Airframe Wingbox Preliminary Design and Weight Prediction

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Currently Collier Research is evaluating the performance of many panel concepts for highly loaded, commercial transport wingbox structures. Software development is taking place as well as many new commercial contracts to further the effort. Simultaneously the HyperSizer® commercial software is being used world-wide on many commercial aircraft for the analysis and design sizing of composite and metallic aircraft structures. This paper focuses on trade studies performed on an aircraft wingbox structure. The wingbox is modeled after a commercial transport jet. The loads on the upper and lower wing skin surface are axially compression paired with high shear loading caused by wing twist. Weight trends are quantified considering all possible design possibilities in order to determine the most structurally efficient combination of composite layups and panel cross section dimensions to achieve the lightest weight. Included are results obtained with HyperSizer for Metallic Integral Blade and Bonded Zee Stiffened panel concepts as well as Composite Hat and Tee Stiffened panel concepts. The HyperSizer optimum designs have scored based on the weight maturity level (WML) of the panel design. The study shows the composite hat stiffened panel is the lightest concept for a wingbox structure and is 30% lighter than the lightest metallic design.

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2 Introduction

Currently, HyperSizer is used by the aerospace community primarily for product development (PD). This is due to its ability to perform rapid structural analysis and design sizing that includes many failure analyses at all load conditions for all areas of an airframe. The intent of this research is to mature HyperSizer's usability for final design and certification of traditional transport wing box structures. A primary focus of this report will be to demonstrate the HyperSizer's accuracy and ability to be used for preliminary design. In order to accomplish this a finite element model is used to derive internal element loads and size the wing box structure based on robust optimization methods.

This is in contrast to optimization for strength or stability using FEA packages such as MSC/NASTRAN Sol 200 that require the user to code in failure analyses. This process usually incorporates approximate analyses that are not the same ones used during final design. Robust optimization is defined here as one that uses most of the same analyses in the optimization process as used in the final detailed analyses. Therefore all potential failure modes are evaluated early and to a high degree of accuracy in trade studies. Robust optimization also means finding several design solutions that are able to achieve the same performance.

The HyperSizer® software is used to perform panel optimization for each concept considered, for all manufacturable layups, and as a function of ringframe spacing. This paper presents weight trends for each panel concept as a function of ringframe spacing. In order to establish the correct weights of each design, accurate failure analyses were performed by HyperSizer. All panel concepts reported achieved positive margins of safety for all relevant failure modes and for all load cases.

The results reported here are those from Collier Research Corporation developers of the HyperSizer software.

2.1 *About HyperSizer*

HyperSizer software automates the optimization process for stiffened panels, sandwich panels and open and closed cross section beams. To use the software the user applies general edge loadings and/or boundary conditions in the software interface and HyperSizer solves for the resulting ply level stresses and strains then evaluates the structural integrity using over 100 different failure analyses. The failure analyses include traditional industry methods, modern analytical and computational solutions, as well as some unique approaches [8]. Methods development has been on-going since the late 1980's to present.

2.2 *Purpose of this Paper*

The intent of this research is to mature HyperSizer's usability for final design and certification of traditional transport wing box structures. A primary focus of this report will be to demonstrate the HyperSizer's accuracy and ability to be used for preliminary design.

3 Weight Maturity Level

Table 2, Weight Maturity Level (WML) for Panel Concepts Sized for Wingbox Structure

Panel Concept (Metallic and Composite)	Analysis Foundation	Industry Use	FEM/FEA	Sizing Time (LOE)	Adhering to Best Practices	Total WML	Weight impact of Increasing WML from current level
Level 1 - Blade Stiffened	10	10	5	3	10	30	Lighter by 10%
Level 2 - Hat Stiffened (c)*	10	10	5	10	10	100	Lighter by 15%
Level 2 - Tee Stiffened (c)	10	10	5	5	10	50	Lighter by 10%
Level 2 - Zee Stiffened	10	10	5	5	10	50	Lighter by 10%
Level 3 - Hat Stiffened (c)	10	10	5	0	10	0	N/A
Level 3 - Tee Stiffened (c)	10	10	5	0	10	0	N/A
Level 3 - Zee Stiffened	10	10	5	0	10	0	N/A

(c) Composite Panel Concepts
* Total Score Normalized to Highest WML

The weight maturity level (WML) is not necessarily a measure of performance of the panel concept for the given structure, nor does it describe the accuracy of the weight prediction. The weight maturity level is a measure of confidence in the weight statements and is comparable to a technology readiness level (TRL) or a manufacturing readiness level (MRL). Higher fidelity panel designs are represented with a higher total WML. Concepts such as the hat stiffened panel heaver high WMLs so their weights are not expected to drastically change as the project continues, provided no major load/geometry or design criteria changes are imposed.

4 Airframe Wingbox Background

4.1 *Aircraft Type*

The wingbox is modeled after a traditional commercial transport jet most closely representing a Boeing 737 aircraft.

4.2 *Aircraft GTOW*

The Gross Take Off Weight (GTOW) of the aircraft is 170,000 lb.

4.3 *Wing Load*

The external loads are based on the flight characteristics of the aircraft. For this analysis the following assumptions are made to determine the load sets applied to the model. (1) The Gross Take Off Weight (GTOW) of the aircraft is 170,000 lb, (2) the fuel weight is 41,000 lbs (6900 gallons). The top skin surface has upward (positive) aerodynamic pressure and the bottom skin surface has a downward (negative) fuel pressure that is mapped to the elements to represent the airfoil effect. Note that the fuel pressure at the root of the wing is twice as high as the fuel pressure on the mid portion of the wing.

There are three unique mechanical load sets: (load set 1) A 2.5 g up gust that includes a 50,000 lb thrust load of both engines. The applied loads for this load set include the load caused by aerodynamic pressure which is equal to 170,000 lb * 2.5 g = 425,000 lb and the load caused by the fuel weight which is equal to 41,000 lb * 2.5g = 102,500 lb. (load set 2) A 1.0g roll maneuver where both engines impose a 50,000 lb thrust load, the aerodynamic pressure load is equal to 170,000 lb * 1.0g = 170,000 lbs and the fuel pressure is 41,000 lb * 4.0g = 164,000 lb. (load set 3) 2.5g main landing gear load caused by a severe taxi bump is represented by 170,000 lbs * 2.5g = 425,000 lbs.

4.4 *Structural Layout*

4.4.1 Rib Spacing

Wing ribs transfer load from the wing skin to the primary spars. The physical rib spacing should be determined early in the preliminary design phase, many times before a finite element model is created. Since the rib weight is a large contributor to the overall weight of the wing, it is important to include the rib weight in preliminary weight estimates.

The weight optimum rib spacing is determined using closed form equations for calculating panel bending moments and out-of-plane shears due to fuel pressure. Then both the panel and rib optimizations are performed with Hypersizer. The weight results are recorded for each rib spacing to generate weight curves for the derived loadings and boundary conditions. Fig. 4.4.1 graphs the results and aids in the visual determination of the proper rib spacing to use for the next step of FEM(s) creation.

4.4.1.1 Determining Optimum Rib Spacing without a Finite Element Model

Fig.4.4.1, Optimum Rib Spacing trade Study using HyperSizer Workspace Approach. Unit weight, psf, on vertical axis and span distance horizontal axis, inches.

When pressure effects and the resulting beam-column non-linear response is considered, the optimum rib spacing is seen to fall around 28"- 32" which closely matches traditional wing designs. Given a range of rib spacings that produce a similar weight structure, it is advantageous for cost and fabrication reasons to choose the highest optimum rib spacing.

4.4.2 Spar Spacing

Spars are used in wing box structures to transfer shear load between ribs and distribute the load based on the web stiffness of the ribs.

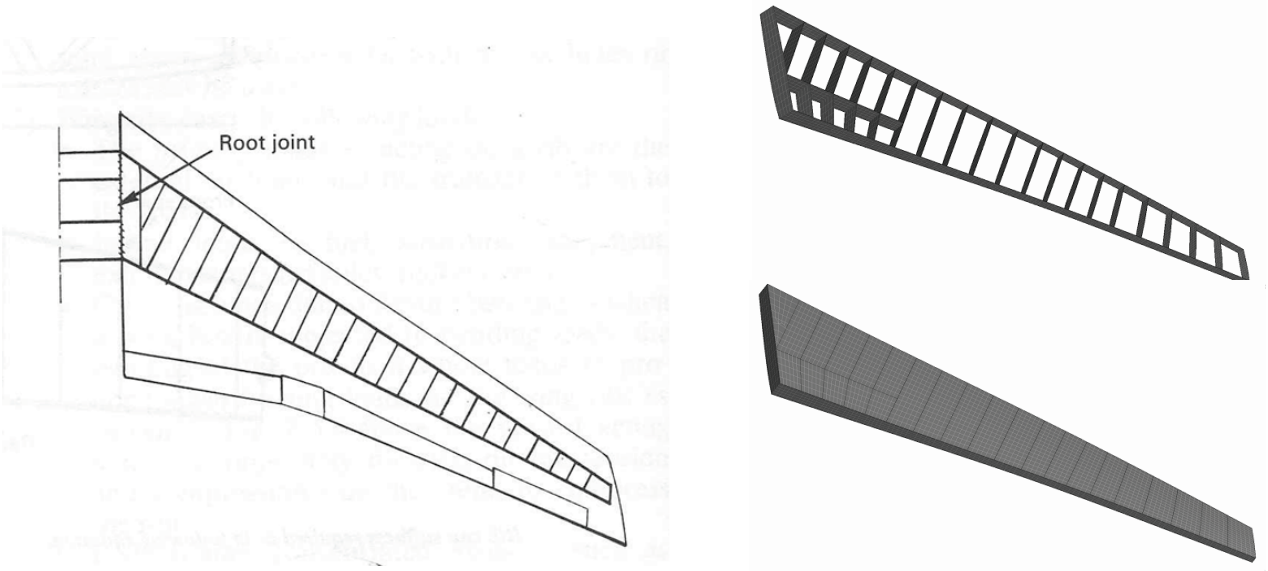


Fig. 4.4.2, The wing box spars and ribs as shown to the left image of a traditional transport wing, is modeled in the FEM with 31" rib spacing in the sweep direction. The spar spacing is modeled as XX at the root tapering to XX at the wing tip. The top image shows the internal substructure, the bottom image the OML surface panels.

The spar spacing imposes a physical transverse buckling span on the skin and rib panels. The two main spars taper so the spar spacing at the wing root is just over 90in and the spar spacing at the wing tip is approximately 35in.

4.5 *Stiffness Requirement*

The required membrane stiffnesses of the wing skins is determined per panel bay by setting a design limit for wing tip deflection. For this analysis two nodes are identified as control points for the purpose of calculating wing deflection and limiting the aeroelastic deformation. Fig. 4.5.1 illustrates the nodes chosen to track the wing tip deflection of the wing box.

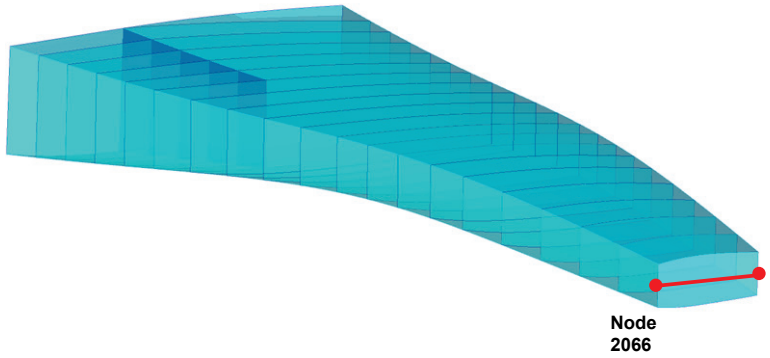


Fig. 4.5.1, Nodes at Tip of wing chosen to track aeroelastic deformation

A general assumption is made that the stiffness increase is linear with respect to tip deflection. For example, if the wing tip deflection is 15% higher than the target tip deflection then the required A11 for each panel in the top and bottom wing skins is increased by 15%.

4.5.1 Wing Tip Deflection Limits

For the transport wing, a total tip deflection limit of 30" at ultimate load is imposed. To meet this requirement the upper and lower wing skins will increase the longitudinal membrane stiffness (A11).

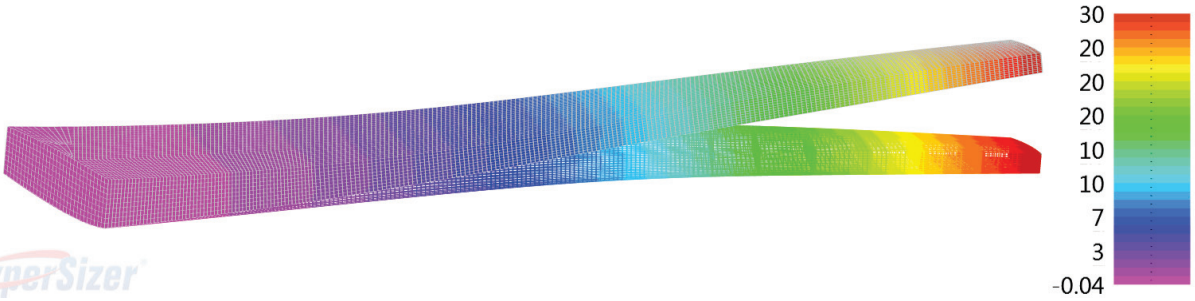


Fig. 6.6.2, Deformation Response of Wing, Load Case 103: Up Gust 2.5G + Thrust, causes largest wing deformation

4.5.2 Wing Twist Limits

The torsional membrane stiffness (A33) for each component is determined by imposing a global twist limit. This has not been applied to this project.

4.6 *Skin Thickness Requirements (minimum gage limits)*

4.6.1 Metallic

Metallic skin minimum gage thickness for AL 2219 = 0.1" . For the stiffener, the minimum gage thickness for the web and flange for AL 7050 = 0.05"

Initial sizing of the metallic sheets is done assuming continuous thicknesses. This is readily accomplished with HyperSizer sizing variables. However, each metal alloy can have predefined stock sizes that are available from the Vendor. HyperSizer can conveniently use these as well for discontinuous sheet thickness sizing variables. Unique stock sizes can be defined for each of the individual materials in the HyperSizer database.

4.6.2 Composite

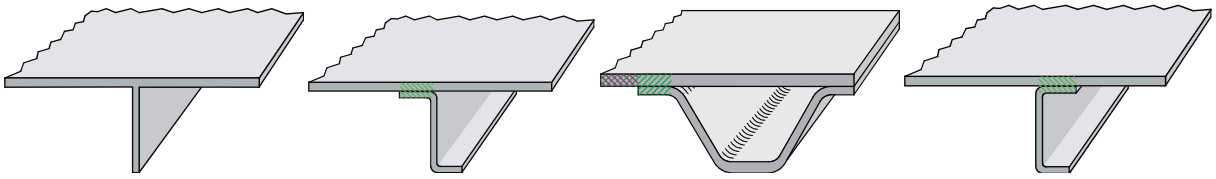
Composite minimum gage is usually determine in terms of minimum number of plies. From the ply thickness, the minimum laminate thickness is set. Alternatively, minimum laminate thicknesses are set based on manufacturing experience and operational considerations such as damage tolerance to hail storms.

5 HyperSizer Sizing Optimization and Analysis Process

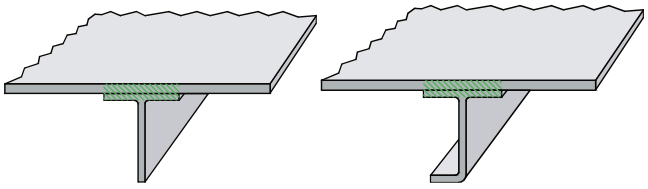
HyperSizer software automates the optimization process for stiffened panels, sandwich panels and open and closed cross section beams. To use the software the user applies general edge loadings and/or boundary conditions in the software interface and HyperSizer solves for the resulting ply level stresses and strains then evaluates the structural integrity using over 100 different failure analyses. The failure analyses include traditional industry methods, modern analytical and computational solutions, as well as some unique approaches. The primary panel/beam concepts available in HyperSizer are illustrated below

5.1 *Available Panel Concepts in HyperSizer*

5.1.1 Stiffened Panel concepts



Integral Blade Stiffened *Bonded Zee Stiffened* *Bonded Hat Stiffened* *Bonded C Stiffened*

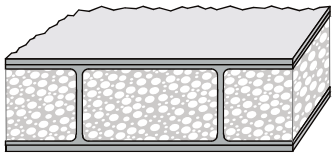


Bonded Tee Stiffened *Bonded J Stiffened*

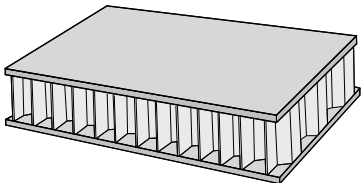
5.1.2 Bonded vs. Fastened Stiffeners

HyperSizer is capable of optimizing both metallic and composite stiffened panels with either bonded or fastened stiffeners. The primary difference in the bonded vs. fastened stiffener analysis is the assumption that no transverse or shear load is transferred from the stiffener flange through the fasteners to the facesheet. Another assumption is the flange of a fastened stiffened panel provides no added buckling boundary condition, so only the local buckling of the entire spacing span is considered during analysis.

5.1.3 Honeycomb Sandwich Concepts

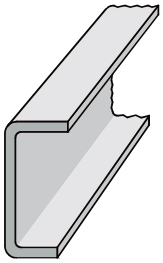


Reinforced Core Sandwich

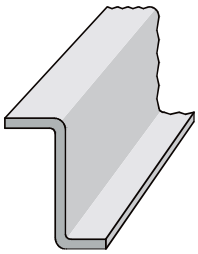


Honeycomb Sandwich

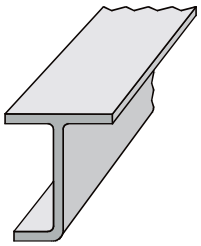
5.1.4 Open Section Beam Concepts



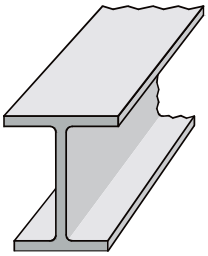
C-Channel Beam



Zee Beam



J Beam



I Beam

5.2 *Panel Accuracy with HyperSizer Smeared Approach*

The first step in characterizing any HyperSizer panel concept is in formulating the strain response of a panel to an applied load or thermal environment. All HyperSizer analysis are built on this foundation. HyperSizer analyzes stiffened panels composed of arbitrary composite laminates through stiffener homogenization, or "smearing", techniques.

The primary advantage of smearing the stiffeners into the panel is the freedom to optimize all panel variables simultaneously. Without this technique, panel variables such as stiffener spacing, stiffener height and flange width are discreetly set into the panel geometry. The different modeling techniques for stiffened panels are illustrated in figure 5.2.1.

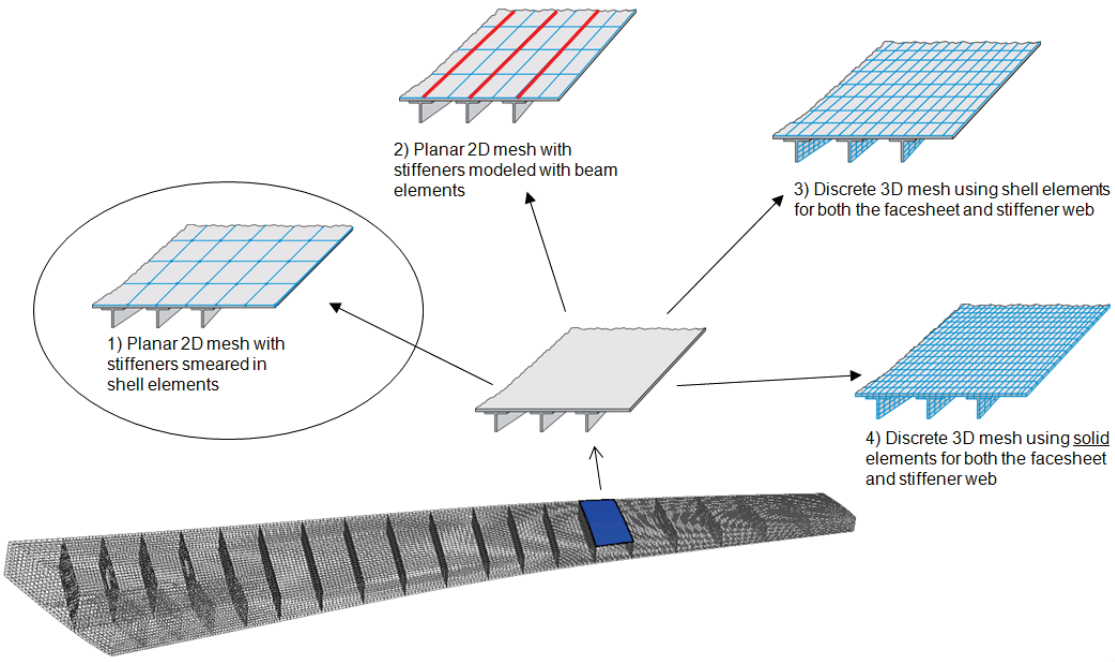


Fig. 5.2.1, Common Stiffened Panel Modeling Techniques. HyperSizer uses the Smeared Technique, (Circled) to optimize stiffened panels

In short the HyperSizer optimization approach is based on the formulation of the ABD stiffness matrix for the entire smeared panel. Once the stiffness matrix is obtained, the strain response of a panel can be determined for any in-plane loading. HyperSizer's Thermoelastic Stiffness formulation is described in detail in ref [1].

5.3 *Statistically Determined Design-to Loads*

HyperSizer optimizes using structural components. Usually many finite elements are used to model a structural component. Each element in the component can be in a different load state which presents a 'pulling loads' problem from elements at a higher load state.

Designing the entire component to the maximum element load could be far too conservative and result in overweight designs. This is especially true if the modeled panel has substantial load variation from mid-span to edge, or from one edge to another edge.

To resolve the "Pulling Loads problem" HyperSizer post-processes the computed FEA element forces by applying statistical analyses to each loading component (Nx, Ny, Nxy, Mx, My, Mxy, Qx, and Qy) for each individual load case. HyperSizer uses statistical methods to determine the appropriate design-to load which will account for possible FEA solution non-convergence due to a lack of mesh refinement.

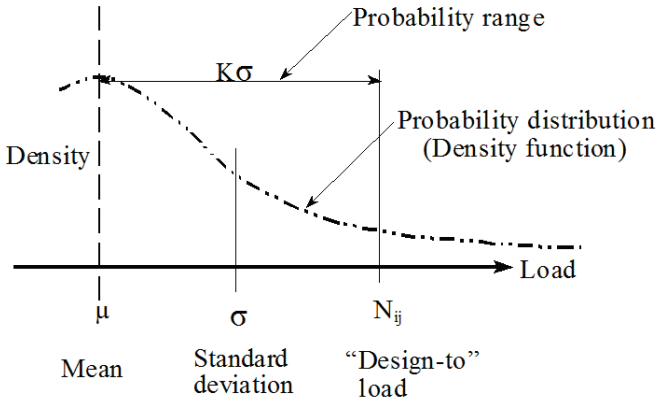


Fig. 5.3.1, Post-Processed Design-to Loads used for Strength Analysis determined by a user-defined statistical loading Method

The higher the K standard deviation (σ), the higher the confidence limit is for capturing peak loads. For the wing box optimization study, a 2-sigma statistical average is enforced. The statistically averaged design-to loads are further resolved into individual panel segment forces for strength and instability analyses such as material strength, local buckling and crippling. The loads are further resolved to derive ply-by-ply stresses/strains.

For panel instability, HyperSizer uses a different statistical approach, which is dependent on an integrated, compressive type load rather than an element peak load that may be located at the panel's corner. A key aspect of this approach is to statistically determine the percentage of the component's area that is in the compressive, buckling zone and integrate the compressive magnitude over that area.

5.4 *Optimization Sizing Variables*

The panel sizing process begins by specifying a minimum and a maximum thickness or width bound, a number of permutations (step size) and available laminates for each sizing variable. This allows the optimizer to choose the best material and panel dimensions to meet the internal loading conditions. During the optimization, HyperSizer generates a margin of safety for each active failure analysis and determines the optimum solution as the lightest panel that passes all active failure criteria.

Stiffened panels like Tee and Zee stiffened panels have many sizing variables. The increased number of sizing variables provide more opportunities for weight savings but this makes the optimization of these concepts increasingly more difficult. To effectively optimize a stiffened panel an automated tool such as HyperSizer is needed to fully explore all the variables in the design space.

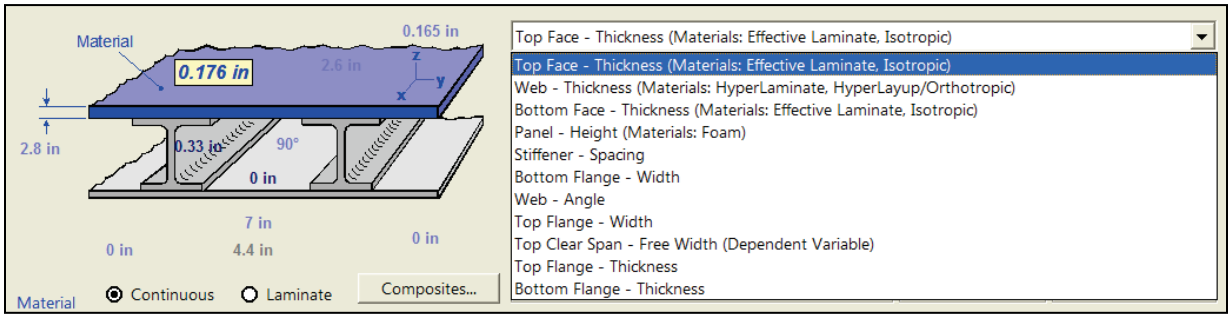


Fig. 5.4.1, Stiffened Panel Optimization Variables

5.5 *Composite Laminates*

For initial sizing, effective laminates are used for sizing variables. In HyperSizer ***Effective Laminates*** (EL) have 'smeared' material properties based on fiber orientation and thickness. Effective laminates are valuable for preliminary sizing since the laminate thickness is a continuous sizing variable like an isotropic material. The EL approach homogenizes a ply-by-ply layup to achieve smeared Ex and Ey modulus. The orthotropic ratio of Ex and Ey is used, but the bending stiffness for each laminate thickness does not distinguish which ply orientations are on the outer fibers and which are near the mid-plane. Hence, the true variation of Dij for a given laminate is lost, which is necessary to provide optimal buckling stable skins, a particular issue for stiffened panels.

Once the appropriate laminate thickness and fiber orientation percents are determined with effective laminates, all panel designs are matured to a discrete laminate design. For detailed sizing ***Discrete Laminates*** (DL) are used and the ply sequence is considered to determine the laminate properties. Since the ply sequence will have a significant effect on the bending properties of the laminate, discrete laminate designs provide higher fidelity buckling results and weight estimates. Maturing the design to discrete laminates will reduce the weight of stiffened

panels because the actual layup sequence can be a benefit. For thin laminates, most likely additional plies will have to be added to achieve symmetric and balanced laminates while meeting fabrication rule of thumbs such as a minimum of 10% in each direction, etc.

5.6 *Analysis Margins-of-Safety (MS)*

In the software, the active failure methods set the design criteria in that HyperSizer will not choose a panel design unless it passes all active failure methods. The failure methods used during stiffened panel optimization are listed in figure 5.6.1 below.

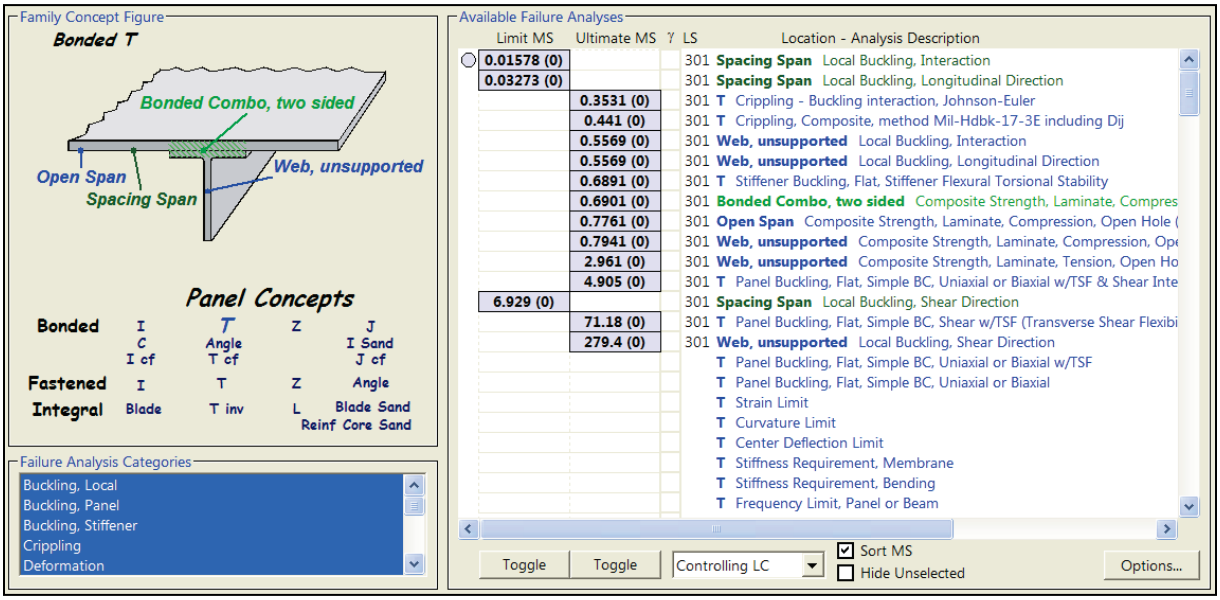


Fig. 5.6.1, Stiffened Panel Failure Methods

5.7 *Stiffened Panel Specific Analysis*

Stiffened panels experience many failure modes not present in other panel concepts. These failures include local buckling of the spacing span and stiffener web, as well as crippling, flexural torsional buckling, bonded flange to skin joint failure and geometric requirements. Each of these complex failure modes is captured in HyperSizer software.

5.8 *Composite Material Allowables/Strength*

The composite material system chosen for all subsequent trade studies is the Hercules graphite epoxy AS4-3502 tape. This is a commercially available material system whose properties are published in many material sources and open literature.

5.8.1.1 Laminate Based Strength Analysis

The allowables for the laminate strength analysis are defined as a function of 45 degree fibers. These allowables are derived from test data and are used for true damage tolerant strength analysis.

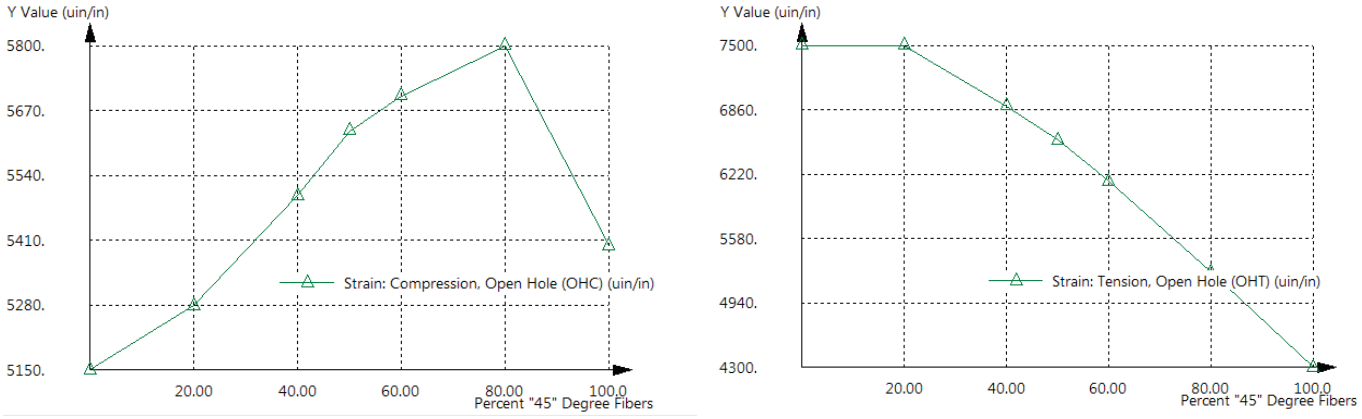


Fig.5.8.1, Strain based laminate allowables are used for all composite strength failure analysis. These curves are a function of %45 degree fibers and are generally based on test data.

5.8.1.2 Ply Based Strength Analysis

Tsai-Hahn is used as the ply based strength analysis instead of max strain. The concern is that max strain alone does not capture the biaxial loading or axial-shear loading interactions properly. Collier Research has performed extensive calibration comparisons to the Wide World Failure Exercises (WWFE) data, along with other publically available data and observed that the Tsai-Hahn quadratic failure theory best matched all the failure data. It should be noted that the test data comparison was for pristine/undamaged laminates.

5.8.1.2.1 Correction Factors

Correction factors are applied for BVID, laminate thickness, Co-Cure etc. to knockdown the pristine lamina and laminate allowables for damage tolerance considerations.

6 HyperFEA

6.1 *Automated coupling with an FEA solver*

HyperFEA automates the process of iterating between a Finite Element Analysis (FEA) and HyperSizer to ensure that sizing loads used by HyperSizer are consistent and converged with load-paths in the FEM. A new additional requirement such as a wing tip deflection, wing twist, or adding a higher fidelity analysis such as non-linear beam column or bonded joint strength, can be inserted into, or removed from the progressive design process at any iteration to quantify their weight impact. Typically as the process is started, the design is heavy and within one iteration with the sizing software is reduced and converged on total weight. The loads were converged for each panel concept.

6.2 *Sectional Bending and Torsional Stiffness*

Each component in the wing skins contribute to the total bending stiffness of the wing. The stiffness contribution of each component in the section is found by the individual membrane stiffness (A11) and distance away from the center of mass. Fig 6.2.1. shows the calculation of the total wing bending stiffness at a representative structural cut.

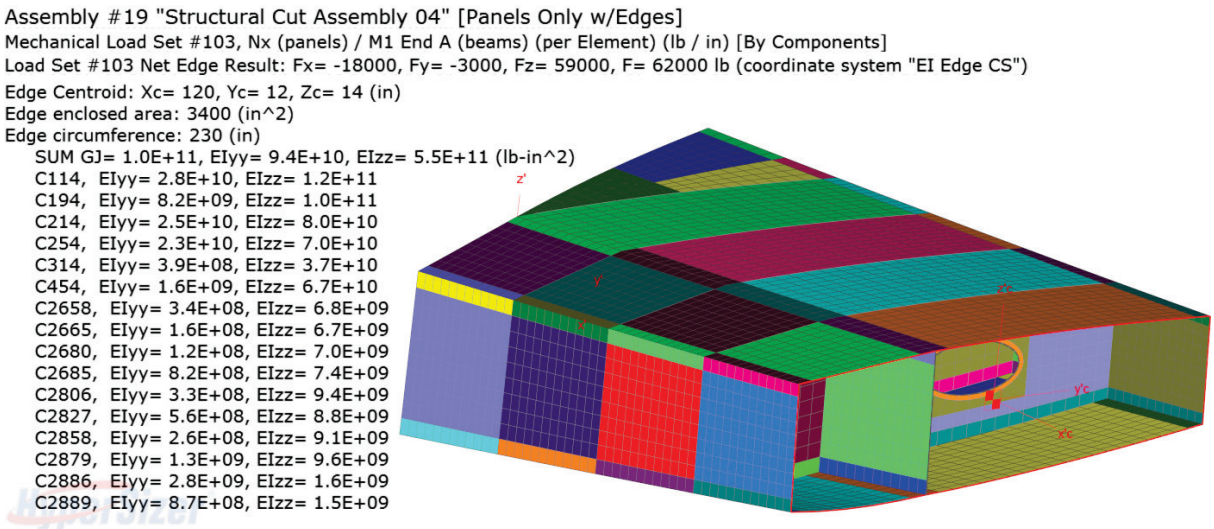


Fig. 6.2.1, Individual Component's membrane stiffness Contributes to the Bending stiffness of the wing at a single sectional cut

The total bending and torsional stiffness at each sectional cut is determined using the method described above. These stiffnesses are a result of material strength and buckling requirements as well as the enforced maximum wing tip displacement of 30in. The stiffnesses are represented graphically in figure 6.2.2.

		30in Tip Displacement		
Edge Cut	X Distance	GJ	Eiyy	Eizz
1	30	1.60E+11	1.80E+11	9.50E+11
2	60	1.40E+11	1.50E+11	8.10E+11
3	90	1.10E+11	1.20E+11	7.00E+11
4	120	1.00E+11	9.40E+10	5.50E+11
5	150	8.40E+10	8.00E+10	4.90E+11
6	180	7.45E+10	6.00E+10	4.00E+11
7	210	6.50E+10	4.60E+10	3.20E+11
8	240	4.50E+10	3.50E+10	2.50E+11
9	270	4.00E+10	2.80E+10	2.00E+11
10	300	3.20E+10	2.70E+10	1.80E+11
11	330	2.80E+10	2.00E+10	1.20E+11
12	350	2.40E+10	1.80E+10	1.00E+11
13	380	1.80E+10	1.60E+10	8.50E+10
14	410	1.50E+10	1.50E+10	7.00E+10
15	440	1.20E+10	1.30E+10	5.40E+10
16	470	9.60E+09	1.10E+10	4.20E+10
17	500	7.70E+09	9.60E+09	3.20E+10
18	540	6.10E+09	8.00E+09	2.40E+10
19	570	4.10E+09	6.90E+09	1.80E+10

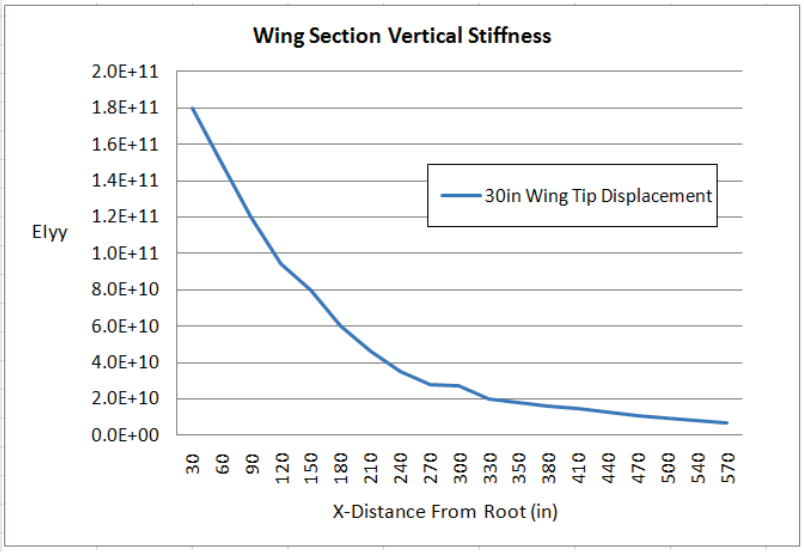


Fig. 6.2.2, Total Bending and Torsional Stiffness at Each Sectional cut, required to enforce a maximum wing tip deflection of 30in

6.3 ***Required Membrane Stiffness (A11)***

The required membrane stiffness for the upper and lower skins, to enforce a maximum 30in wingtip displacement are represented in figure 6.3.1.

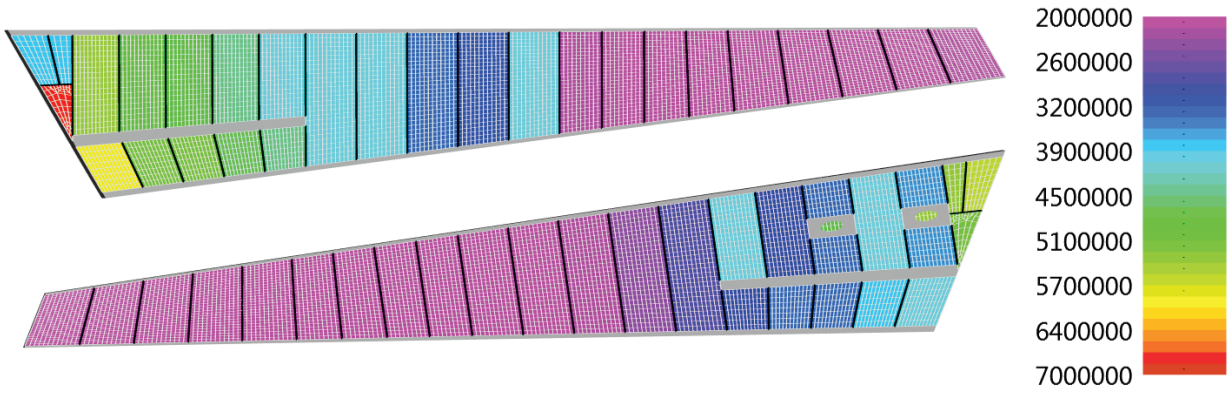


Fig. 6.3.1, Upper and Lower Wing skin panel, membrane stiffness, required to enforce a maximum wing tip deflection of 30in

7 Level 1, Low WML: An initial weight estimate for the entire wingbox using one panel concept

As discussed in the previous section. HyperFEA is used to converge the internal loads for this one panel concept. For the initial weight estimate, the entire wing skin is sized as a Metallic Blade Stiffened panel concept. The cross section dimensions are varied to determine the 'ball park' wing design to meet all active failure criteria.

7.1 *Internal Load*

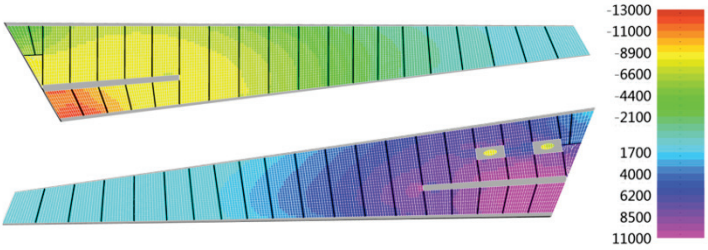


Fig. 7.1.1, Metallic Blade Stiffened Wing Skin, Nx Load,(Load Case 1: Upgust +Thrust)

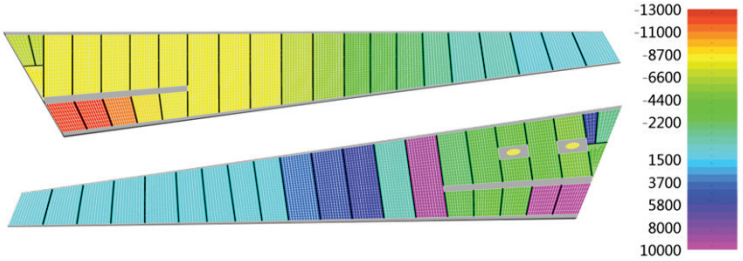


Fig. 7.1.2, Metallic Blade Stiffened Wing Skin, Design-to Nx Load,(Controlling Load Case)

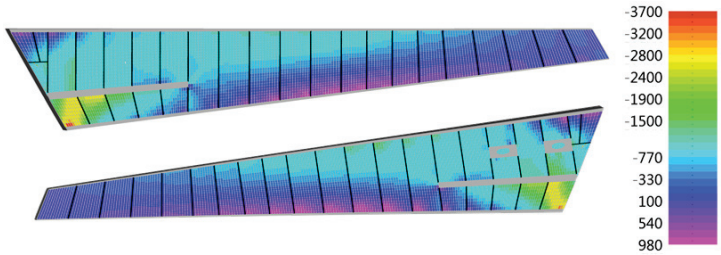


Fig. 7.1.3, Metallic Blade Stiffened Wing Skin, Ny Load,(Load Case 1: Upgust +Thrust)

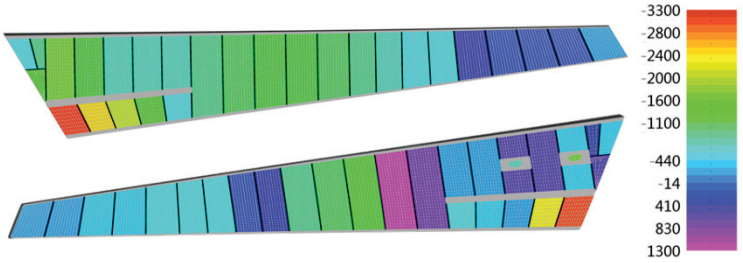


Fig. 7.1.4, Metallic Blade Stiffened Wing Skin, Design-to Ny Load,(Controlling Load Case)

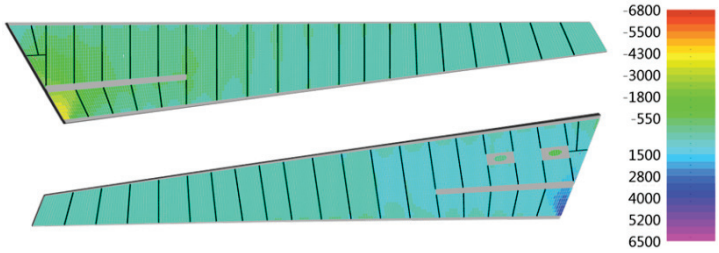


Fig. 7.1.5, Metallic Blade Stiffened Wing Skin, Nxy Load,(Load Case 1: Upgust +Thrust)

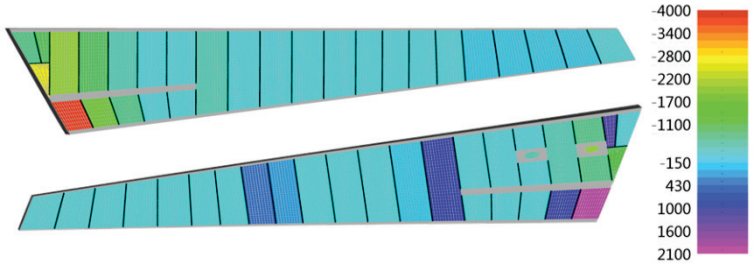
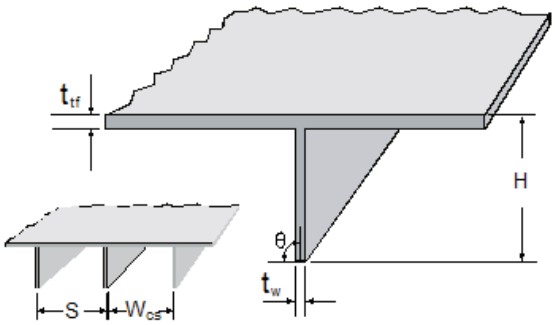


Fig. 7.1.6, Metallic Blade Stiffened Wing Skin, Design-to Nxy Load,(Controlling Load Case)

7.2 *Wing Skin Panel Dimensions*



Object	Material
Top Facesheet	Al 2219
Stiffener	Al 7075

~32" Rib Spacing Integral Blade Stiffened Panel, Wing Skin Component Dimensions

Panel Type	Top Facesheet Thickness t_{tf}	Web Thickness t_w	Panel Height H	Stiffener Spacing S	Skin Panel Unit Weight	Wing Skin Area	Wing Skin Total Weight
	(in)	(in)	(in)	(in)	(lb / ft^2)	(ft^2)	(lb)
Blade Stiffened	0.1-0.24	0.1-0.49	1.6-2.8	5.0-8.0	3.2	507.5	1624

Fig. 7.2.1, Integral Blade Stiffened Panel Wing Skin, Optimum Panel Dimensions

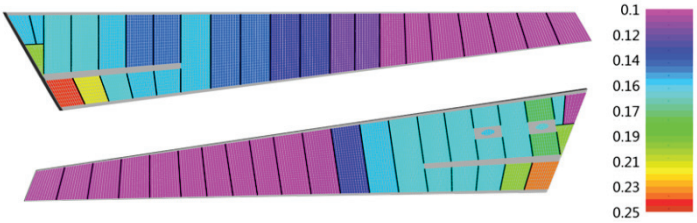


Fig. 7.2.1, Metallic Blade Stiffened Wing Skin, Top Facesheet Thickness

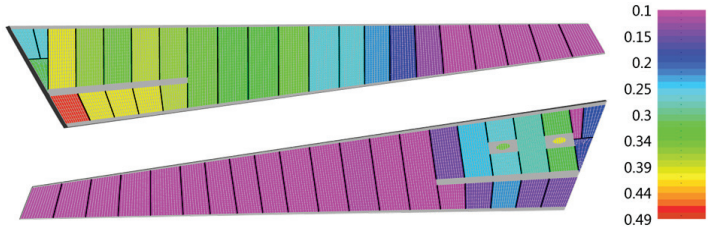


Fig. 7.2.2 Metallic Blade Stiffened Wing Skin, Stiffener Web Thickness

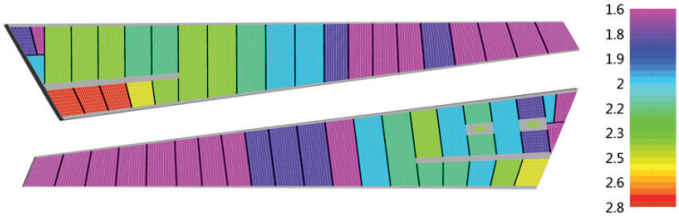


Fig. 7.2.3, Metallic Blade Stiffened Wing Skin, Total Panel Height

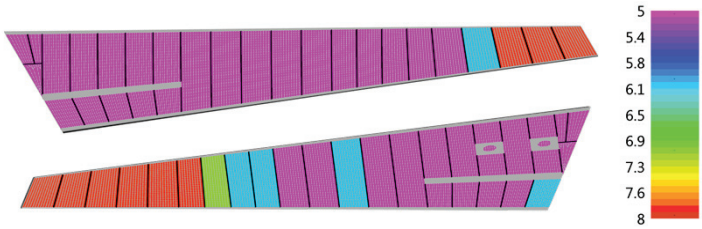


Fig. 7.2.4, Metallic Blade Stiffened Wing Skin, Stiffener Spacing

7.3 *Margin of Safety and Unit Weight Distribution*

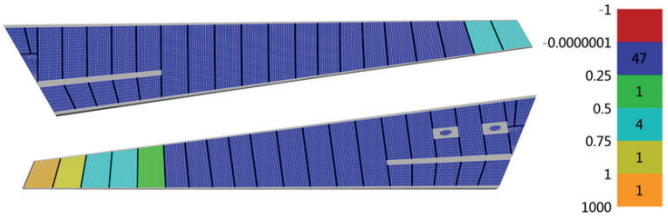


Fig. 7.3.1, Metallic Blade Stiffened Wing Skin, Margins of Safety

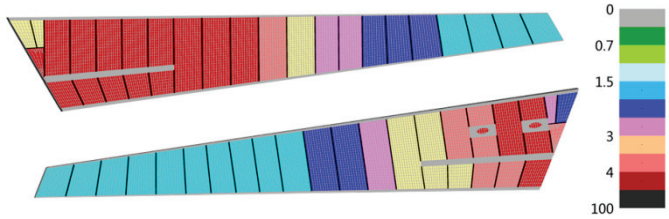


Fig. 7.3.2, Metallic Blade Stiffened Wing Skin, Unit Weight

7.4 *Controlling Load Cases and Failure Analysis*

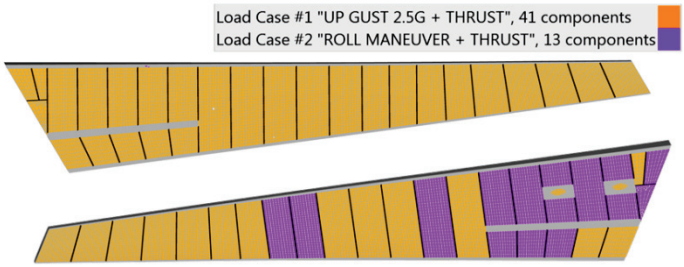


Fig. 7.4.1, Metallic Blade Stiffened Wing Skin, Controlling Load Case

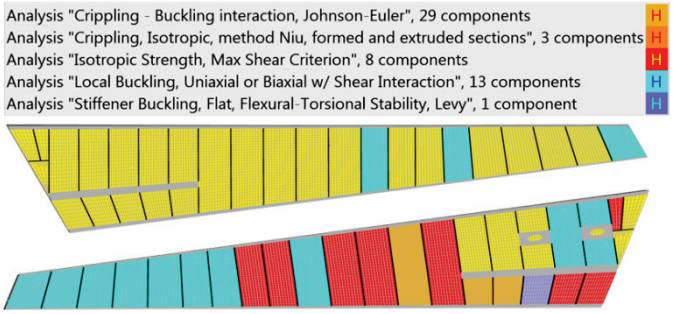
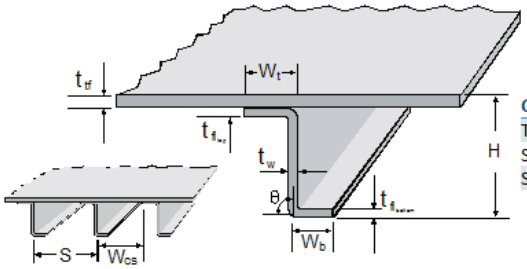


Fig. 7.4.2 Metallic Blade Stiffened Wing Skin, Controlling Failure Analysis

7.5 *Spar and Rib Dimensions*

The wing spar and ribs are Bonded Metallic Tee Stiffened panel concept. These designs are frozen and carried throughout the next 2 optimization levels.



Object	Thickness (in)	Material
Top Facesheet	0.1-0.29	Al 2219 Sheet
Stiffener Flange	Linked to Web	Al 7075 Extruded
Stiffener Web	0.05-0.11	Al 7075 Extruded

Bonded Zee Stiffened Panel, Internal Sub-Structure Component Dimensions

Panel Type	Facesheet Thickness	Web Thickness	Top Flange Thickness	Bottom Flange Thickness	Top Flange Width	Bottom Flange Width	Panel Height	Stiffener Spacing	Panel Unit Weight	Area	Total Weight
	t^{tf}	t^{w}	t^{fl}_{top}	t^{fl}_{bottom}	W^{fl}	W^{b}	H	S			
	(in)	(in)	(in)	(in)	(in)	(in)	(in)	(in)	(lb / ft^2)	(ft^2)	(lb)
Zee Stiffened	0.1-0.29	0.05-0.11	0.05-0.11	0.05-0.11	1.0-2.0	0.75-1.0	1.5-2.0	5.0-8.0	2.85	310	883.5

Fig. 7.5.1, Bonded Zee Stiffened Panel Internal Sub Structure, Optimum Panel Dimensions

7.6 *Spar Cap Dimensions*

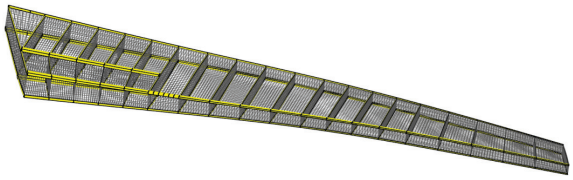
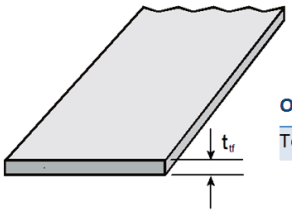


Fig. 7.6.1, Spar Caps Modeled as Solid Laminate Padups



Object	Thickness (in)	Material
Top Facesheet	0.1-0.29	Al 2219 Sheet

Solid Metallic Sheet Panel, Spar Cap Component Dimensions

Panel Type	Facesheet Thickness t^{tf} (in)	Panel Unit Weight (lb / ft^2)	Area (ft^2)	Total Weight (lb)
Solid Metallic Sheet	0.2	2.97	79.2	235

7.7 *Skin Cover to Internal Sub Structure Joints*

When joining the internal substructure to the wing skin, the tension attachment of wing skins to internal ribs and spars should be considered. The shear attachment of the rib to the spar is also a controlling design consideration. Usually a combined pull-off and shear load is present in these connection joints. HyperSizer is capable of quantifying margins of safety based on pull-off and shear allowables derived from test. The resulting element based margins of safety are illustrated on the figure below.

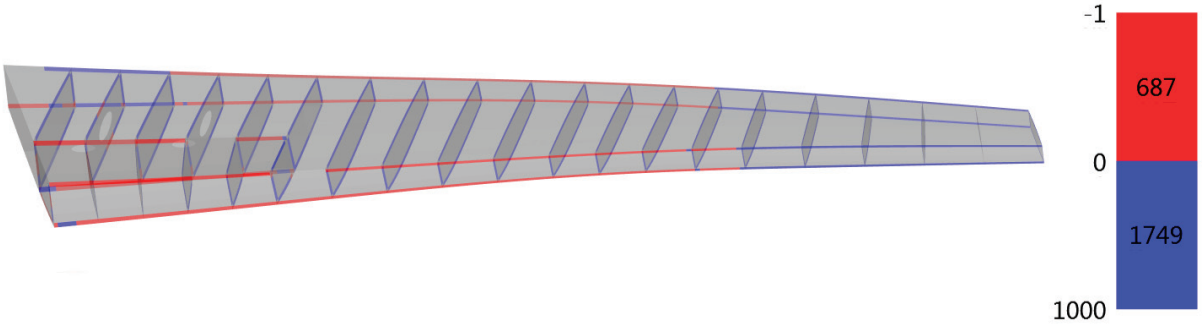


Fig. 7.7.1, FEM Joints Analyzed in HyperSizer Based on Pull-off and Shear allowables determined from Test. Determined that Current Joint designs are not Strong enough and require further optimization

8 Level 2 WML: A trade study weight estimate for the wing skin comparing four panel concepts

For this analysis the designs focuses on the wing skin, however the design space is expanded to include many panel concepts. Three primary panel concepts are considered for the wing skin, and each concept is optimized to find the lightest weight combination of cross sectional dimensions, materials and layups. One metallic concepts, (1) bonded Zee stiffened panels and two composite panel concepts, (2) bonded Hat stiffened and (3) bonded Tee stiffened are studied.

For preliminary design, the most robust failure methods are used to determine margins of safety for each candidate and find the lightest weight panel design. Since these analyses methods will also be used in the detailed panel analyses, there are less unanticipated failure modes to disrupt the analysis as the wing design matures. Therefore all potential failure modes are evaluated early and to a high degree of accuracy in the level 2 optimization studies.

8.1 *Metallic Bonded Zee Stiffened Panel*

8.1.1 Panel Dimensions

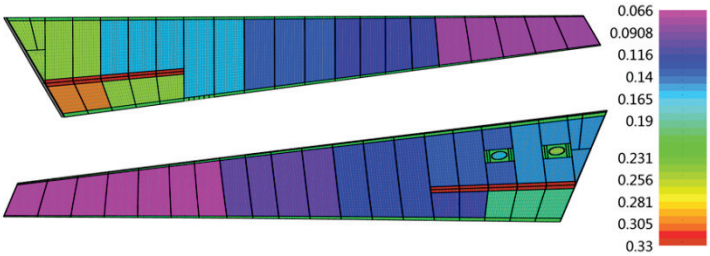


Fig. 8.1.1, Bonded Metallic Zee Stiffened Wing Skin, Top Facesheet Thickness

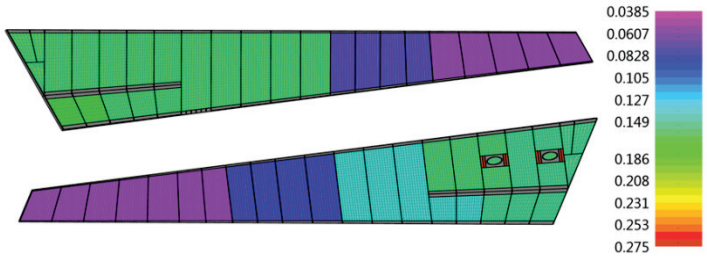


Fig. 8.1.2 Bonded Metallic Zee Stiffened Wing Skin, Stiffener Web Thickness

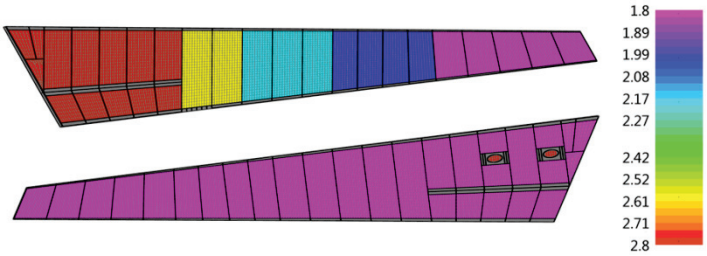


Fig. 8.1.3, Bonded Metallic Zee Stiffened Wing Skin, Total Panel Height

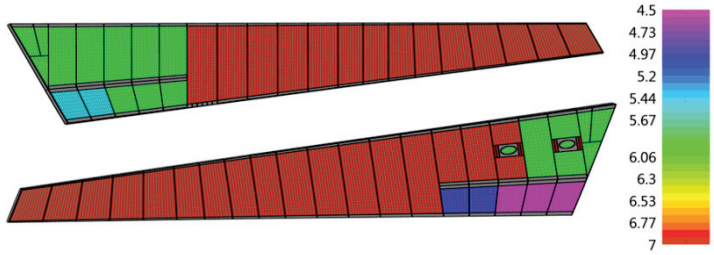


Fig. 8.1.4, Bonded Metallic Zee Stiffened Wing Skin, Stiffener Spacing

8.1.2 Margins of Safety, Controlling Failure Methods and Unit Weight

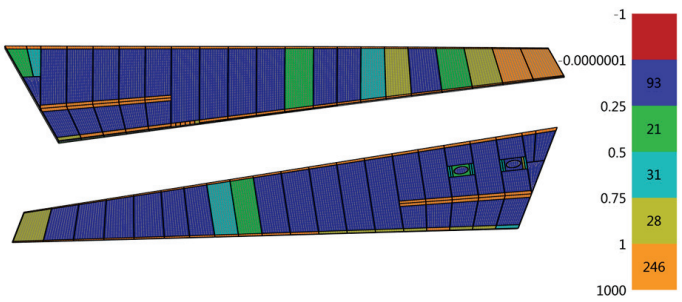


Fig. 8.1.5, Bonded Metallic Zee Stiffened Wing Skin, Margins of Safety

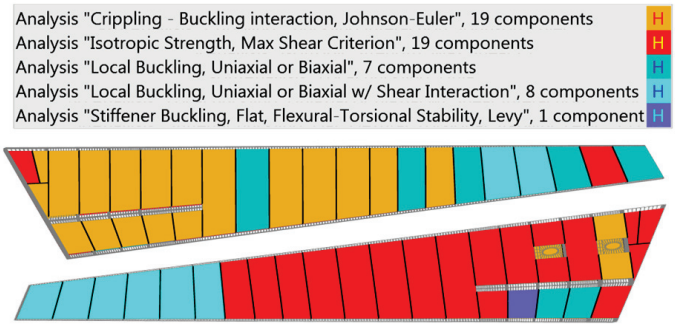


Fig. 8.1.6 Bonded Metallic Zee Stiffened Wing Skin, Controlling failure analysis

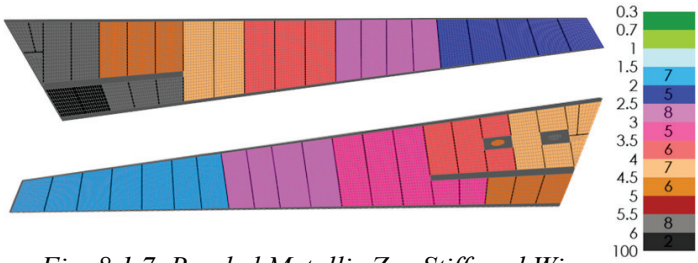


Fig. 8.1.7, Bonded Metallic Zee Stiffened Wing Skin, Panel Unit Weight

8.2 *Composite Bonded Tee Stiffened Panel*

Tee stiffened panels are used in wing skin structures for ease of manufacturing and low volume consumption in wing skin. Since stiffened panels have open section stiffeners, they provide better means for inspection and repair. The stiffeners provide high axial stiffness (A11) to the panel and increases the axial strength. However the open stiffener provides little overall torsional stiffness (D33) to the panel. The bonded Tee concept is less efficient at preventing panel buckling than concepts with a lower flange to increase the overall bending stiffness (D11) of the panel.

8.2.1 Panel Dimensions

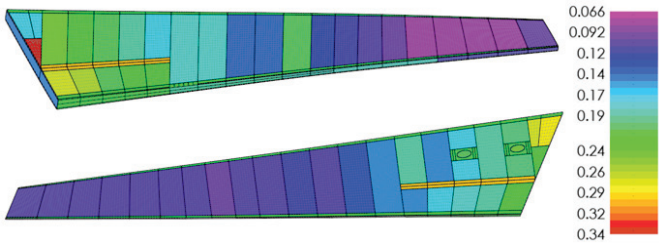


Fig. 8.2.1, Composite Tee Stiffened Wing Skin, Top Facesheet Thickness (range 12 - 62 Plies)

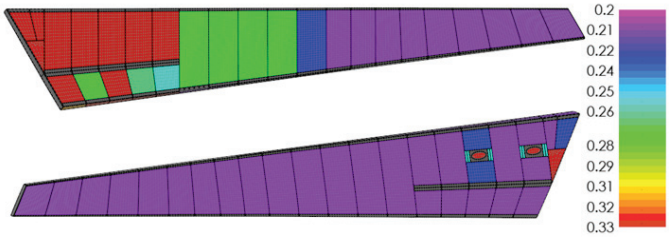


Fig. 8.2.2, Composite Tee Stiffened Wing Skin, Stiffener Web Thickness (range 38 - 60 Plies)

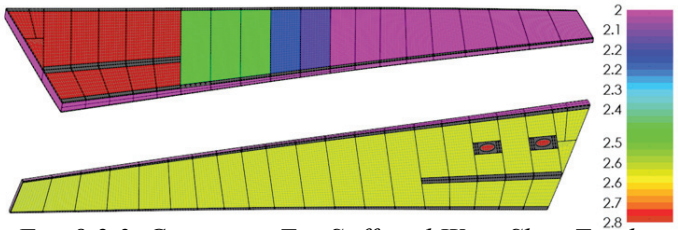


Fig. 8.2.3, Composite Tee Stiffened Wing Skin, Total Panel Height

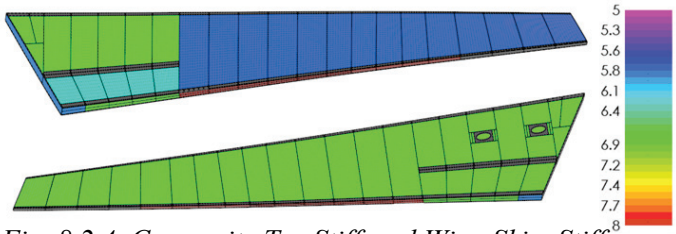


Fig. 8.2.4, Composite Tee Stiffened Wing Skin, Stiffener Spacing

8.2.2 Margins of Safety, Controlling Failure Methods and Unit Weight

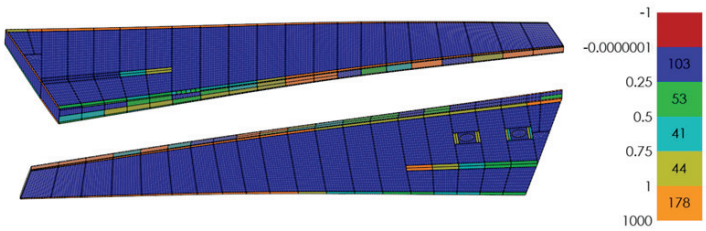


Fig. 8.2.5, Bonded Tee Stiffened Wing Skin, Margins of Safety

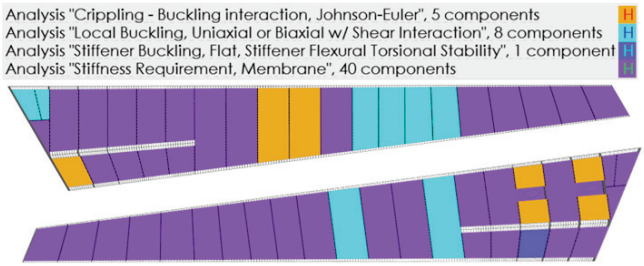


Fig. 8.2.6 Bonded Tee Stiffened Wing Skin, Controlling failure analysis

Fig. 8.2.7, Bonded Tee Stiffened Wing Skin, Panel Unit Weight

8.3 *Bonded Hat Stiffened Panel*

Hat stiffened panels are used in wing skin structures because hat stiffened panels have a closed section stiffener that provides high axial membrane stiffness (A11) and high axial bending stiffness (D11). Since the hat stiffener is a closed section, the GJ of the cross section is higher than open-section stiffeners like I's, or Tee's. The closed cell provides torsional rigidity and provides the panel with a higher bending-twisting stiffness (D33). The D33 stiffness is an influential term in the buckling stability. However the transverse bending stiffness (D22) is much lower for all stiffened panels than for a sandwich panel. For this reason ribs are required for buckling stability

8.3.1 Optimum Panel Dimensions

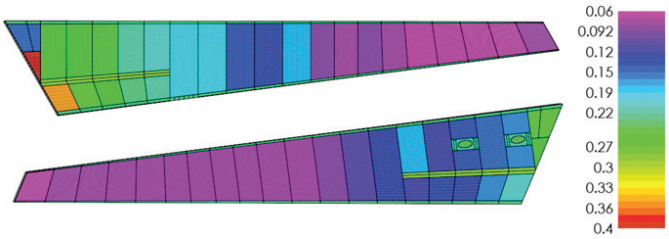


Fig. 8.3.1, Composite Hat Stiffened Wing Skin, Top Facesheet Thickness (range 11 - 72 Plies)

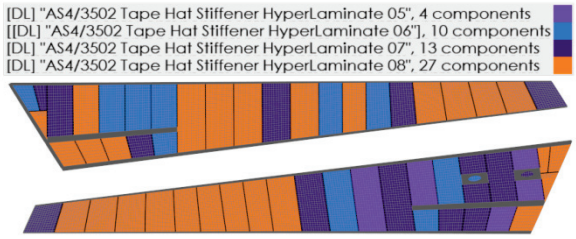


Fig. 8.3.2, Composite Hat Stiffened Wing Skin, Stiffener Web HyperLaminate (Ref Appendix X)

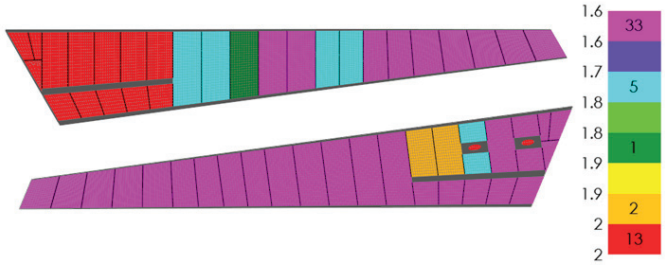


Fig. 8.3.3, Composite Hat Stiffened Wing Skin, Total Panel Height

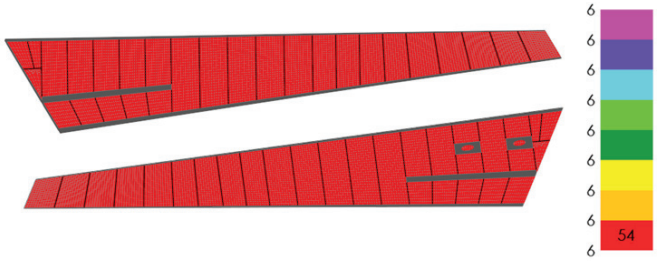


Fig. 8.3.4, Composite Hat Stiffened Wing Skin, Stiffener Spacing

8.3.2 Margins of Safety, Controlling Failure Methods and Unit Weight

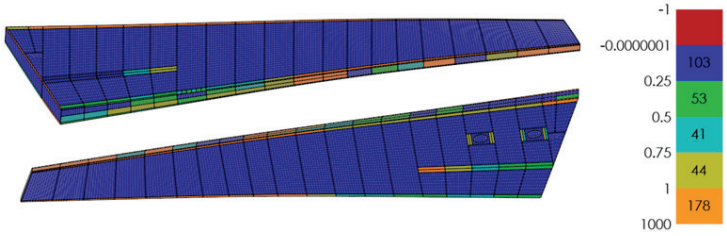


Fig. 8.3.5, Bonded Hat Stiffened Wing Skin, Margins of Safety

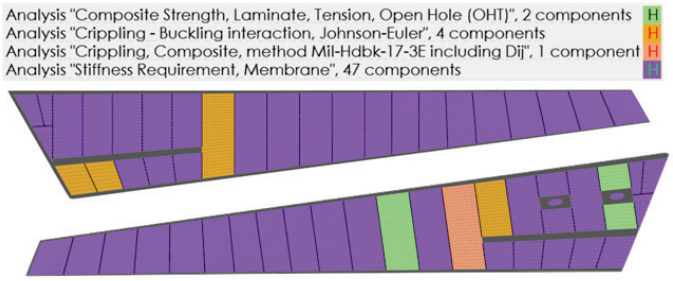


Fig. 8.3.6 Bonded Hat Stiffened Wing Skin, Controlling failure analysis

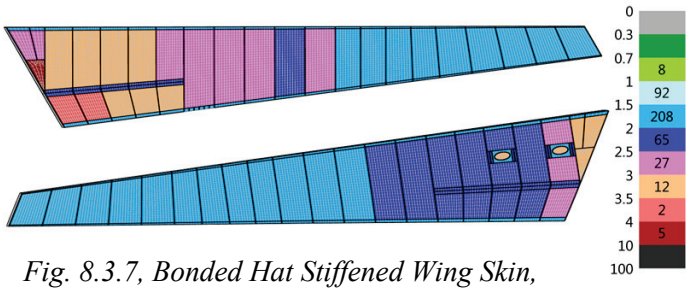


Fig. 8.3.7, Bonded Hat Stiffened Wing Skin, Panel Unit Weight

8.4 *Estimated Weight Savings using Composite Materials*

8.4.1 Low WML Weight Comparison

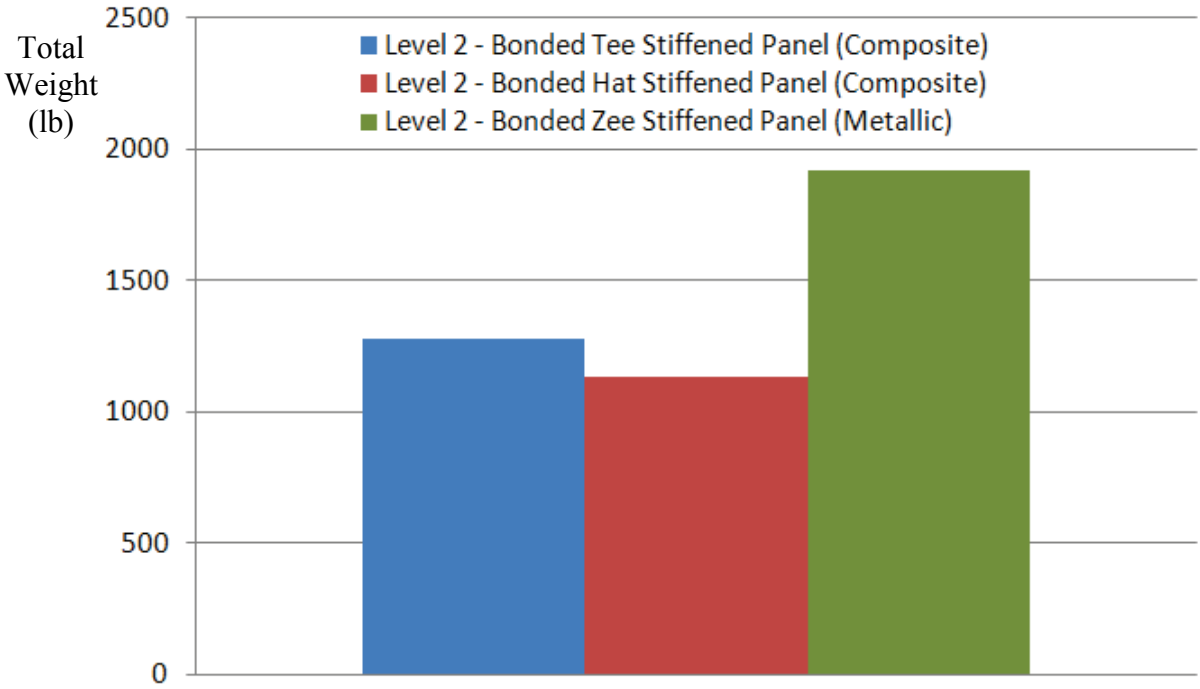


Fig. 8.4.1,Total Weight Comparison Chart, Composite Concepts are approximately 33% lighter than metallic Concept

9 Level 3 Intermediate WML: A detailed trade study weight estimate for a single panel bay skin comparing Three panel concepts

9.1.1 Sizing optimization refinement for Three Panel Concepts

A representative area of the wing is chosen for further trade studies. At this location each of the three primary panel concepts are evaluated and detailed margins of safety are obtained using HyperSizer.

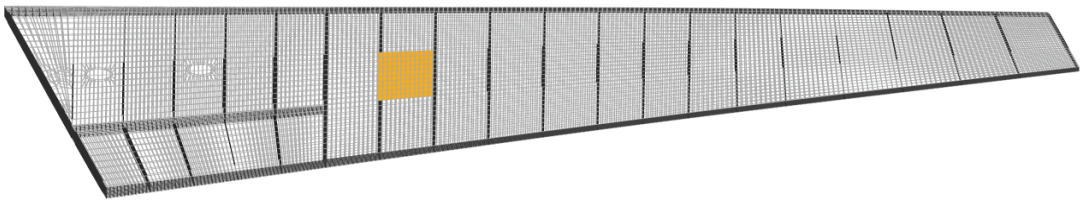


Fig. 9.1.1, Representative Area for Further Evaluation of Four Primary Panel Concepts. Design-to Loads, Nx = -6025lb/in, Ny = -100lb/in, Nxy = -420lb/in (limit)

9.1.2 Zeroing out FEA Computed Bending Moment Loads

Each panel design will generate unique computed moments to enforce uniform strain with minimal curvature. HyperSizer includes the ability to compute the correct internal moments required to enforce uniform end shortening for each stiffened panel design. This allows us to use the same internal loads to achieve an apples to apples comparison of each panel concept.

9.1.3 A Focused Weight Comparison

SECTION REMOVED

Note high weight savings opportunity with composite, especially hat stiffened panel concept.

9.2 *Metallic Bonded Zee Stiffened Panel*

SECTION REMOVED

9.3 *Composite Bonded Tee Stiffened Panel*

SECTION REMOVED

9.4 *Composite Bonded Hat Stiffened Panel*

SECTION REMOVED

10 Increasing the WML

10.1 *Refining the Component Definition*

The component definition is redefined based on the internal loads. Each panel bay is split into 3 components along the chord length.

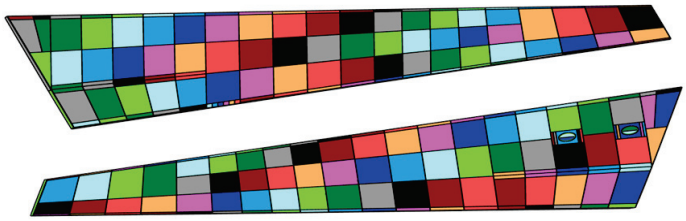


Fig. 10.1.1, Detailed Analysis Model Component Definition

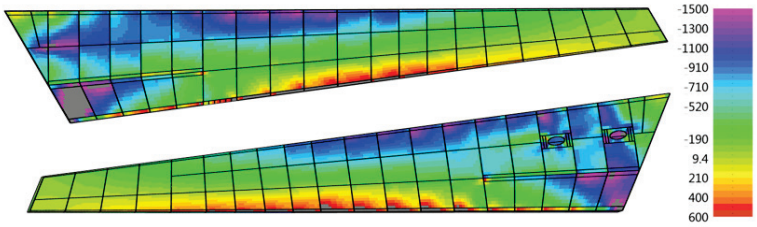


Fig. 10.1.2 Nxy Load, Load Set 103

10.2 *Activating Detailed Failure Analysis*

HyperSizer includes many failure analysis required for aircraft certification. These analysis include, Beam Column, offline fuel pressure, bonded joint analysis, local post buckling etc. By analyzing the wing panels to generate detailed margins of safety for these analysis and printing detailed stress reports, the WML of these designs will increase.

10.3 *Offline Fuel Pressure*

HyperSizer calculates panel bending moments caused by offline fuel pressure on the wing skin and internal sub-structure. The superposition of these moments is handled automatically by HyperSizer and the strength loads are updated to account for the panel pressure. Since panel pressure applied to curved panels imposes hoop load, this method is only valid for flat panels.

Also local bending moments in the skin between the stiffeners are important for the strength and bonded joint analysis. The local bending moments in the facesheet between stiffeners causes a localized deformation because the transverse bending stiffness of the facesheet is very small.

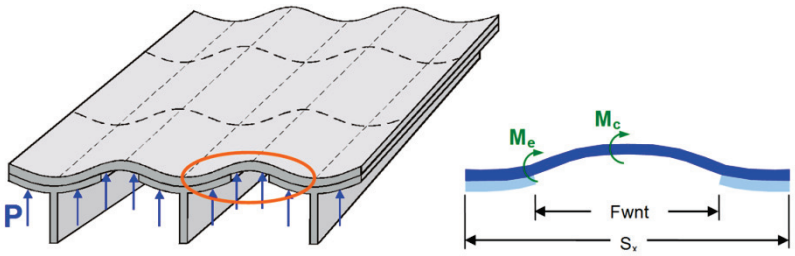


Fig.10.3.1, Local Response of Tee Stiffened Panels due to Panel Pressure

This state of localized curvatures and stresses are passed to HyperSizer's bonded joint analysis.

10.4 *Beam Column Analysis*

For axially loaded stiffened panel structures with off-line pressure loads, Beam-Column effects are considered. Beam-Column analysis is often referred to as a unique failure mode margin of safety check. However, it is actually an increase in loads due to secondary displacement effects. These loads are superimposed moments and transverse shear loads caused by bending deformation from panel pressure.

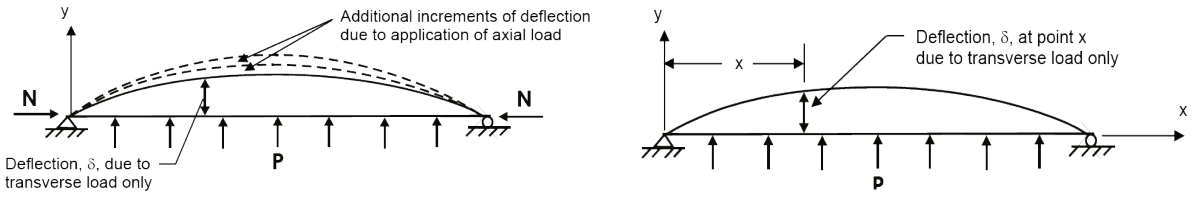


Fig.10.4.1, Left image, Initial Imperfection caused by internal fuel pressure. Right image, additional load causing more static deformation

The HyperSizer Beam-Column analysis computes a geometrically non-linear moment caused by the combined effect of in-plane compression and lateral deflection from panel pressure. The Beam -Column moment is superimposed with the FEA computed panel moment. This increased moment changes the distribution of panel or beam cross section stresses and therefore requires updated material strength and local buckling margin of safety checks.

10.5 ***Bonded Skin to Flange Joint***

HyperSizer includes the ability to size stiffened panels including the effect of interlaminar stresses in the stiffener-to-facesheet bondline. The panel level loads are automatically applied to the stiffened panel, "bonded doubler" joint with consistent internal forces and boundary conditions. By transforming the internal loads derived for the bonded combo, the bondline is examined to determine the effect of interlaminar stresses in the stiffener to facesheet bondline. The two primary bonded joint failure methods used in this analysis are shown in figure 10.5.2. below:

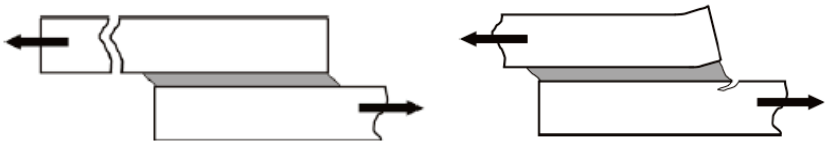


Fig.10.5.1, (Left) Adherend Fracture, Far Field Sectional Failure, (Right) Adherend Composite Interlaminar Delamination

Sectional failure is a common failure mode in composite bonded joints. In general, maximum in-plane stress or strain criterion is used for this type of failure. Interlaminar delamination is a typical failure mode for adhesive bonded joints. This failure is caused from the weakness of the composite adherends in the through thickness direction. The method for determining failure in the bonded doubler is illustrated in figure 10.5.2.

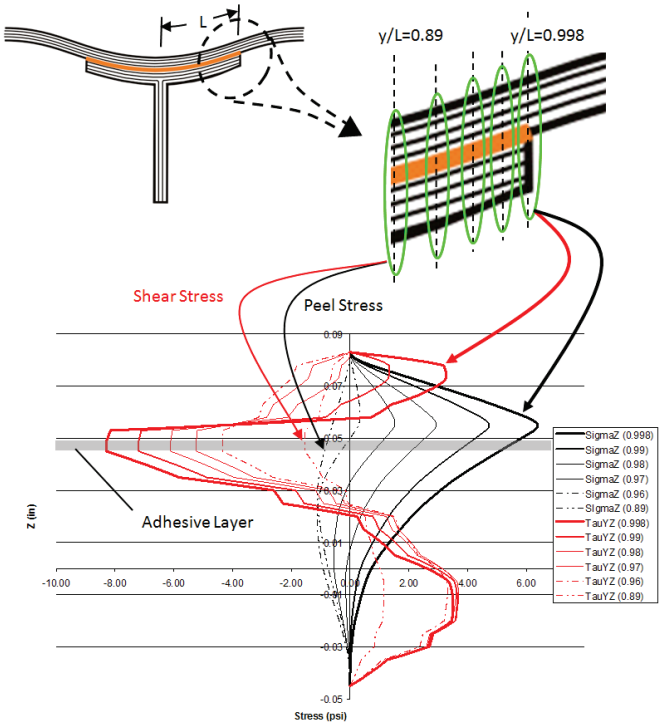


Fig.10.5.2, HyperSizer Bonded Joint Analysis Summary, out of plane stresses computed at vertical cuts through the bonded doubler

In this process the in-plane outer fiber stresses are determined for each ply using classical lamination theory. The out-of plane interlaminar shear and peel stresses are computed by transforming these loads. This transformation is done along many hundreds of vertical cuts through the thickness of the bonded flange to skin. At a user-defined characteristic distance, the bonded joint strength analysis is performed at the inner and outer fiber locations, the purpose of which is to determine the worst location of critical stresses that may cause failure. The lowest joint strength margin of safety is used for optimization.

10.6 ***Stiffened Panel Post Buckling***

Due to the high axial stiffness provided by the stiffener, stiffened panels have high buckling stability and are efficient at supporting compression loading. However the lowest failure mode of a stiffened panel with high stiffener spacing is usually local buckling of the skin between stiffeners. Local post buckling is based on the assumption that after the skin local buckles, the stiffened panel can continue to support more load.

The unbuckled portion of the skin (effective width) is included in the stiffener's cross section properties for additional loading past the point of bifurcation. The reduction of effective width due to additional loading is illustrated in figure 10.6.1.

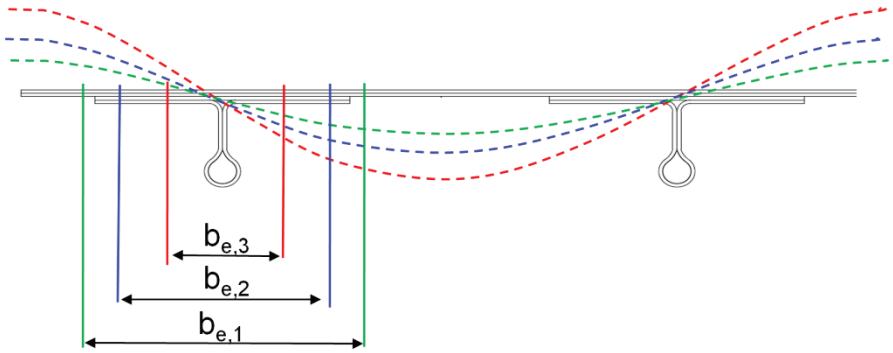


Fig.10.6.1, Local Post Buckling Bifurcation Process

This figure illustrates the sequence of local buckling. First an initial bifurcation local buckling mode occurs denoted by the green line. Then the additional load causes the 2nd mode shape amplitude to be greater, blue line. Finally, as the full local post buckling strength is realized and the mode shape becomes its largest, red line. As the load is increased, the buckling mode shape becomes larger, and the effective width of the remaining stable skin becomes narrower.

As the skin local buckles the effective width is calculated and a new object load distribution is determined. For the wing skin, which is buckling critical, the spacing span is allowed to local buckle at limit load. As a result, the stiffener spacing increases by as much as 20%. The load redistribution causes other failures in the panel such as material strength, crippling and flexural torsional buckling.

10.7 ***Composite Fabrication Methods***

HyperSizer provides the ability to link any number of cross sectional dimensions, such as stiffener material, spacing and height variables, across adjoining structural components. This capability is used on the stiffened wing skins by enforcing a similar stiffener layup and geometry spanning the root to tip of the wing. This is valuable because few mandrel shapes are needed to fabricate the wing section which decreases the manufacturing complexity.

A typical hat stiffened panel layup scheme is shown in figure X.X. From this figure we see many 0 degree fibers are applied to the L4 (stiffener crown) and are dropped in the crown to web transition region.

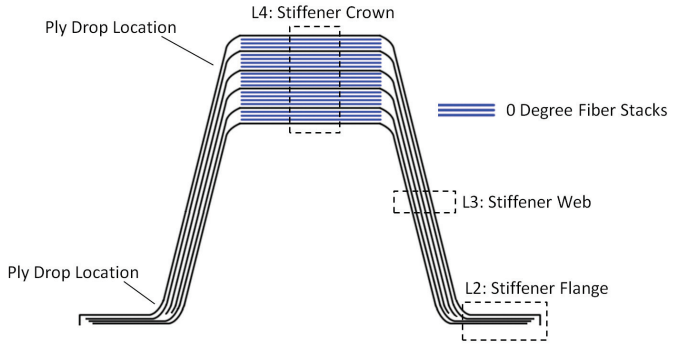


Figure 10.7.2: Hat Stiffener Layup Sequence, Ply drops allowed from Stiffener Crown to Stiffener Web. Web and Flange are 100% 45 Degree Fibers

Hat stiffened panels achieve panel buckling stability primarily by adding 0° plies in the crown and increasing hat height to obtain a high EI. The hat skin has a higher % of 45° and 90° plies to provide material strength for hoop loads and skin local buckling stability. The web is all 45° plies for laminate buckling stability. In fact, adding 0° plies to the web is detrimental in that it will cause the web to pick up more axial load and buckle sooner.

10.8 ***Ply Drop Analysis***

All weights reported have assumed the facesheets of each panel bay may optimize independently of the adjacent panel bay. In reality the adjacent panels will require a similar-manufacturable layup where ply sequences and laminate thicknesses are based on the adjacent skin component. To maintain symmetric and balanced layups, plies are to be dropped from the mid-plane of the laminate in an alternating order. The length of the joint is determined based on a 20-1 drop ratio.

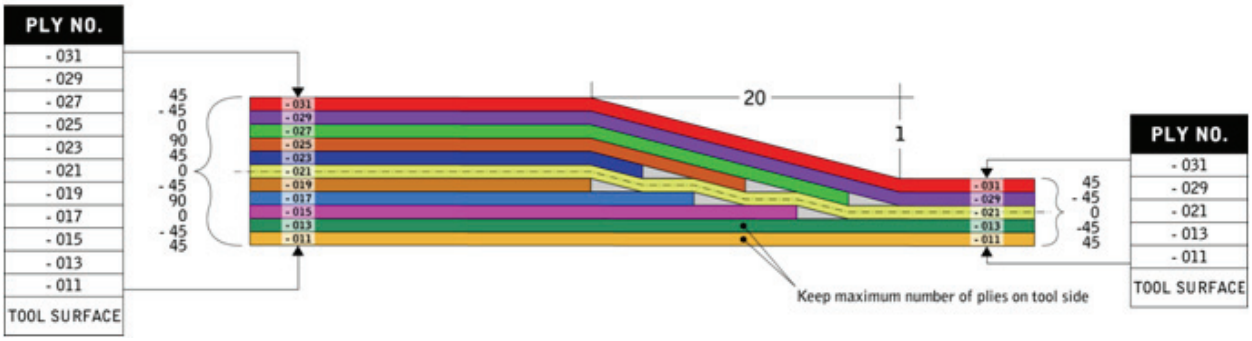


Fig 10.8.1, Composite Ply Drop Boundary, Plies dropped from Laminate Mid-plane and enforced to maintain a balanced-symmetric layup, 20-1 slope maintained

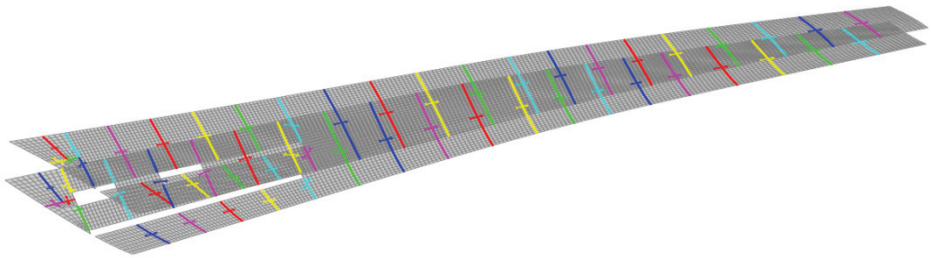


Fig.10.8.2, Composite Ply Drop Boundaries for Top and Bottom Wing Skins

The ply drop boundaries for the top and bottom wing skins are show in figure 10.8.2. These define the component boundaries where the ply drops/adds will be quantified. Weight trade studies are performed on the wing skin to reduce the plies dropped and added across component boundaries and determine the optimum ply orientation for both weight and manufacturing considerations.

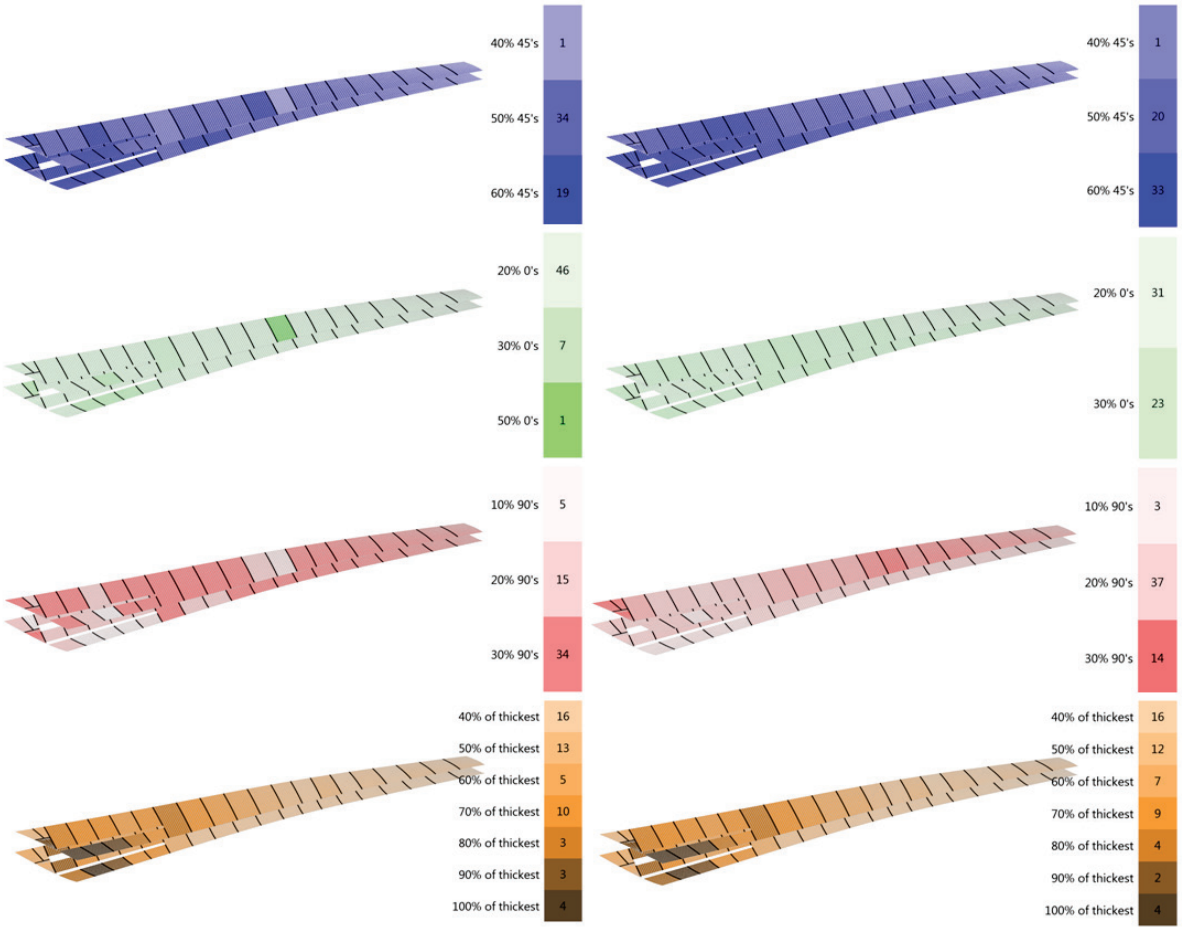


Fig.10.8.3, Composite Fiber Orientation, Blue- Percentage of 45 degree Fibers, Green- Percentage of 0 degree Fibers, Red- Percentage of 90 degree Fibers, Brown- Percent Thickness of Thickest Facesheet. Left Images are before plydrop analysis and right images are after plydrop analysis

By decreasing the number of Plydrops, the ply orientation gradients become smoother across the transition from root to tip. After the ply compatibility analysis there are no longer components that stand out because there fiber orientations are much different than the adjacent components. The weight impact of reducing the total number of ply drops/adds is shown in table 10.8.1.

Table 10.8.1, Weight impact of Reducing Ply Drops/Adds in Wing Skin

	Ply Drops	**Ply Adds**	**Wing Skin Weight**
Lightest Weight Design	215	114	1182
Manufacturable Design	142	44	1200

It should be noted that since HyperSizer is capable of finding many optimum solutions that are evaluated for manufacturability, the weight impact of significantly reducing the plydrop/adds in the wing skin, as done for this model, is less than 1.5%.

11 Conclusions

Hat stiffened panels are a lighter weight solution for wing skin applications. The increased number of sizing variables provide more opportunities for weight savings but this makes the optimization of these concepts increasingly more difficult. To effectively optimize a stiffened panel an automated tool such as HyperSizer is needed to fully explore all the variables in the design space.

HyperSizer is a automated tool for composite optimization and analysis that should be used throughout the design process, not just in the preliminary design phase. The software incorporates many failure analysis required for aircraft certification. As a project matures new design criteria will be imposed. To ensure the weight impact of the design criteria is low, HyperSizer should be used to perform panel trades and generate many designs. Since multiple, equally performing optimum designs can be found using HyperSizer, if one particular design is later discovered to have difficulty in certification, alternate fall back designs are readily available and can easily be substituted. The ultimate objective is to find a robust design that is easily manufacturable and certifiable.

12 Acknowledgments

This material is based upon work fully funded mostly by internal Collier Research Corporation funding.

13 Trademarks

HyperSizer® is a registered trademark of Collier Research Corporation

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NX NASTRAN® is a registered trademark

NEI NASTRAN® is a registered trademark

14 References

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