# LOCAL DESIGN OPTIMIZATION FOR COMPOSITE TRANSPORT FUSELAGE CROWN PANELS[1]



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## ABSTRACT

Composite transport fuselage crown panel design and manufacturing plans were optimized to have projected cost and weight savings of 18% and 45%, respectively. These savings are close to those quoted as overall NASA ACT program goals. Three local optimization tasks were found to influence the cost and weight of fuselage crown panels. This paper summarizes the effect of each task and describes in detail the task associated with a design cost model.

Studies were performed to evaluate the relationship between manufacturing cost and design details. A design tool was developed to aid in these investigations. The development of the design tool included combining cost and performance constraints with a random search optimization algorithm. The resulting software was used in a series of optimization studies that evaluated the sensitivity of design variables, guidelines, criteria, and material selection on cost. The effect of blending adjacent design points in a full scale panel subjected to changing load distributions and local variations was shown to be important. Technical issues and directions for future work were identified.

## INTRODUCTION

Boeing is studying transport fuselage applications in the NASA/Boeing Advanced Technology Composite Aircraft Structures (ATCAS) program. The ATCAS design build team has adopted a two phase approach for minimizing structural cost and weight that includes global evaluation and local optimization (Refs. 1 and 2). During global evaluation, the cost and weight characteristics of several "design families" are quantified. One of the families is then selected for local optimization based on cost/weight merits and the potential for additional savings. To date, both global and local design phases have been completed for a 15 ft. by 31 ft. crown quadrant in the section directly behind the wing to body intersection of a 20 ft. diameter fuselage.

For the purpose of review, final results from the crown global evaluation studies performed in 1990 are shown in Figure 1. An intricately bonded skin/stringer/frame design (i.e., Family C) was selected by ATCAS for local optimization studies.

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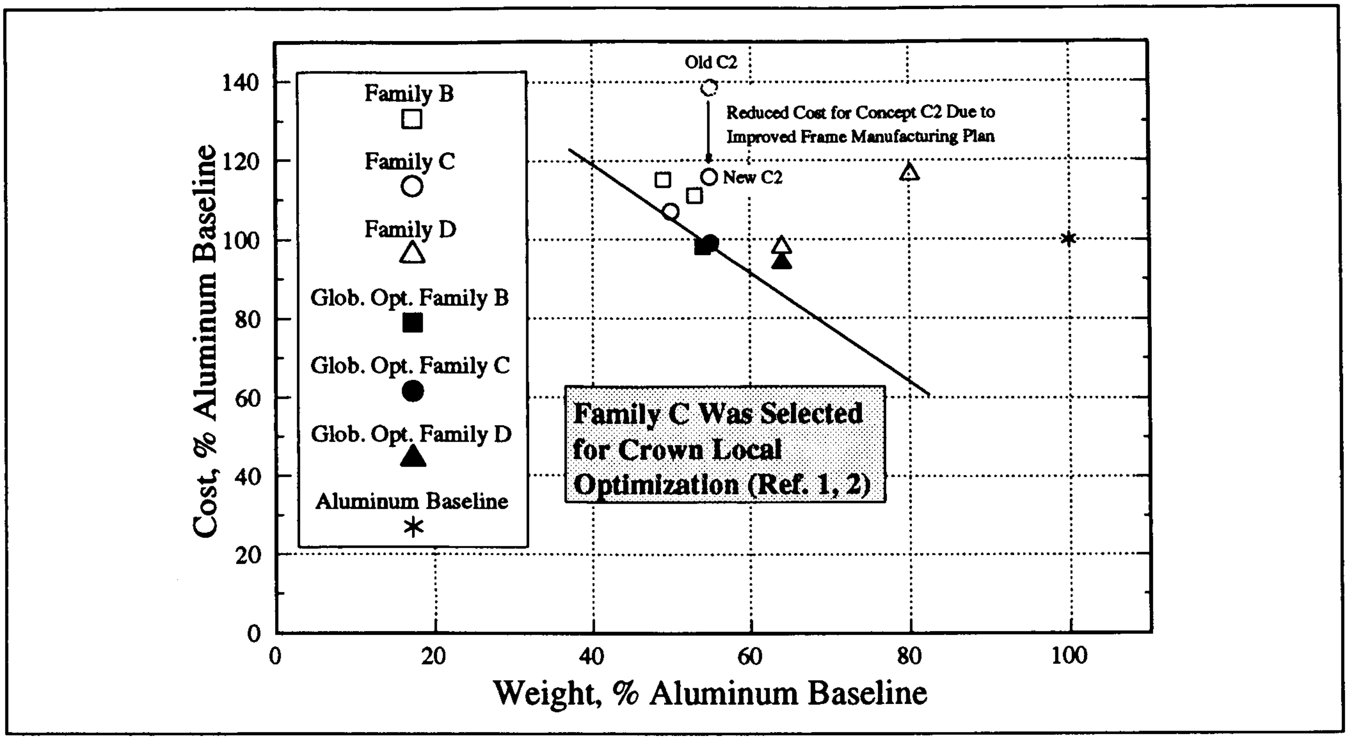


**Figure 1: Results of the ATCAS Global Crown Panel Evaluation (Refs. 1 and 2)**

The beginning of this paper summarizes how three tasks supporting local optimization of crown panels affected cost and weight. Two of the tasks are detailed in other papers appearing in this proceedings (Refs. 3 and 4). The third task, involving the development and application of a design tool for assessing the effects of design details on cost and weight will be described in this paper. Discussions will include (a) the steps to develop the design tool, (b) the sensitivity studies performed to identify the critical crown panel variables, and (c) the technique used to arrive at a final optimum crown panel design.

## ATCAS FUSELAGE CROWN STUDIES

Local optimization in the ATCAS program is essentially a more detailed study of a given design. The three tasks that support local optimization include:

1. perform tests for selected materials to augment the database on critical performance issues
2. develop design/cost analyses to be used to optimize design details for selected processes
3. perform fabrication trials and optimize manufacturing plans to improve process efficiency.

In general, the cost and weight of the design can either increase or decrease depending on results generated in task 1. Task 3 attacks cost centers by exploring possible improvements in manufacturing process steps. Task 2 attempts to minimize the cost and weight by evaluating the effects of design details. This task makes use of results from tasks 1 and 3.

Figure 2 summarizes how each local optimization task affected the manufacturing cost and weight of the ATCAS crown quadrant. The final crown design was found to have a projected cost and weight savings (relative to 1995 aluminum technology) of 18% and 45%, respectively. These savings are close to those quoted as overall NASA ACT program goals.

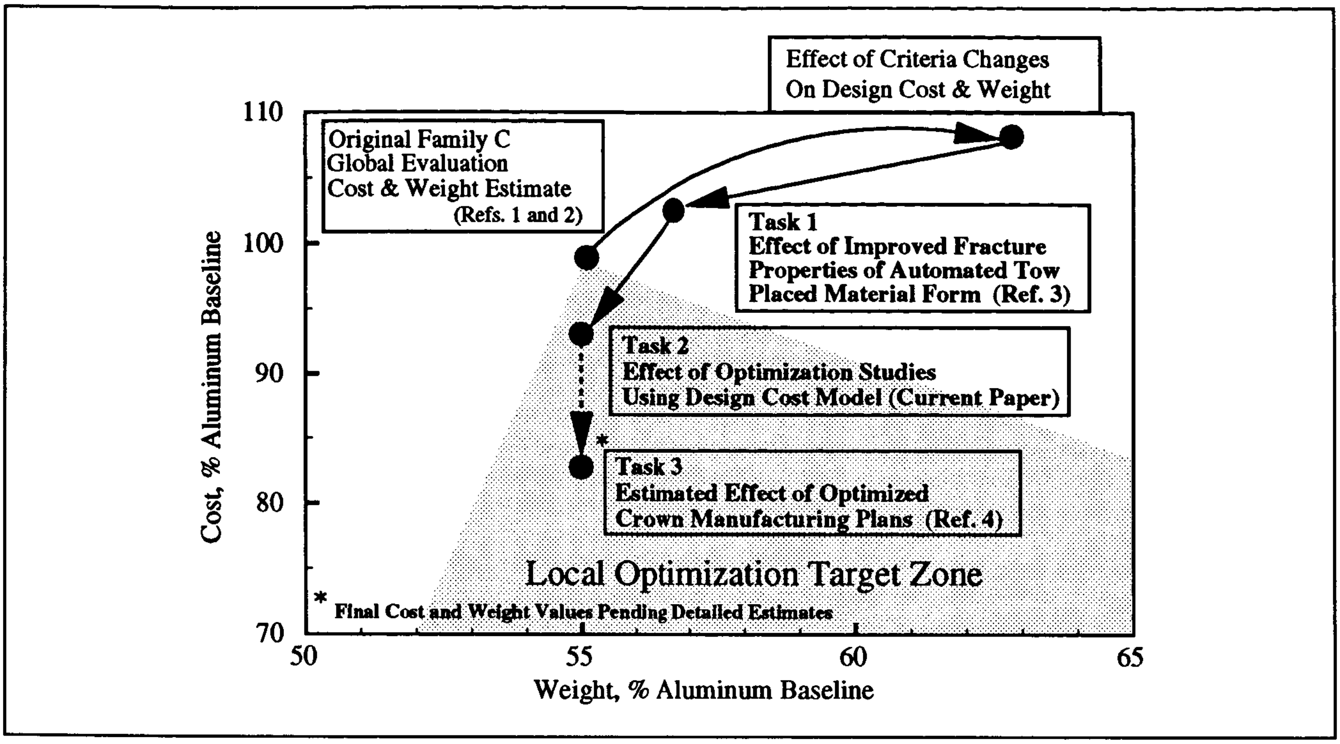


**Figure 2: Effects of the Criteria, Material Properties, Design Details, and Manufacturing Processes on ATCAS Crown Panel Local Optimization**

Referring to Figure 2, the cost and weight of the original design sized for global evaluation changed due to modifications in design criteria. Criteria were changed to include larger through penetration damage sizes, minimum stiffness requirements (axial and shear), and a minimum skin buckling load level. Initial global sizing efforts used a relatively small penetration for the failsafe damage condition, and had no minimum skin buckling or stiffness criteria imposed. After applying the additional criteria to obtain an acceptable design, both cost and weight were found to increase. Of the three criteria changes, a larger damage size was found to have the strongest effect on this initial shift in structural cost and weight.

Collection of tension fracture test data for candidate skin materials was the focus of task one for local crown optimization. Laminate fracture test results for the automated tow placed material form were found to be superior to the tape properties assumed during global evaluation. As shown in Figure 2, the improvement resulted in lower cost and weight due to a reduced skin gage. In this case, the generation of a tension fracture database was found to help reduce design cost and weight, essentially counteracting some of the effect of the design criteria for larger damage size. It was not possible to take full advantage of the improvements because other criteria, such as minimum stiffness and skin buckling constraints, were found to become design drivers as the skin gage decreased. The improved fracture properties and their effect on the design are discussed further in another paper included in these proceedings (Ref. 3).

The third task for crown local optimization considered changes in the manufacturing plans to reduce cost. As shown in Figure 2, the total effect of several changes was projected to decrease cost by approximately 10%. Cost centers that were attacked included the fabrication of skin, stringer, and frame elements, and panel cure. Modifications having the strongest impact on cost related to automation, reduced numbers of tools, elimination of processing steps, and deletion of unnecessary design details. Fabrication of curved braided frames and panel cure trials using soft tooling concepts provided supporting data for changes in the manufacturing plans. The changes in crown manufacturing plans and supporting data from process trials are discussed further in another paper included in these proceedings (Ref. 4).

The remainder of this paper will focus on task 2 of crown local optimization, namely the development and application of a design cost model for the Family C, intricately bonded, panel concept. A software design tool was developed to support this effort. The tool combined a random search optimization routine with software modules containing design/cost relationships, structural mechanics sizing tools, and design criteria. Analyses were performed with the tool to determine the cost drivers and design sensitivities. The overall effect of optimizing design details for the crown concept can be seen in Figure 2. As shown in the figure, task 2 efforts decreased the relative cost and weight of the crown panel design such that it is within a target zone identified at the start of local optimization (Ref. 1).

## DESIGN TOOL DEVELOPMENT

A computer program was developed to evaluate the effects of design details on cost and weight. The design tool combines three components: cost and performance constraints and a random search optimization algorithm. The optimization algorithm is capable of minimizing cost and weight objective functions in a global, discontinuous space. The cost constraint algorithm relates the manufacturing process costs to the detailed design variables. This algorithm provides for the ability to optimize for minimum cost. The performance constraint module accounts for load conditions, design criteria, material properties, and design guidelines. The three design tool components complement each other to insure structural integrity while optimizing for both cost and weight (Ref. 5).

### Optimization Routine

The design tool uses a sequential random search algorithm which globally searches the design space to find the optimum configuration.[2,3] The global nature of this algorithm is different from the more common gradient search methods in that it is not dependent on the initial starting point. Gradient search methods require multiple runs with varying starting points to ensure that an optimum design has been located. Figure 3 shows a schematic of the how the random search optimization method considers the entire design space. This approach is efficient for composite structures applications that include many variables and a design space having discontinuous functions. Since laminates contain an integer number of plies, an optimizer that is insensitive to discontinuous functions is a benefit.

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[2] Z. B. Zabinsky, D. L. Graesser, M. E. Tuttle, G. I. Kim, "Global Optimization of Composite Laminates Using Improved Hit and Run", Recent Advances in Global Optimization, edited by C. A. Floudas and P. M. Pardalos, Princeton University Press, to appear 1991.

[3] D. L. Graesser, Z. B. Zabinsky, M. E. Tuttle, G. I. Kim, "Designing Laminated Composites Using Random Search Techniques", Journal of Composite Structures, to appear 1991.

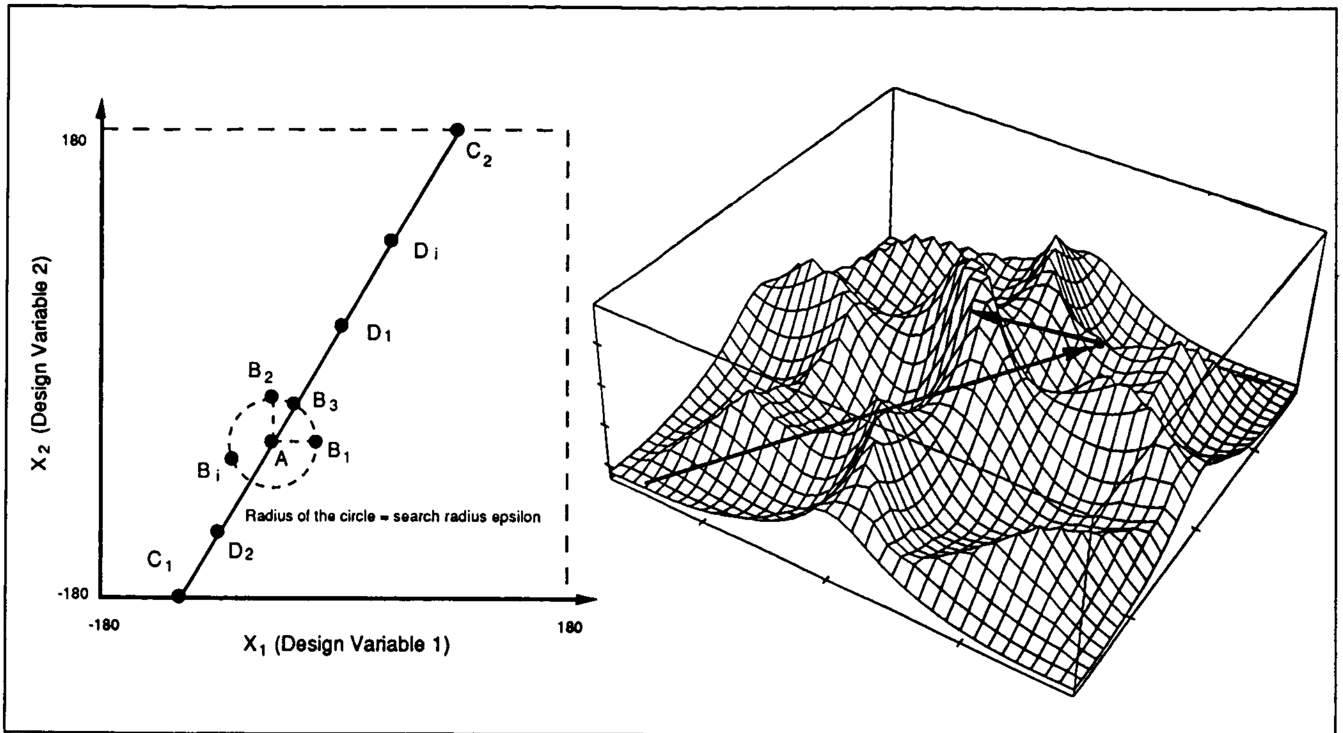


**Figure 3: Random Search Method Schematic Diagram**

The results given in this paper utilize the random search algorithm to determine the optimum design. In practice, however, the global nature of the random search method becomes computationally inefficient because it continues to search a large design space as the optimum solution is approached. Current work is considering the combination of the random search algorithm with an efficient gradient based optimization code to search the entire design space and then converge to the solution more effectively. This work supports the larger optimization problems envisioned for a tool which blends the design for multiple load points in a large aircraft structure. The framework for this advanced development is discussed in further detail in Reference 6.

## Cost Constraints

Design/manufacturing cost relationships were developed in order to optimize crown panels for cost. These relationships were added to the optimization tool as cost constraints. They were based on data collected during the crown global evaluation process (Refs. 1 and 7), when a comprehensive manufacturing plan was compiled for each design to support a detailed cost estimate. Focussing on the design concept chosen for local optimization, individual cost drivers were determined from the detailed cost breakdown. This was accomplished by evaluating the detailed cost steps in terms of how they relate to the design details. By considering how each step may be affected by variables relating to the design, relationships were determined and the costs were normalized to the baseline design. Using this approach, any variance in a given design detail can be accounted for in the part cost.

Figure 4 shows an example of how design/manufacturing cost relationships were derived from detailed estimating data. The figure includes a list of the processes considered in the crown panel development, a list of the design functions used in the cost breakdown, and an example of how the functions were assigned to each detailed process step. As shown in Figure 4, each detailed process step was coupled with the design function that directly affects the cost. If none of the design functions were perceived to have a direct effect, that individual step was assumed to be constant. Following this analysis, all terms were summed to obtain a single equation for total crown panel costs. A representation of this cost equation for the skin/stiffener/frame cobonded crown panel assembly is shown in Figure 5. The coefficients in this equation are valid only for this particular design family, panel size, and associated manufacturing processes. Using this equation, small variations in the design details from the baseline design could be evaluated from a cost standpoint and the major cost drivers exploited. Any major design differences from the global design or any process changes are likely to result in changes to the equation coefficients. A more generalized cost evaluation analysis is envisioned for future work to evaluate different types of structures. Again, the framework for this is discussed in Reference 6.

Categories in Cost Estimates:

Stringer fabrication
Skin fabrication
Panel Co-cure/Co-bond process
Frame fabrication
Circumferential splice straps
Stringer splice fabrication
Installation to barrel section
Body join

**Design Functions:**

f1 = Constant
f2 = Total stringer weight
f3 = Number of stringers
f4 = Total stringer material cost
f5 = Total skin weight
f6 = Total skin material cost
f7 = Number of frames
f8 = Number of stringer * frames
f9 = Stringer Trim Parameters
f10 = Skin Trim Parameters

Example of Stringer Fab Breakdown:

| Stringer fabrication | recurring | | non-recurring | | | Related Design Function |
|---|---|---|---|---|---|---|
| | rec hrs | mat'l $ | tooling | | planning | |
| | | | hrs | mat'l $ | hrs | |
| - prepare ATLM | 0.05 | 0.00 | -- | -- | -- | Constant (f1) |
| - layup material | 1.55 | 1000 | -- | -- | -- | Total stringer wt (f2) |
| - identify part number | 0.03 | 0.00 | -- | -- | -- | No. of stringers (f3) |
| - prepare for trim | -- | -- | -- | -- | -- | Constant (f1) |
| - trim | -- | -- | -- | -- | -- | Str. Trim Param. (f9) |
| - etc ... | -- | -- | -- | -- | -- | etc ... |

**Figure 4: Design Variables and Their Relationship to the Manufacturing Cost**

## Performance Constraints

The criteria used to design a composite fuselage crown panel are very similar to those used for its aluminum counterpart since both structures perform the same function. Many design checks were made to evaluate structural performance for each loading condition. A summary of the constraints used during local optimization are shown in Table 1. Using these criteria to constrain investigations to a feasible design space, structural cost and/or weight was used as an objective function in the optimization routine to find the best possible design.

Of the constraints and guidelines listed in Table 1, the minimum skin buckling, minimum stiffness, and tension damage tolerance constraints tended to be the most critical. The minimum skin buckling criteria was initially limited to be no less than 40% of the ULTIMATE compression load (i.e., skin buckling was not allowed to occur below this load level). This effectively limited the amount of post-buckling that occurred in the structure. It was later reduced to 33% of the ULTIMATE load, as discussed in "Criteria and Guideline Sensitivities". The minimum stiffness criteria used was based on

**Design Variables:**

C1 = Number of Stringers
C2 = Number of Frames
C3 = Stringer Cross-sectional Area ($in^2$)
C4 = Stringer Material Density ($lb/in^3$)
C5 = Stringer Material Cost (\$/lb)
C6 = Skin Laminate Thickness (in)
C7 = Skin Material Cost (\$/lb)
C8 = Skin Material Density ($lb/in^3$)
C9 = Stringer Thickness (in)
L = Length of Crown Panel Quadrent (in)
W = Width of Crown Panel Quadrent (in)

**Design Functions:**

f1 = constant (3.132E-01)
f2 = C1 * C3 * C4
f3 = C1
f4 = C1 * C3 * C4 * C5 * (L-4)
f5 = C6 * C8 * L * W
f6 = C6 * C7 * C8 * L * W
f7 = C2
f8 = C1 * C2
f9 = C1 * C9 * L
f10 = C6 * L * W

**Cost Relationship Equation:**

f1 + 6.848E-3 * f2 +
1.176E-2 * f3 + 1.087E-5 * f4 +
8.034E-5 * f5 + 1.098E-5 * f6 +
1.054E-2 * f7 + 5.586E-4 * f8 +
8.875E-6 * f9 + 1.106E-7 * f10 =

Cost for Design Family C1 Relative to Aluminum Baseline

Cost is based on global optimization results for family C1 (Ref. 1 and 7)

**Figure 5: Cost Relationship Used During Local Optimization**

90% of the baseline aluminum airplane fuselage stiffness. The aluminum design is heavier in the forward end due to the higher load levels. This directly corresponds to a higher stiffness in that region. The minimum stiffness was lower in the aft crown panel where the skin gages are smaller due to the lighter loads. The stiffnesses used to constrain the composite crown designs may not be the absolute minimum fuselage stiffness allowed for this type of structure. Without extensive analysis of the effects of fuselage stiffness on aerodynamic control, ride quality, and flutter limitations, however, it was assumed to be sufficient. A longitudinally oriented through penetration that included a central failed frame element was used to evaluate hoop tension damage tolerance. Analytical corrections for configuration, stiffness, pressure, and curvature were included.

The loading conditions applied to the crown panel include both flight loads and internal pressure loads. The critical flight loads are derived from a 2.5g symmetric maneuver, factored by 1.5 to an ULTIMATE load condition, with a 13.56 psi internal pressure differential applied simultaneously. This loading combination gives the maximum axial tension load in the crown panel. The tension load distribution and the associated shear loads are shown in Figure 6. The maximum compression load in the crown comes from a -1.0g symmetric maneuver and was derived from the 2.5g case by using a 40% reversal assumption, again factored by 1.5 to achieve an ULTIMATE load condition. Two pressure cases are also used to design the fuselage structure. An ULTIMATE pressure load case (18.2 psi pressure differential) is applied without any additional flight loads. This case is critical in the crown for frame loads and for the longitudinal splices. A FAILSAFE pressure load (10.3 psi pressure differential) is used to evaluate the tension damage tolerance in the hoop direction.

**Structural Criteria Related Design Checks**
- o Ultimate failure strains
- o Tension damage tolerance (axial and hoop directions)
- o General panel stability
- o Local buckling/crippling

**Structural Guidelines**
- o Minimum overall axial and shear stiffness no less than 90% of an aluminum counterpart stiffness
- o Minimum skin buckling percentage of 33% ULTIMATE load
- o Maximum of 60% of the total load in either the skin or stringer element
- o Maximum stringer spacing based on skin area between adjacent stringers and frames
- o Minimum skin gage based on impact damage resistance data

**Composite Laminate Guidelines**
- o Poisson ratio mismatch between skin and stringer laminate less than 0.15
- o A minimum of four ±45°, two 0°, and two 90° plies in any laminate.
- o Ply angle increments of 15° in final laminate

**Geometric, Configuration, or Manufacturing Constraints**
- o Maximum stringer height
- o Minimum stringer flange widths
- o Stringer web angle limitations

**Table 1: Structural Performance Constraints and Guidelines**

The design criteria, structural guidelines, and loading conditions were all included in the design tool for crown panel applications. When appropriate, each criteria was checked for the four load cases applied at a given point on the crown panel. Only designs that met all of the design criteria and constraints were evaluated for weight and cost using the objective function. Seven different locations on the crown panel were evaluated, each having unique load requirements. Combined, these seven load points were used to optimize the entire crown panel. The blending of the individual design points is discussed in the section entitled "Blending Function".

During the course of crown panel local optimization, many different design combinations were considered. Certain cost trends and sensitivities to specific design variables and constraints were observed. A few of the trends and relationships stood out as being significant. The effects of structural geometry, namely stringer spacing, was found to have a large impact on the total panel cost. In addition to the geometry, the material type chosen for use also impacted the final cost significantly. The materials traded in this study included a low cost, low modulus graphite/epoxy system, a higher cost, intermediate modulus graphite/epoxy system, and a graphite/fiberglass hybrid system. In addition, the panel design and cost were found to be sensitive to small changes in the critical structural guidelines. The structural guidelines considered for this sensitivity study included the minimum initial skin buckling load level and the minimum axial fuselage stiffness.

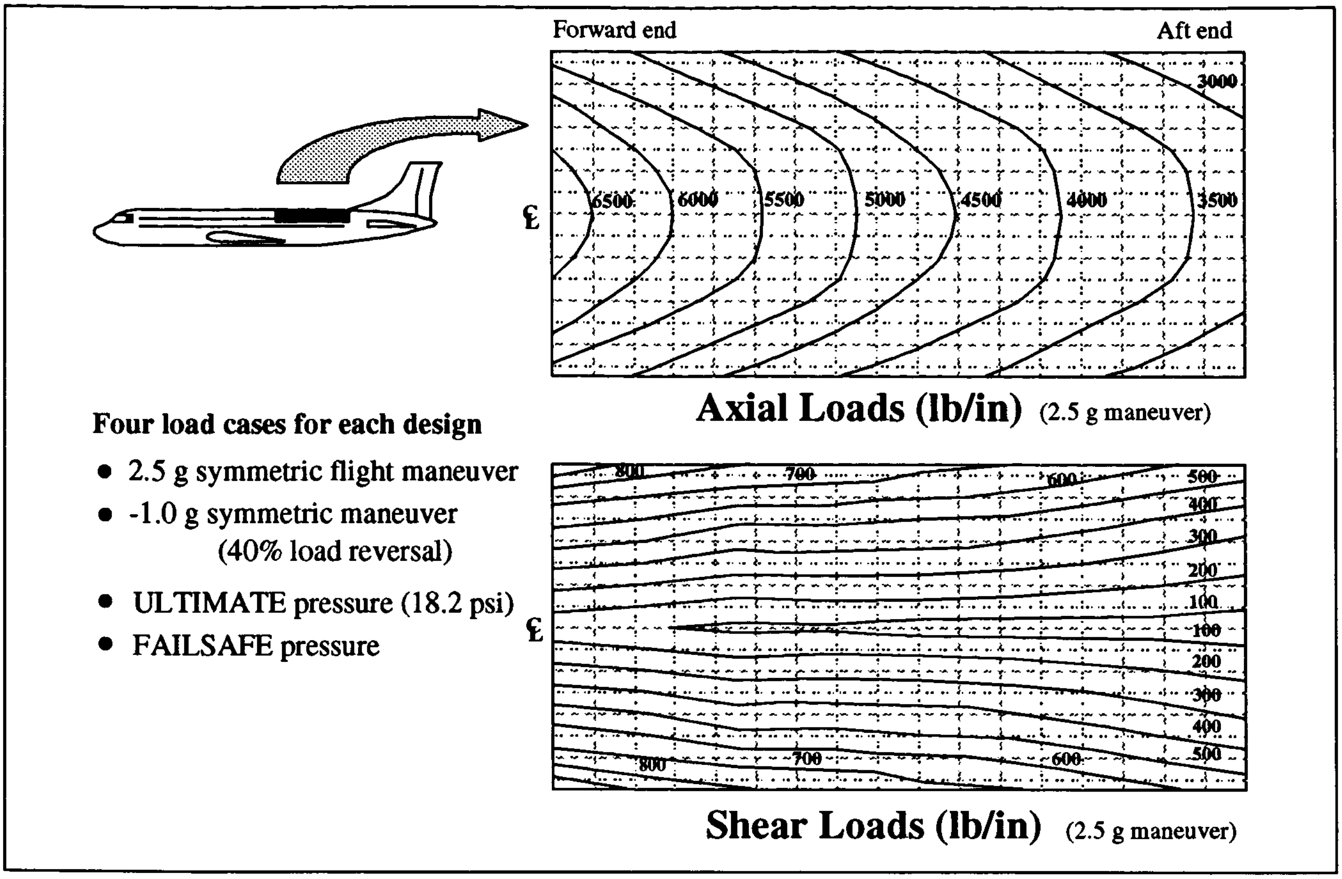


**Figure 6: Crown Panel Loads**

## SENSITIVITY STUDIES

### Geometric Parameters

For a given load point on the crown panel, the optimum design was determined by considering a wide range of skin and stringer thicknesses. For each individual design, the analyst defined the number of skin and stringer plies. The design tool was used to determine the cross-sectional geometry and spacing of the stringers, and the skin and stringer ply angles that simultaneously meet the design criteria and minimize the cost. For a given load condition, this involved hundreds of combinations of skin and stringer laminate thicknesses. As an example, a thin skin and thin stringer tended to be relatively inefficient and expensive since the required stringer spacing was very small and the design was relatively heavy. A thicker skin and stringer laminate, however, was more efficient in terms of cost due to a wider stringer spacing. Note that the stringer spacing became limited by a trade with skin weight, minimum skin buckling, and maximum stringer spacing guidelines.

The results of this design exercise are shown in Figure 7. Each point represents the best design for a given skin and stringer laminate thickness. From the scatter of points shown in Figure 7, a trend relating to the stringer spacing is shown by grouping the points with similar stringer spacings. These groupings are shown in the shaded areas. The wider stringer spacings typically correspond to a lower cost and higher weight. The optimum design for this load condition is defined by a constant value line, which corresponds to the value of a pound of weight savings.

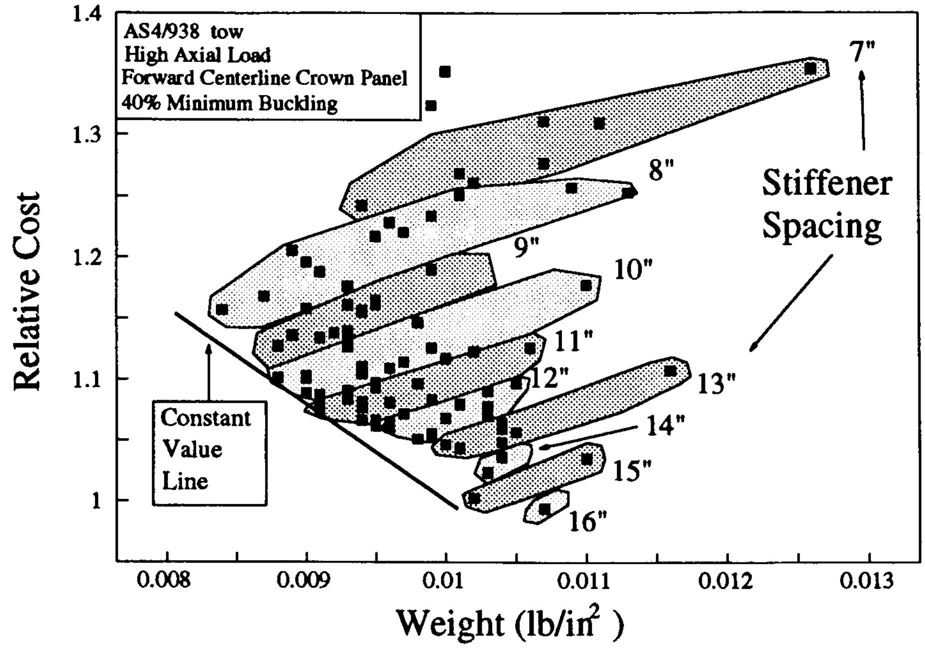


**Figure 7: Design of the Forward Crown Panel - Designs for All Practical Combinations of Skin and Stringer Thickness Combinations**

The relationship between stringer spacing and cost can also be seen in Figure 8. The high axial load case shown in Figure 8 is the same data presented in Figure 7. A similar design exercise was performed for an aft crown panel case and is included to show the effects of a lower axial load on the cost. The design points for the lower load case tend to have larger stringer spacings due to the lower loads. The minimum skin buckling constraint limits the maximum stringer spacing for both load cases.

The effects of the design constraints and guidelines on the results can also be seen in Figure 8. For almost every design point, the minimum skin buckling constraint defined the stringer spacing. For stringer spacings less than 10 inches, the designs were also limited by tension damage tolerance issues. The smaller stringer spacings typically had thinner skins which directly affect the hoop damage tolerance properties. The larger stringer spacings are affected by the minimum stiffness constraint. In the forward crown panel, where the axial loads are highest, a heavier, and therefore stiffer, structure is required than in the aft section where the axial loads are less severe. For stringer spacings greater than 10 inches, the tendency for the high load, forward crown designs to be higher in cost than similar designs in the aft crown panel can be attributed to the different stiffness constraints for these load conditions. The effect of the stiffness constraints is significant in that it can penalize the cost of the design by requiring either smaller stringer spacing or additional material to meet the required minimum target. As discussed earlier, this particular constraint needs to be evaluated further to avoid any arbitrary penalties to the cost and weight of a composite fuselage by requiring it to be stiffer than is necessary.

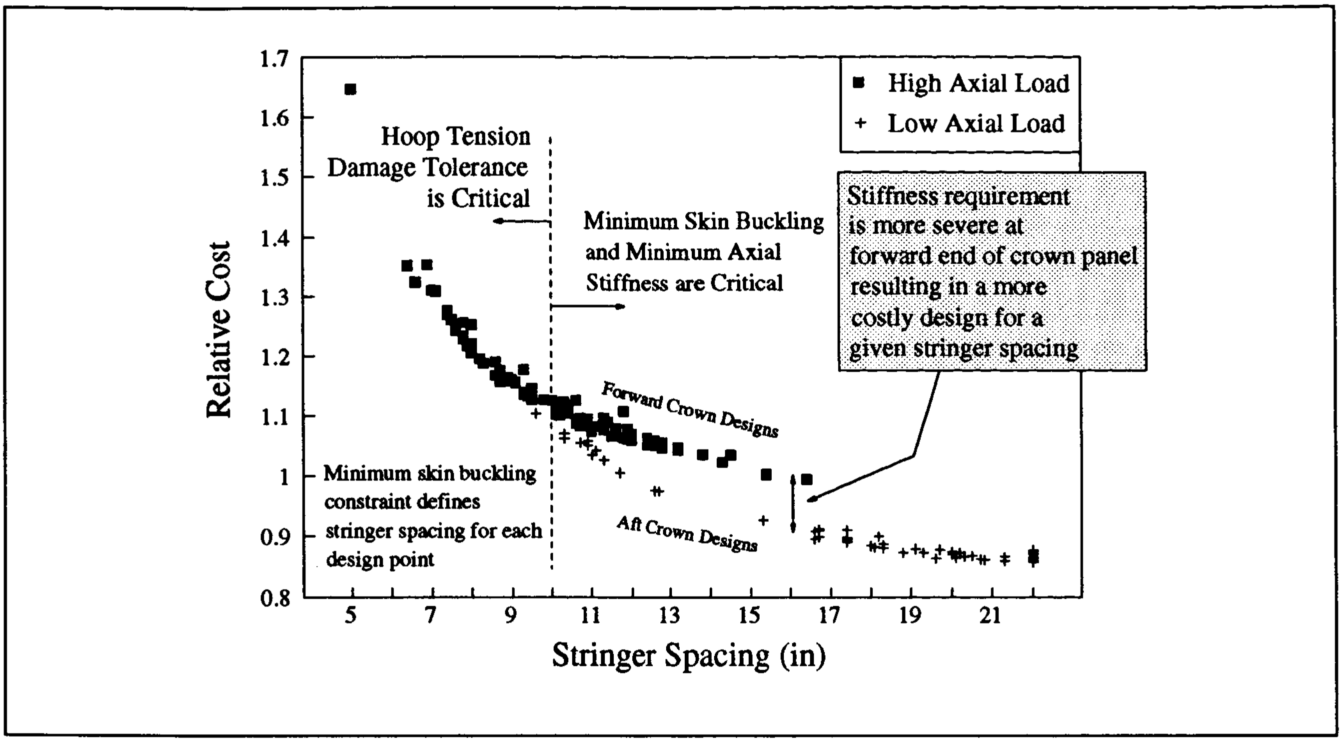


**Figure 8: Relationship Between Stringer Spacing and Cost**

Strong relationships with stringer design variables can be explained by looking closer at the cost breakdowns. Figure 9 shows the breakdown of total crown panel cost in percentages. The categories shown that are affected by the number of stringers account for 61% of the total cost. The effect of the number of stringers on each category may not be directly proportional, but is still significant. For example, in the case of the crown panel assembly, both longitudinal and circumferential splices are included in the cost breakdown. The number of stringers affects this cost center only through the stringer splices in the circumferential splice operation. A significant part of the assembly cost is therefore directly proportional to the number of stringers, yet the remaining part is unaffected.

Sensitivity of the optimum design configuration to changes in individual element costs provides further insight into design/cost relationships. As an example, a study considering a range of stringer costs was conducted, with the results shown in Figure 10. It is evident that the original trend to eliminate as many stringers as possible to minimize cost is true for stringer element costs varying from 50% to 400% of the original assumptions. For this range, the details for each optimum design point were nearly identical and cost differences directly related to the assumed change in stringer costs. The current study indicates that from a geometric standpoint, the most significant variable is stringer spacing.

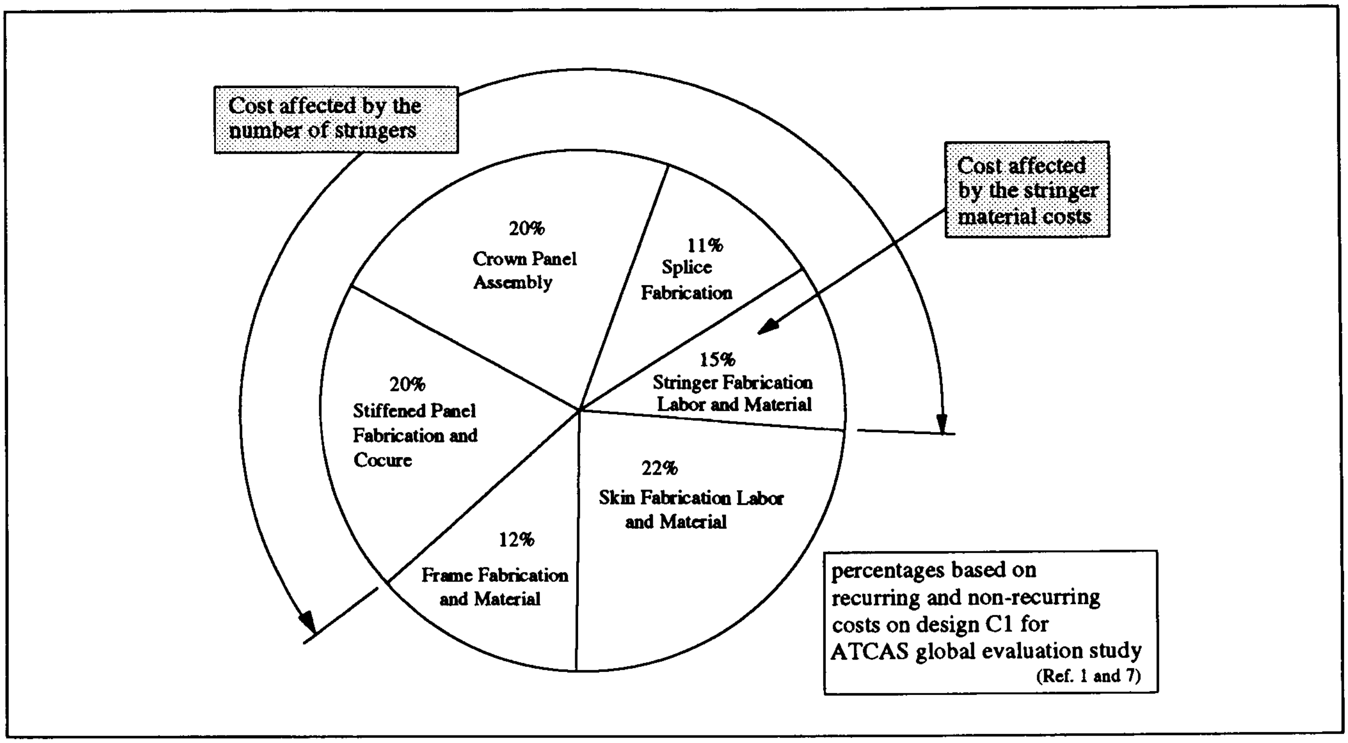


**Figure 9: Cost Breakdown for Baseline Crown Panel From Global Evaluation Study**

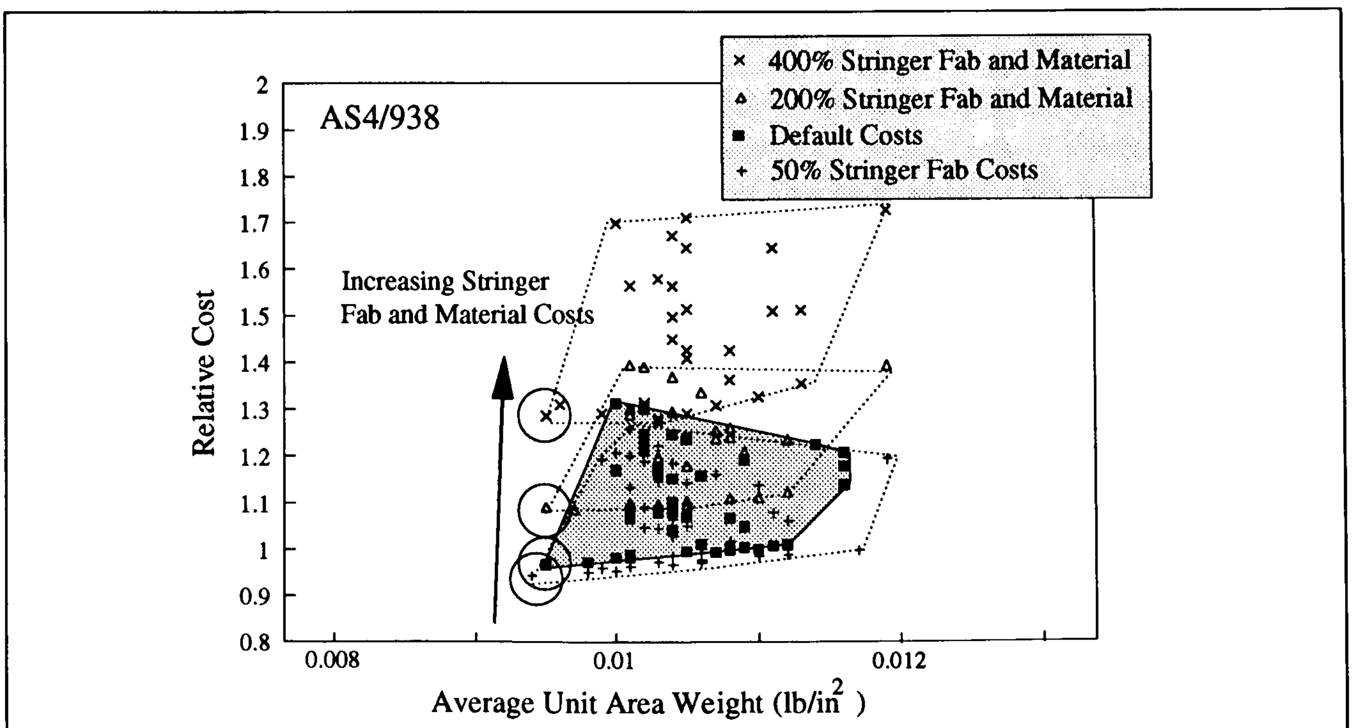


**Figure 10: Effect of Increased Stringer Element Costs on Design**

**Material Parameters**

The final crown panel design used an AS4[4]/938[5] material system. This choice was made based on a comparison of cost/performance relationships with other material systems. These relationships were determined by using the appropriate material properties during design/cost optimization studies. Some important properties for fuselage performance, such as tension fracture strength, have complex relationships with fiber stiffness, matrix properties, and material form. Reference 3, which is included in this proceedings, discusses results from ATCAS tension fracture material characterization tests.

Design optimization results are shown for two material systems in Figure 11. The higher modulus of the IM6[6]/938 material system is evident in that the best IM6 design case is lower in weight than the best AS4/938 design; however, the AS4 design was found to be more attractive after considering the value of a unit weight savings. A discussion of optimization studies involving the graphite/fiberglass hybrid appears later in the subsection entitled "Criteria and Guideline Sensitivities".

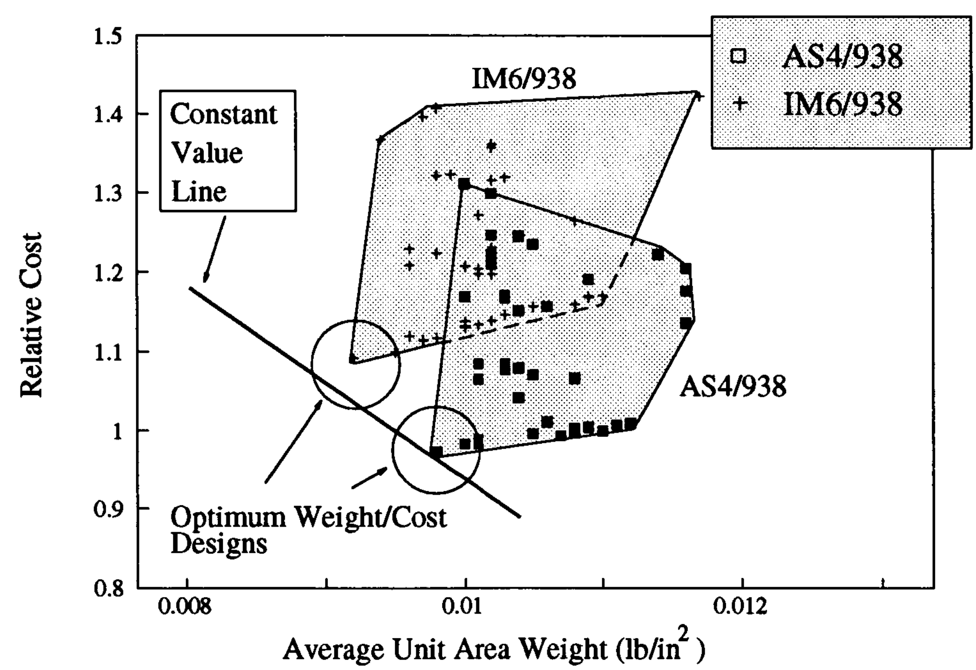


**Figure 11: The Effect of Material Choice on the Design**

**Blending Function**

In order to transition from a number of point designs into a final, cohesive design, the individual points must be blended together. For the current study, the seven load points shown in Figure 12 were considered, each having unique load requirements. In order to blend the individual points in the crown

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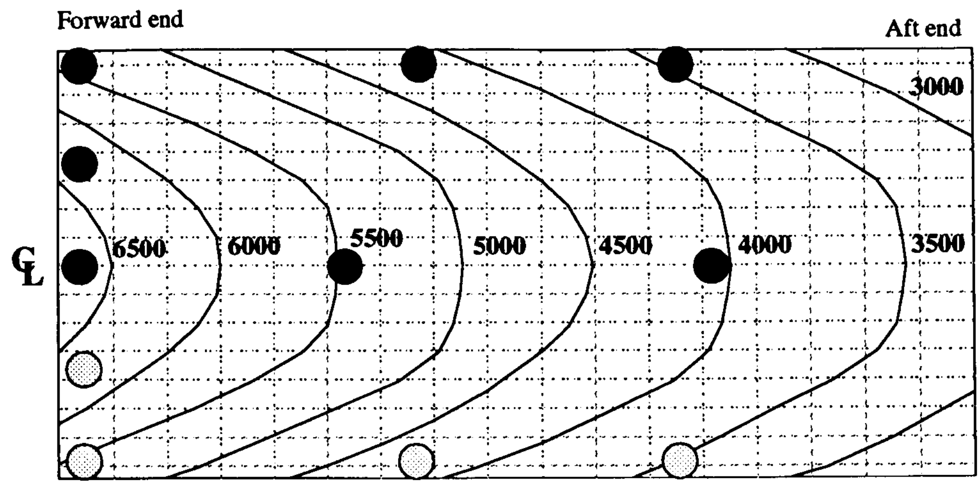
4 AS4 is a graphite fiber system produced by Hercules, Inc.

5 938 is a epoxy resin system produced by ICI/Fiberite.

6 IM6 is a graphite fiber system produced by Hercules, Inc.

panel without changing the cost relationships, a number of manufacturing constraints were imposed. The first imposed constraint is that the stringers remain straight and, therefore, the stringer spacing between two adjacent stringers is constant along the length of the crown panel. Stringer spacing was, however, allowed to vary across the crown panel width (i.e., the stringer spacing at the edge of the panel could be different than at the center). In addition to the stringer spacing constraint, it was also assumed that the individual ply angles will remain constant, forcing the laminates at any adjacent points to be consistent. Ply dropoffs were allowed between design points, as long as fabrication rates were unaffected and the remaining laminate was a reasonable subset of the adjacent laminates.




Crown Panel Axial Loads (2.5g maneuver) lb/in
Symmetric about centerline

**Figure 12: Load Points Used to Design the Crown Panel During Blending**

Many interesting combinations of design variables result from trying to blend an entire design. For example, the optimum stringer spacing at the more highly loaded forward end of the crown panel tended to be smaller than the stringer spacing at the lightly loaded aft end. This was seen in Figure 8. The dominating reason for this difference was the effect of the minimum skin buckling constraint that was imposed. When blending the stringer spacings, the larger stringer spacing possible in the aft end would penalize the forward end for both cost and weight. Likewise, the smaller spacing trend in the forward end would penalize the aft end of the crown. After considering both of these scenarios, it was determined that the penalty of the larger stringer spacing on the forward end was smaller than the penalty imposed by forcing a smaller stringer spacing on the aft end. This result is reasonable if one considers the cost breakdown and stringer effects of the baseline design shown in Figure 9.

Based on results from the initial point design optimization exercise, a series of blended crown panel designs were developed. The stringer spacings obtained from the initial study were imposed for the entire crown panel. The results of this study are shown in Figure 13. Initially, the laminate layups were not constrained and were still somewhat inconsistent between adjacent design points. This condition is labeled as "unblended". Three stringer spacing scenarios were chosen for further consideration. These three designs were further blended to achieve consistency between adjacent laminate design points and are labeled as "blended". Figure 13 shows that blended designs generally have higher cost and weight since additional plies were needed to satisfy the requirements.

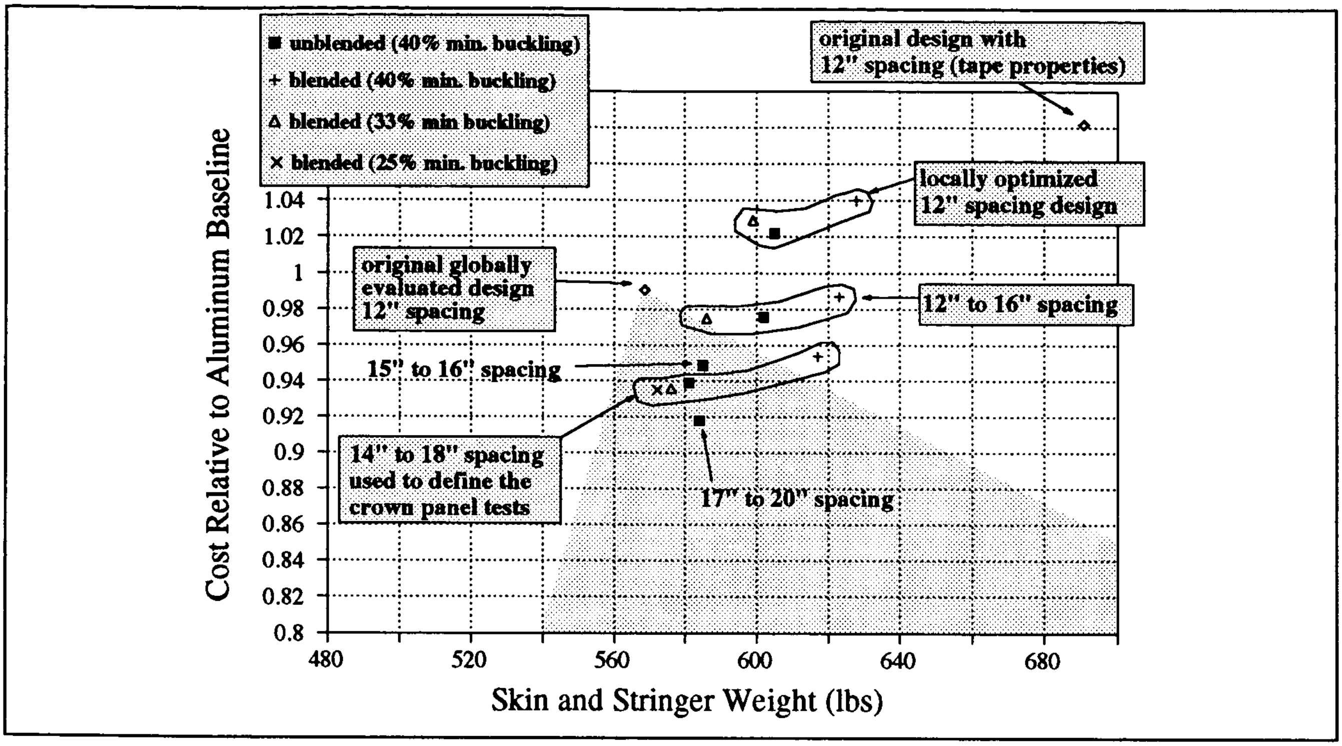



**Figure 13: Local Crown Panel Optimization and the Effects of Blending**

The development of advanced tow placement technology which allows some point to point variation in fiber angle within a ply would help minimize this effect. In addition, the generalization of optimization schemes used for the design cost model would enable analysis of blended designs, resulting in lower costs and weights than achieved in current efforts.

## Criteria and Guideline Sensitivities

Initially, a minimum skin buckling constraint was imposed that limited the design such that no skin buckling could occur below 40% of the ULTIMATE load levels. This constraint was critical to the cost of most designs in that it controlled the maximum stringer spacing. To determine the effect of this criteria, the crown panel was redesigned with the same stringer spacings, but with a minimum buckling load of 33% of the ULTIMATE load. This lower constraint, along with no change in the stringer spacing, resulted in thinner laminates and different design drivers, essentially lowering the cost and weight of the design. The design was no longer limited by minimum buckling but was critical for minimum stiffness and hoop damage tolerance. Further reduction of the minimum buckling criteria had no effect on the design and only increased the margin of safety on the minimum buckling since this criteria was no longer critical. The effects of the minimum buckling criteria can be seen in Figure 13 by the points labeled "blended (33% minimum buckling)".

Since the cost of the crown panel is sensitive to the minimum skin buckling guideline, research is needed to better understand the effects of design on this requirement. The current guideline assumes a concept is at risk when skin buckling occurs below a cut-off level, independent of design details. This is likely not the case and a better definition of the requirement is needed to avoid overly conservative and costly designs.

It can be seen that an increase in stringer spacing significantly improves both the cost and weight of the structure. Other design guidelines that limit the stringer spacing will become critical as the stringer spacing increases. One of these guidelines is often referred to as a blowout panel. The blowout panel is defined as the maximum skin area between adjacent stringer and frame elements and is limited to a given size defined by the aircraft's environmental system capabilities. Using typical values for this guideline from existing aircraft, a maximum stringer spacing for the composite hat stiffened crown panel is about 18 inches. Therefore, for the final crown panel design, the maximum stringer spacing was limited by this value.

The effect of the fuselage stiffness was discussed previously and is shown graphically in Figure 8 by comparing the trends for the two load conditions considered. The difference in these trends can be attributed to a difference in the overall stiffness requirements between the forward and aft crown panel.

The only remaining design criteria that was consistently a critical design driver in the crown panel is tension damage tolerance. The effect of this criteria on the design is most apparent when a material system that has superior tension damage tolerance properties is considered in the design. An intraply graphite/fiberglass hybrid material system is a good example of a material system with excellent damage tolerance properties and low material cost, but lower modulus. This material is discussed in detail in Reference 3. Using a minimum skin buckling criteria of 33%, the 14- to 18-inch design in Figure 13 was designed using the hybrid material system. The results of this exercise are shown in Figure 14. Assuming for a moment that no stiffness criteria existed, the improved tension damage tolerance of the hybrid material reduced the crown panel cost about 6% with a small weight penalty due to the increased density of the hybrid material. When the stiffness criteria are imposed, a number of additional plies are required, increasing the cost of the hybrid crown panel close to that of the graphite design, with a significant weight penalty. Looking at the entire airplane, however, there are many locations on the fuselage where the hoop tension damage tolerance criteria are critical. For certain fuselage sections forward of the wing and immediately forward of the empennage, stiffness may not be a critical design guideline as it is in the highly loaded center sections. In these more lightly-loaded sections, a hybrid material design may provide for cost-effective structure.

## LOCALLY OPTIMIZED CROWN PANEL DESIGN

The many sensitivity studies and design combinations performed during local optimization resulted in a final design for the crown panel. A sketch of the details of this final design are shown in Figure 15. The stringer spacings chosen were based on the results of the blending exercise and were limited by the blowout panel criteria discussed earlier. A stringer spacing of 14 inches at the center of the crown was determined by the higher axial load at the center of the panel. Lower axial loads at the edge allowed for a wider spacing resulting in a lower total panel cost. The stringer laminate ply angles tended towards 0°. A minimum number of ±45° and 90° plies were included to satisfy laminate layup guidelines. The skin plies were also constrained to have a minimum number of 0°, ±45°, and 90° plies. In the aft end, the remaining plies at the center of the panel tended towards 90° to resist the hoop tension damage tolerance criteria. Towards the side of the aft crown panel, laminate thickness increased, with the remaining plies tending towards ±45° to resist the minimum shear buckling criteria. A compromise was found that minimized the total cost and incorporated ±60° plies to resist shear buckling at the edge and hoop tension damage tolerance at the center. In the forward end, this same base laminate required additional plies to resist the increased axial and shear loads. In the center of the panel, longitudinal plies were required to resist the additional axial loads. However, at the edge, the higher shear loads required more angle plies. The final ±15° plies added to the forward crown provided the shear requirements at the edge and the axial requirements at the center. Additional details to account for the joints and frames were included in the final design and cost estimates, but there was no attempt to optimize these details.

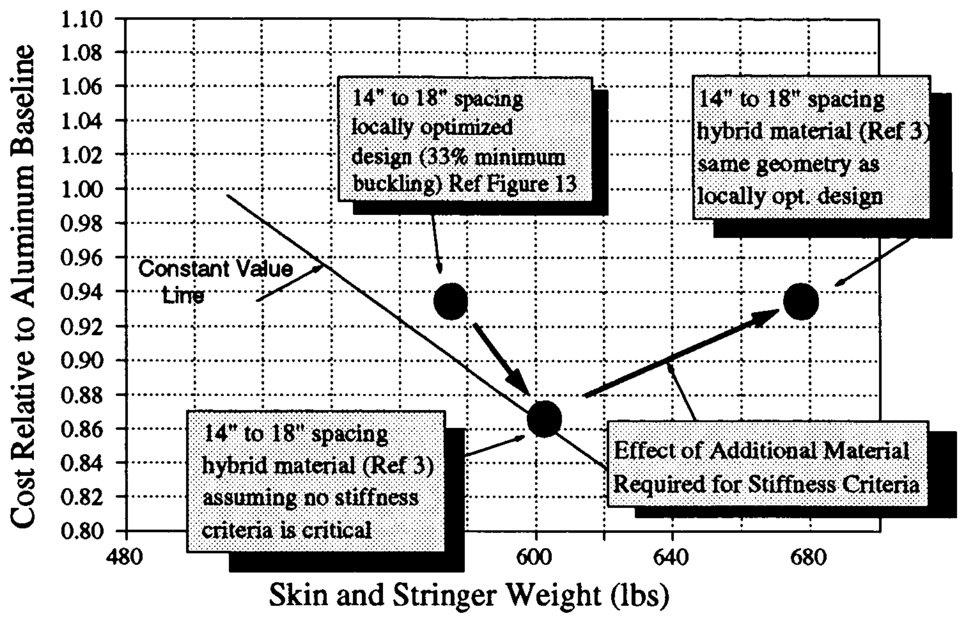



**Figure 14: Effect of Hybrid Material on Crown Panel Design**

The criteria and guidelines that drove the final crown panel design are shown in Figure 16. Hoop tension damage tolerance was more critical in the aft crown panel where the skin laminate was thinner. The axial stiffness tended to be critical almost everywhere, suggesting that this particular criteria be studied to ensure that the design is not arbitrarily over-constrained. The minimum buckling guideline was not as critical in the aft crown as it was in the highly loaded forward crown panel. Any increase in stringer spacing or a higher minimum buckling, however, would quickly make this criteria a dominant design driver. Finally, the shear stiffness criteria was critical only in the forward part of the panel towards the lower side, where the shear loads were highest and the stiffener spacing largest.

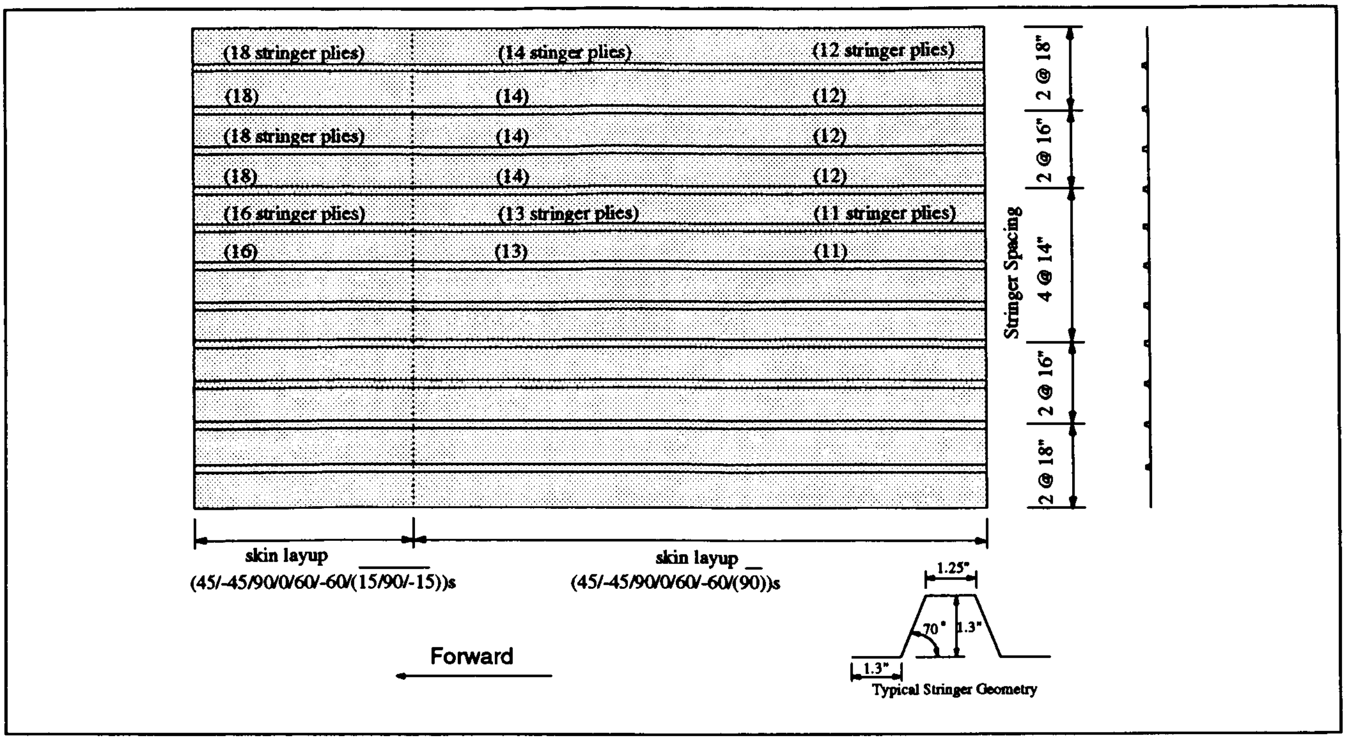



**Figure 15: Final Crown Panel Design**

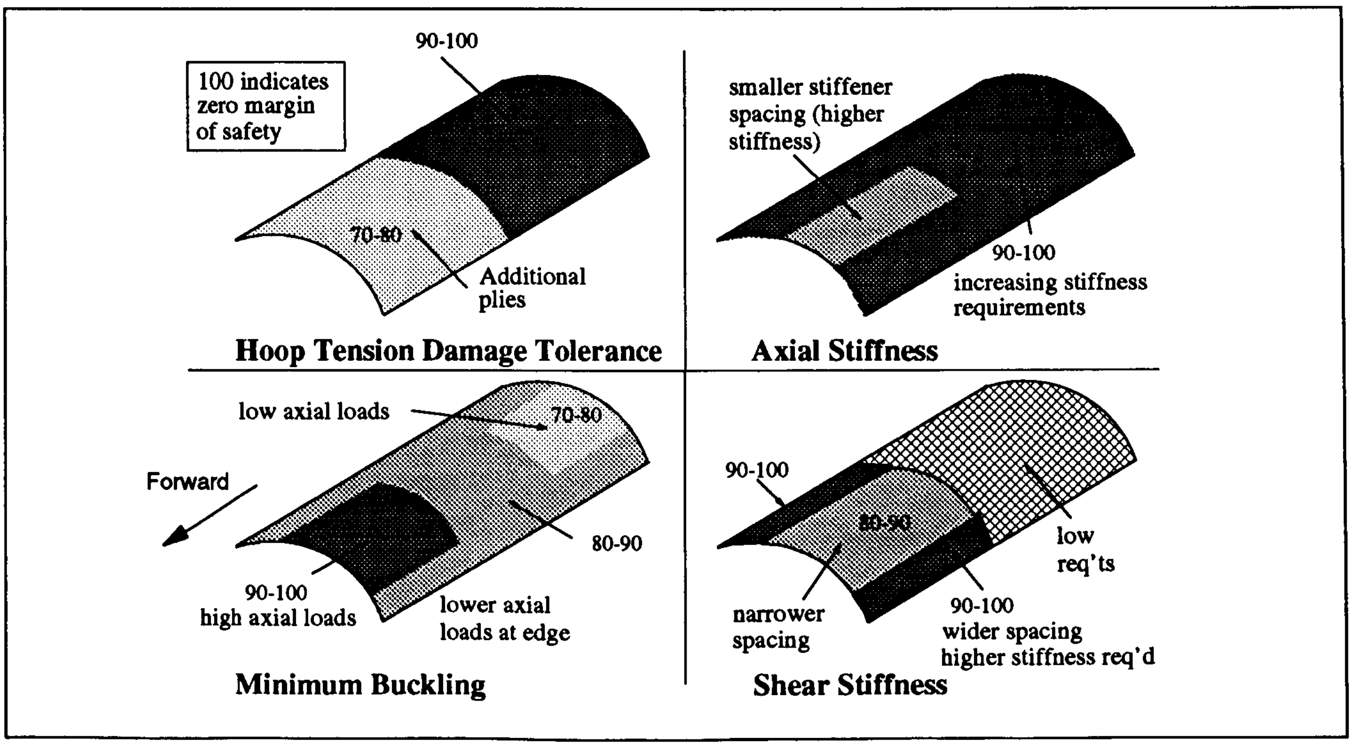



**Figure 16: Design Drivers for Locally Optimized Crown Panel**

## LESSONS LEARNED DURING LOCAL OPTIMIZATION

The usefulness of a design tool that combines optimization with realistic design criteria and an ability to evaluate the manufacturing cost has been shown to be quite effective in understanding the sensitivities of the design details to the criteria and cost. Many improvements have been identified during the course of this initial work with a design tool of this type. The most significant is the need to design the entire panel, combining the different design trends and loading levels in each part to make the design consistent and feasible for manufacturing. The current point optimization characteristics of the design tool, typical of most structural sizing tools, makes the job of blending the point-to-point variations in a real design very labor intensive. Future work with the design tool will develop algorithms and optimization approaches to effectively blend designs for a full scale panel subjected to changing load distributions and local variations in the design such as joints, splices, and cutouts.

An understanding of the effect of the design criteria on an optimized design is another important feature that a design tool of this type can provide. As with all optimization, the algorithm will take advantage of the criteria or constraints to minimize its objective function. As is often true, if a design is constrained by a given criteria, there is another criteria that will quickly dominate the design should an improvement be made which relieves the initial constraint. With the many interactions that occur in a design study such as this, careful attention to the trends and criteria can define the direction of future work that would be of benefit to the design.

## CONCLUSIONS AND RECOMMENDATIONS FOR FUTURE WORK

The local optimization study for composite fuselage crown panels revealed many insights into the relationship between manufacturing cost and design details. A design tool was developed to aid in these investigations. Steps taken in developing the design tool, the sensitivity studies that were performed to identify critical variables, and the technique used to arrive at a final optimum crown panel design were discussed.

It was concluded that design constraints used to limit the design can be very important when optimizing a real structure for cost. Constraints such as minimum stiffness and skin buckling can be a significant cost driver. The tension damage tolerance design criteria are also a significant design driver in many parts of the crown panel.

The benefits of a design tool that combines structural constraints and manufacturing costs were also shown. Sensitivity studies showed the effect of different constraints on the cost and weight of optimized designs. Material trade studies showed that many interactions affect the cost effectiveness of improved material properties. Hybrid materials were shown to have promise in a significant portion of the airframe.

A final optimized crown panel design was completed utilizing the data obtained from these sensitivity studies. Stringer spacings ranging from 14 to 18 inches were selected. The optimized design showed significant cost savings relative to the original global evaluation study.

During the course of the study, it became apparent that there are many research areas that need to be addressed. A summary of these items are listed below:

> Stiffness criteria for a composite fuselage must be evaluated further to avoid overly conservative designs. This is a potential cost driver for composite fuselage structure.

Stringer spacing is a dominant design driver in the crown panel. The minimum load below which skin buckling is not allowed needs to be addressed for different design configurations to avoid unnecessary cost penalties.

Blending of adjacent points during an optimization cycle is the key to a realistic structural optimization problem. The development of an automated blending function is critical.

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