Thus a different Q is used for the panels and stringers. Experience has shown that a value close to unity can be successfully used for the panel design, but 0.8 is more effective for the stringer design.

The use of the relaxation factor does not prevent oscillations for those methodologies that use discrete design values. In these instances, the design variables are monitored between iterations. When a specified number of oscillatory cycles (normally two) have been observed in the evaluation of the design variables, the value is fixed at the more conservative of the two, and the element is removed from the design set.

7 Methodology

The design methodology is an iterative process which is illustrated in Fig. 2. It starts with an FE analysis of the aircraft structure using the critical load cases. The post-processor evaluates the design constraints and returns an RF for each element in the design model. From this quantity, the element properties of the FEM are updated and the FEM analysis is repeated with the new design.

The process continues until the incremental change in the design variables is less than a tolerance value, and remaining elements have been removed from the design set. Typically, fewer than 10 iterations are required for convergence.

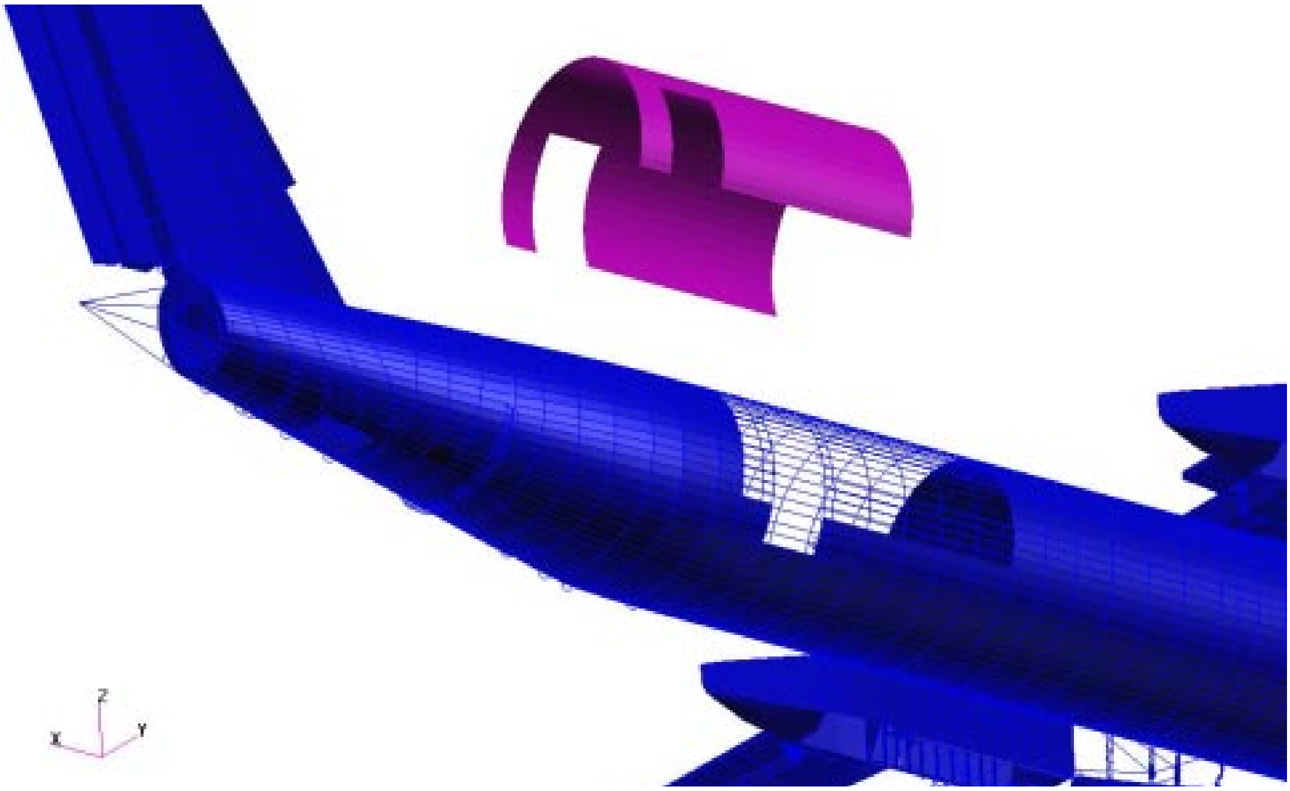

Fig. 3. Fibre-metal laminate panel for the Dash-8 S400 CAST article.

8 Application

The methodologies developed for this project were applied to the design the Dash-8 Q400 Complete Aircraft Structural Test (CAST) panels for the Fibre-Metal Laminate (FML) demonstrator programme. The Finite Element model of the Q400 is shown in Fig. 3 with the replacement panels highlighted. These panels were optimally designed using the complete set of design loads from the Q400 programme. The starting point for the design is illustrated from the PATRAN plot of skin thicknesses in Fig. 4. The RFs corresponding to this design are illustrated in Fig. 5.

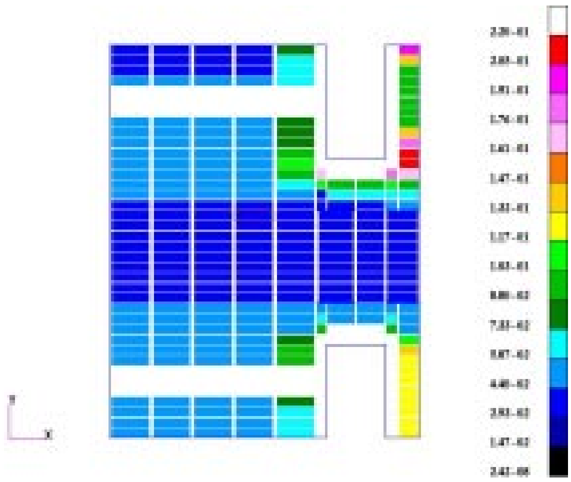

Fig. 4. FML panel starting design.

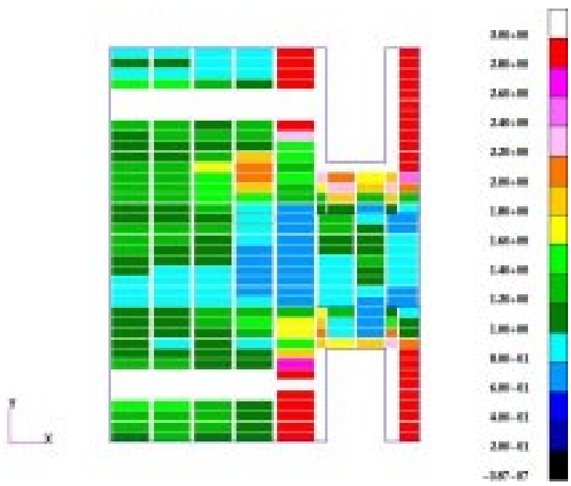

Fig. 5. FML panel starting Reserve Factors.

The material used for the design process is the fibre-metal laminate GLARE. This material is supplied in discrete thicknesses by the manufacturer based upon particular lamination sequences. After ten iterations, the solution has converged to the final design as illustrated in Fig. 6. Notice that the panel thicknesses correspond to the particular values of GLARE3 and GLARE4 laminates. The elements around the door cut-outs and the windows do not appear in the results of Fig. 6 because their properties have been explicitly defined and therefore, are not part of the design model.

The reserve factors (RF) for this design are shown in Fig. 7. A value of unity means that the panel exactly meets the design criteria for the most critical load case. A value greater than one means that there is extra capability and a value less than one indicates that the design criteria are not satisfied. Notice that most of the designed elements have RFs equal to, or slightly larger than unity. The elements adjacent to the door cut-outs and the window belt have higher RFs because their properties were prescribed, and thus not part of the design set. The other elements with RFs typically around a value 2.0 have the higher values because the minimum material thickness has been reached for the structural element; thus a side constraint has become active.

Through the use of PATRAN, the designer/analyst also has the capability of visualising the failure modes (Fig. 8) and the corresponding critical load case information (Fig. 9). This additional information is useful for providing insight into those factors that influence the design of the structure.

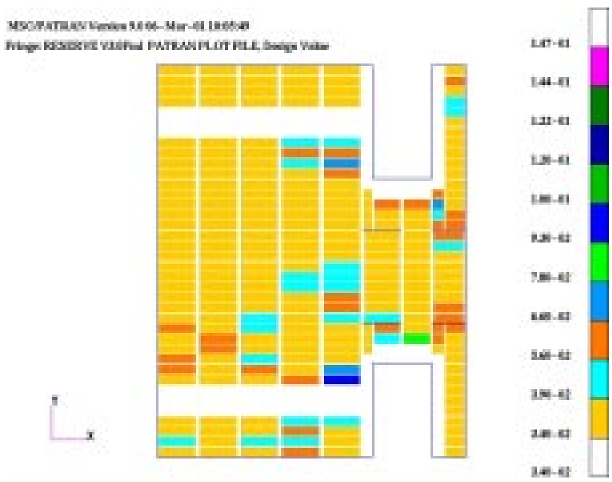

Fig. 6. FML panel final design.

9 Conclusions

The methodology presented here can be used for the preliminary design of an aircraft fuselage structure. It should be noted that the results of this process represent the starting point for the designer. The FML layout shown in Fig. 6 is not necessarily practical from a manufacturing perspective, but can be used to produce a structure which is optimal from the perspective of the applied loads.

The optimal design of the GLARE panel realised *a saving of 20% in the structural weight* over the original design. A portion of the weight savings is due to the greater specific performance of the FML material, but the remainder can be attributed to the optimal design process. Moreover, the capability of this methodology as part of a complete MDO process for preliminary aircraft design is demonstrated. As to efficiency, the process described here shortens a task that has historically taken several months to complete, to one which can be accomplished in several days.

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Fig. 7. FML panel final Reserve Factors.

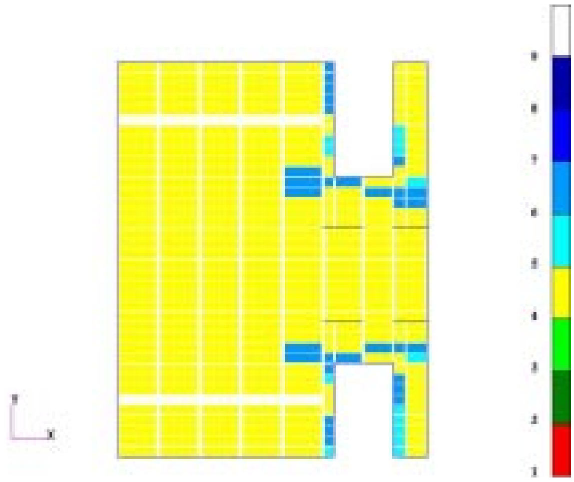

Fig. 8. FML panel failure modes.

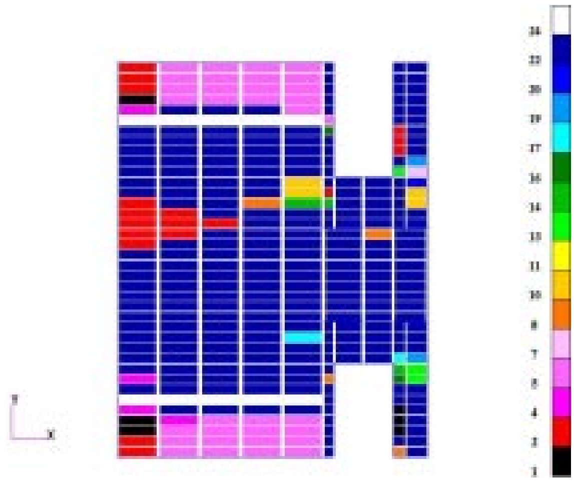

Fig. 9. FML panel critical load case plot.