THE EFFECTS OF DESIGN DETAILS ON COST AND WEIGHT OF FUSELAGE STRUCTURES[1]

G. D. Swanson, S. L. Metschan, M. R. Morris

Boeing Commercial Airplane Group

C. Kassapoglou

Sikorsky Aircraft

ABSTRACT

Crown panel design studies showing the relationship between panel size, cost, weight, and aircraft configuration are shown with comparisons to aluminum design configurations. The effects of a stiffened sandwich design concept are also discussed. This paper summarizes the effect of a design cost model in assessing the cost and weight relationships for fuselage crown panel designs.

Studies were performed using data from existing aircraft to assess the effects of different design variables on the cost and weight of transport fuselage crown panel design. Results show a strong influence of load levels, panel size, and material choices on the cost and weight of specific designs. A design tool being developed under the NASA ACT program is used in the study to assess these issues. The effects of panel configuration comparing postbuckled and buckle resistant stiffened laminated structure is compared to a stiffened sandwich concept. Results suggest some potential economy with stiffened sandwich designs for compression dominated structure with relatively high load levels.

INTRODUCTION

Boeing is studying the technologies associated with the application of composite materials to transport fuselage as part of the NASA/Boeing Advanced Technology Composite Aircraft Structures (ATCAS) program. As part of this program, a designer's cost model [1, 2, 3] is being developed to quantify the complex interactions of aircraft design criteria, multiple load conditions, and the extensive number of design variables associated with composite structures. Analysis, optimization, and design routines plus a theoretical framework for assessing the cost are being combined into a tool that can aid a design engineer in the understanding and design of many structural components. The cost model effort, Composite Optimization Software for Transport Aircraft Design Evaluation (COSTADE), is being developed in coordination with a number of subcontractors, including Sikorsky Aircraft, University of Washington, Massachusetts Institute of Technology, Dow/United Technologies, and Northrop. The current study uses COSTADE in a developmental form to demonstrate and validate its usefulness for a number of composite fuselage crown panel designs.

Composite structural design can be very complicated if the proper design tools are not available. The aircraft industry has had decades to develop design charts and material allowables to aid the engineer in establishing an efficient aluminum design. This capability is still being developed for composite structures. The concept of a designer's cost model which combines preliminary design tools, laminate analysis, and an ability to handle multiple load conditions and criteria using an optimization routine has significant appeal in closing the existing gap in capability.

An important feature being developed for inclusion in COSTADE is a blending function. In the first part of the current study, the cost efficiency of large panel sizes is studied. The trend, which justifies large composite panels, was also discussed in [1, 3]. Since aircraft loadings are sensitive to their relative location, a large panel will likely have a wide range of loading levels from one end to the other. For example, a fuselage crown panel has higher axial loads at the center of the aircraft and lower loads closer to the tail or the nose. The optimum design at the heavily loaded end of the crown panel will likely be quite different than the optimum design at the lightly loaded end. The optimum design for the entire panel may be different than either of these. Most likely, the optimum will be a compromise between them. The intent of the blending function is to optimize the entire panel, accounting for all the local load variations that occur within the panel. In the current study, the blending was done manually. The lessons learned and the pitfalls encountered are being incorporated into the blending function development. This feature of a designer's cost model is likely to be one of the most important features responsible for reducing the cost of a design and will be discussed in future work.

CURRENT STUDY

The present study is intended to show the applicability of the designers cost model to a fuselage crown panel design for different design configurations. The study is divided into two parts, the first being an evaluation of the crown panel design as a function of aircraft size and load levels. Comparisons to equivalent aluminum aircraft designs are made when possible. The second part of the study is focussed on the effect of panel configuration, specifically the effect of large stiffener spacing and sandwich core. Previous studies [1, 4] suggested that minimum cost is achieved by increasing the stiffener spacing and is often limited by skin buckling constraints. The use of sandwich skins to increase the buckling capability of the skins between stiffeners (stiffened sandwich structure) is addressed.

Design Constraints

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 constraint limited skin buckling to no less than 38% of the ULTIMATE compression load.

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 equivalent aluminum design
- o Minimum skin buckling percentage of 38% ULTIMATE load
- o Maximum stiffener spacing based on skin area between adjacent stiffeners and frames (16" for the current study)
- o Minimum skin gage based on impact damage resistance data

Composite Laminate Guidelines
- o Poisson ratio mismatch between skin and stiffener laminate less than 0.15 for both longitudinal and transverse directions
- o A minimum of two $\pm 45°$, 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 stiffener height
- o Minimum stiffener flange widths
- o Stiffener web angle limitations

Table 1:   Structural Performance Constraints and Guidelines

The minimum stiffness criteria used was based on 90% of the baseline aluminum airplane fuselage stiffness. This criteria is discussed further in [1].

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 damage tolerance analysis used in the present study uses assumed material properties for fracture properties and some assumed load redistribution characteristics. Current investigations into the response of composite structure to this type of damage are ongoing [5, 6]. Further work in this area will be incorporated into the designer's cost model. It is expected that the results will be affected, but the trends will be similar.

The loading conditions applied to the crown panel include both flight loads and internal pressure loads. The critical flight loads are derived from a scan of all the critical load cases used to design the aircraft. The typical tension load distribution and the associated shear loads were discussed in [1].   Maximum

longitudinal tension, compression, and shear cases were determined from the existing loads data. 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 FAIL-SAFE pressure load (9.6 psi pressure differential) is used to evaluate the tension damage tolerance in the hoop direction.

Cost Constraints

The relationship between the design details and the cost of a given structure is often hidden in the data provided by a cost estimator since these estimates are based on process and manufacturing parameters. It is the intention of the present study to approach cost estimation such that it forms a framework which allows the relationship between the design details and cost to be bridged [2]. The framework approach, as it is currently conceived, will relate the cost and design variables through a series of coefficients and functions defined in a Design Build Team (DBT) environment [7]. In this environment, the factory flow and the process steps used at any given company can be defined within the framework and these relationships can be used to optimize the structure to its desired objective function.

The cost algorithm in the current study is based on data collected during the crown panel [7, 8] and keel panel [9] global evaluation study of Boeing's ATCAS program. During global evaluation, a comprehensive manufacturing plan was compiled for each design to support a detailed cost estimate. The estimate included the recurring labor and material cost of 300 ship sets and the non-recurring costs. Six crown designs were evaluated and include both hat stiffened skins and sandwich panel designs. The four keel designs include similar concepts. In an earlier study [1], a limited cost relationship for a hat stiffened crown panel was established. This relationship assumed that the stiffener spacing was limited between 10" and 20", overall panel size was unchanged and no changes to the processes or manufacturing steps would be allowed. Additional relationships were developed for the present study to broaden these assumptions.

In the first part of the present study, a number of different design configurations are analyzed for composite material applications in which both the size of the crown panel and its diameter were varied. Two cost centers that are affected by a change in panel size were assumed to be constant in the cost relationship from [1]. These are the tooling and bagging costs. Since detailed cost data were available for the smaller keel panel designs, tooling and bagging costs could be established as a function of area. The variation of the tooling and bagging costs were assumed to vary linearly with size. These additional relationships are shown in Figure 1.

In the second part of this study, the effects of increasing the stiffener spacing to very large values were considered. The major effect of the stiffener spacing variation, not accounted for in the original relationship [1], was the bagging costs. The cost relationship [1] incorrectly tied the bagging costs to the design variable associated with the number of stiffeners. Although this assumption was reasonable for the limited range of stiffener spacings used in [1], the limiting case of no stiffeners yielded incorrect cost trends. Bagging costs from the detailed cost estimate for sandwich crown panel designs (Family D [7]) provided an estimate of the bagging costs for a similar size panel without stiffeners. A relationship was defined based on a linear variation of bagging costs with stiffener spacing. These additional relationships are also included in the equations shown in Figure 1.

604

The larger stiffener spacing resulted in a lower panel cost [1] and was limited in most cases by a minimum skin buckling constraint. It was assumed that by increasing the ability of the skin to resist buckling, a larger stiffener spacing, hence a lower panel cost, could be achieved. One approach to increasing the skin buckling resistance was to add core material to the skin laminate, effectively creating a stiffened sandwich structure. The addition of core to the design had a significant effect on the cost. The cost relationships relating to the core were extracted from the detailed cost estimate for the sandwich crown panel design [7]. The core costs were broken down into cost equations and are shown in Figure 1.

Cost relationships like this must be used carefully since much of the estimate is based on speculation of factory flow and technology development. A more general approach to cost modeling is currently being developed and will eventually be incorporated into the COSTADE program. As with the current approach, a more general theory will include design details and material properties as independent variables, providing insight into the general effect of criteria and design practice on the cost of a composite structure. It is intended that the general cost relationship structure may be customized by a user to fit any factory for which data is available.
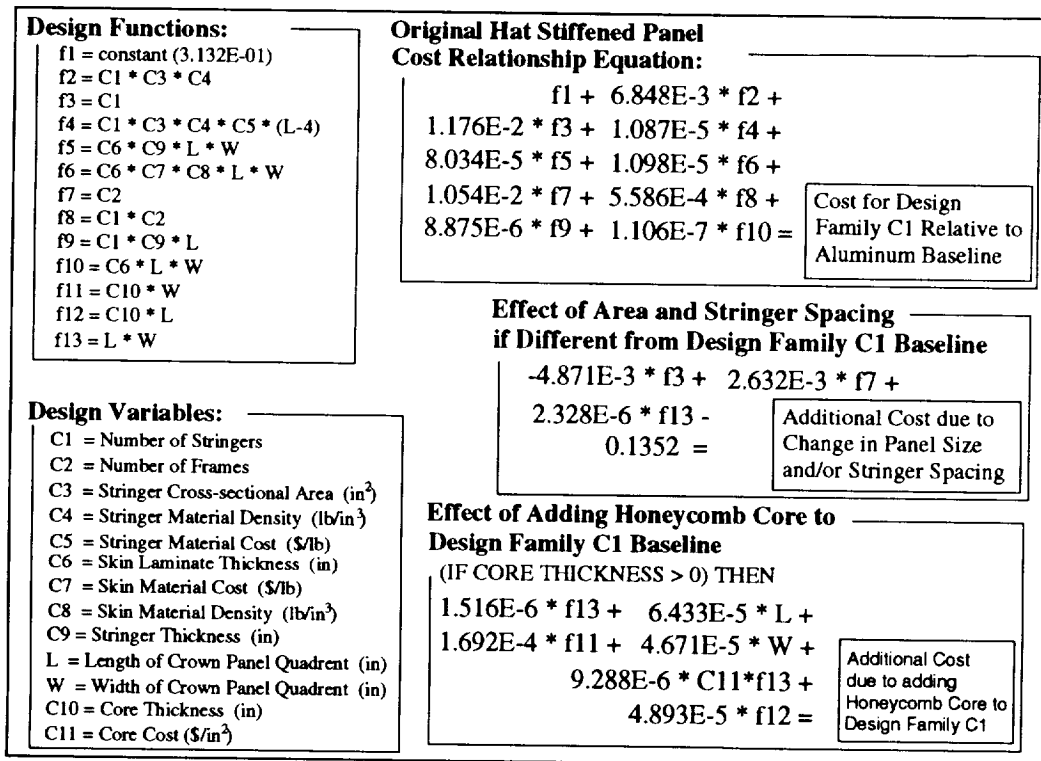
Design Functions:
f1 = constant (3.132E-01)
f2 = C1 * C3 * C4
f3 = C1
f4 = C1 * C3 * C4 * C5 * (L-4)
f5 = C6 * C9 * L * W
f6 = C6 * C7 * C8 * L * W
f7 = C2
f8 = C1 * C2
f9 = C1 * C9 * L
f10 = C6 * L * W
f11 = C10 * W
f12 = C10 * L
f13 = L * W

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)
C10 = Core Thickness (in)
C11 = Core Cost ($/in^3)

Original Hat Stiffened Panel 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

Effect of Area and Stringer Spacing if Different from Design Family C1 Baseline

-4.871E-3 * f3 + 2.632E-3 * f7 +
2.328E-6 * f13 - 0.1352 = | Additional Cost due to Change in Panel Size and/or Stringer Spacing

Effect of Adding Honeycomb Core to Design Family C1 Baseline
(IF CORE THICKNESS > 0) THEN
1.516E-6 * f13 + 6.433E-5 * L +
1.692E-4 * f11 + 4.671E-5 * W +
9.288E-6 * C11*f13 +
4.893E-5 * f12 = | Additional Cost due to adding Honeycomb Core to Design Family C1

Figure 1: Cost Relationships used in Current Study

PART 1: THE EFFECT OF SIZE ON CROWN PANEL DESIGN

The first part of the study is intended to address the issue of size on the cost and weight of a crown panel. Loads and criteria from a number of existing aircraft were used as the basis for the study. Aircraft size ranged from a relatively short, small diameter to a very long, large diameter transport aircraft. The relative sizes of the aircraft used are shown in Figure 2. The fuselage crown panel from

section 46 aft of the wing was used. Loads in the crown panels were a function of the diameter, aft fuselage length, mass distribution, and structural load paths, just to name a few. As is shown in Figure 3, the load distribution on the crown panel is fairly consistent in that the axial loads decrease as a function of fuselage station and that the smaller aircraft have overall lower load levels.

In addition to the axial loads, hoop loads are a function of the diameter. A larger diameter fuselage has larger hoop loads. From Figure 2 it is observed that the larger diameter tends to be accompanied by a longer aft fuselage length, coupling the axial and hoop load levels. A ratio of hoop load to axial load may be an appropriate term to keep in mind when comparing configurations.
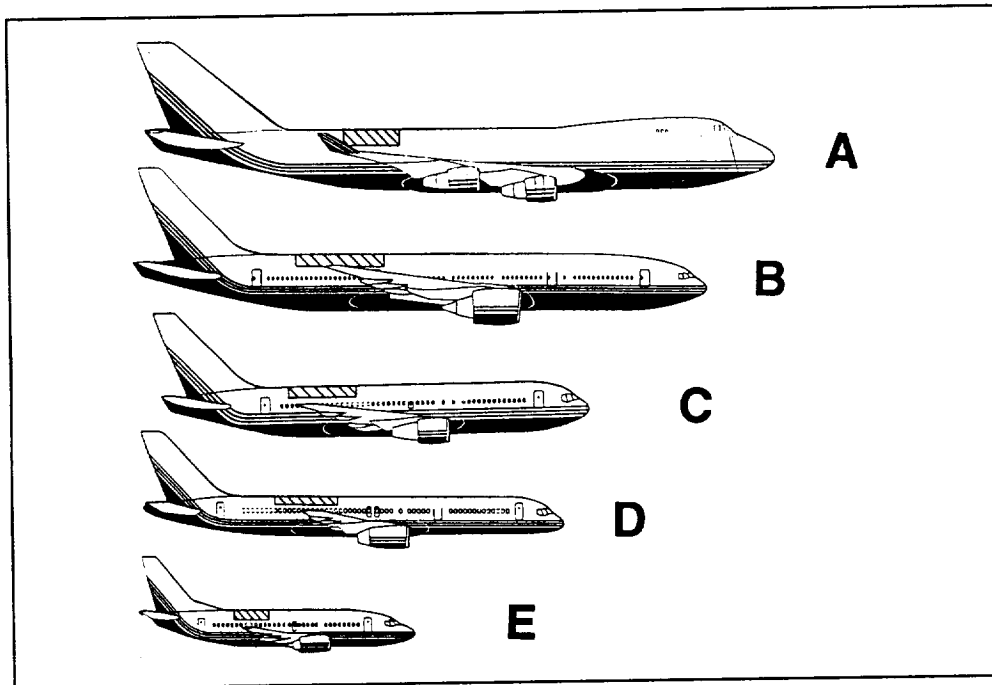
Figure 2: **Aircraft Configurations and Crown Panel Locations used for Current Study**

Effects of Size on Composite Cost and Weight

Composite crown panel designs were derived based on aircraft shown in Figure 2, using appropriate loads, geometry, and design criteria for each configuration. Design constraints listed in Table 1 were applied. AS4[2]/938[3] tow material was used as the primary material for the present study. The potential cost and/or weight savings of using other materials types such as a material with a higher modulus or a tougher resin is discussed in a following section. A brief summary of the design results is listed in Table 2.

---

[2]   AS4 is a graphite fiber system produced by Hercules, Inc.

[3]   938 is a epoxy resin system produced by ICI/Fiberite.
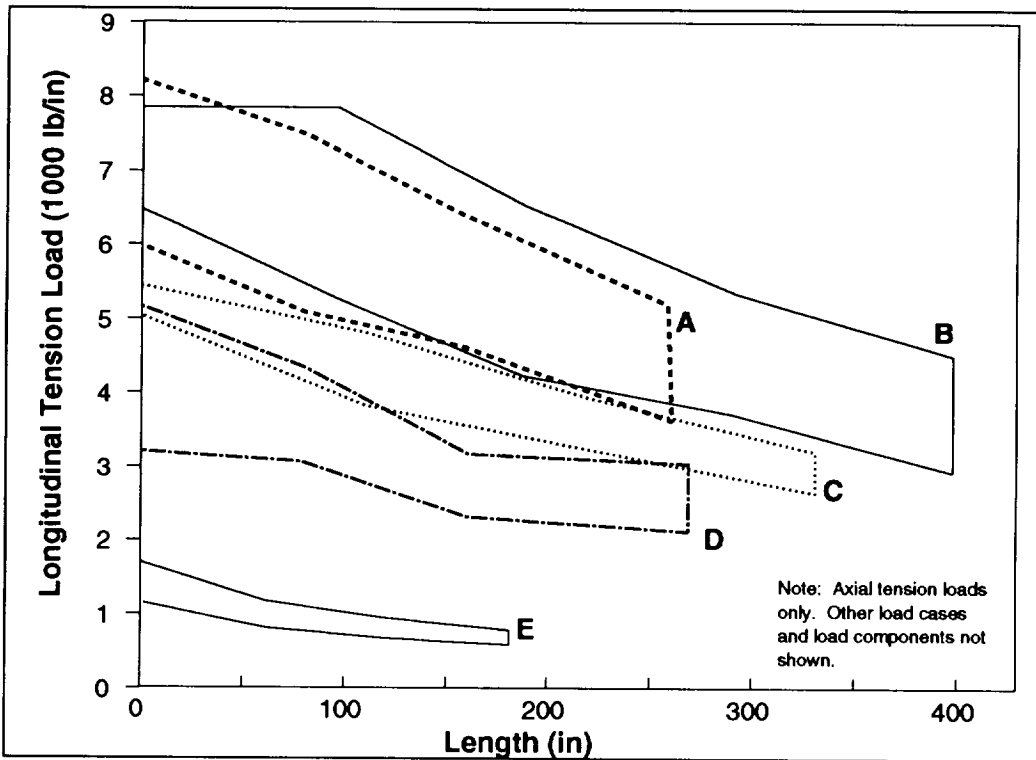
Figure 3: Crown Panel Load Envelope Comparison of Aircraft Configurations used for Current Study

	Radius (in)	Maximum Axial Tension Load (lb/in)	Minimum Axial Tension Load (lb/in)	Maximum Number of skin plies	Minimum Number of skin plies	Length (in)	Width (in)
A	128	8209	3600	18	14	260	213
B	122	7800	3100	18	16	398	211
C	99	5400	2700	14	12	330	150
D	74	5200	1960	14	12	270	180
E	74	1710	800	10	10	180	170

Crown Panel Structure - Section 46
Hat Stiffened Panel -
Stiffener Spacing 16"
Pressure Loads:
    9.6 psi Fail-safe
    13.6 psi Ultimate (with flight loads)
    18 psi Ultimate (acting alone)
	AS4/938 Tow Material (35% R. C.)
	$E_{11} = 19.2e6$ psi
	$E_{22} = 1.36e6$ psi
	$\nu_{12} = 0.32$
	$G_{12} = 0.72e6$ psi

Table 2: Data Relating to Designs used in Size Study

607

The effect of different panel lengths on the cost and weight were assessed. For this study, the crown panel width was held constant for each configuration and the length was varied. The section 46 crown panel length was measured from the aft wheel well bulkhead location. This implies that the longer crown panel lengths include more of the lightly loaded structure typical of the extreme aft crown panel. This effect is seen in the comparison of crown panel weight to the overall size shown in Figure 4. Trends for the four largest aircraft (A-D) show a lighter crown panel weight per unit area as the panel size increases, reflective of the greater amount of lightly loaded structure included in the larger panel sizes. Aircraft configuration E is constant as a function of size. This is indicative of a smaller aircraft that has a relatively low longitudinal loading, causing the design to be dominated by the hoop loading condition, which is constant along the length.

One important note to make regarding Figure 4 is the difference in the relationship of weight with size for a given configuration and the relationship with size between the aircraft configurations. It is often tempting in a study of this type to draw a line between points based on different aircraft configurations and claim a relationship between size and weight. If that were done, a misleading trend could be obtained.

In addition to the weight comparisons, cost data based on the relationships presented in Figure 1 were used to establish the comparative costs of the composite crown panels as a function of size. These data are presented in three forms, relative cost (Figure 5), relative cost normalized to the area (Figure 6), and relative cost normalized to the panel weight (Figure 7). In each figure, the cost is shown as a function of size for each configuration, with the data point showing the actual crown panel design point. In each figure, the recurring labor and material cost and the nonrecurring cost for airplane B are shown. The economy obtained with larger sizes is apparent in the recurring labor and non-recurring (tooling) components. The recurring material is less influenced by size and tends to follow a relationship similar to the weight trends discussed earlier.

An important point to note in Figures 5 through 7 is the lack of economy for very small panels. The effect of tooling costs and recurring labor costs become very large for small panel sizes. This is especially noticeable in Figures 6 and 7 where the cost is normalized to the panel area and weight, respectively.

As discussed earlier for the weight relationship, drawing conclusions using data from different airplane configurations can be easily misinterpreted. Even though the trends in Figures 5 and 6 show a relatively small scatter between the configurations (13% for 40000 in²), the trends are definitely a function of both the aircraft size and load levels. These trends are much greater in Figure 7 where the cost is normalized to the panel weight ("dollars per pound"). A design engineer may make decisions based on a curve fit of the five data points in Figure 7 and not the actual relationship between "dollars per pound" and size for a given configuration. In this case, the cost estimate could have significant error, leading to an incorrect design decision.
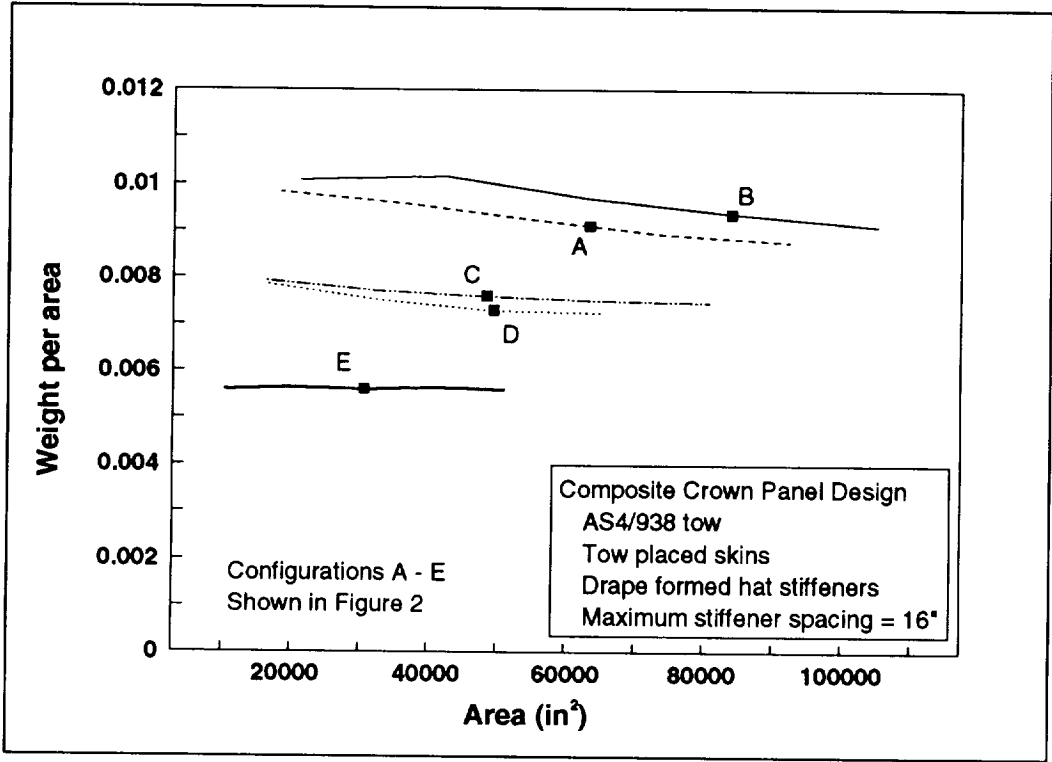
608

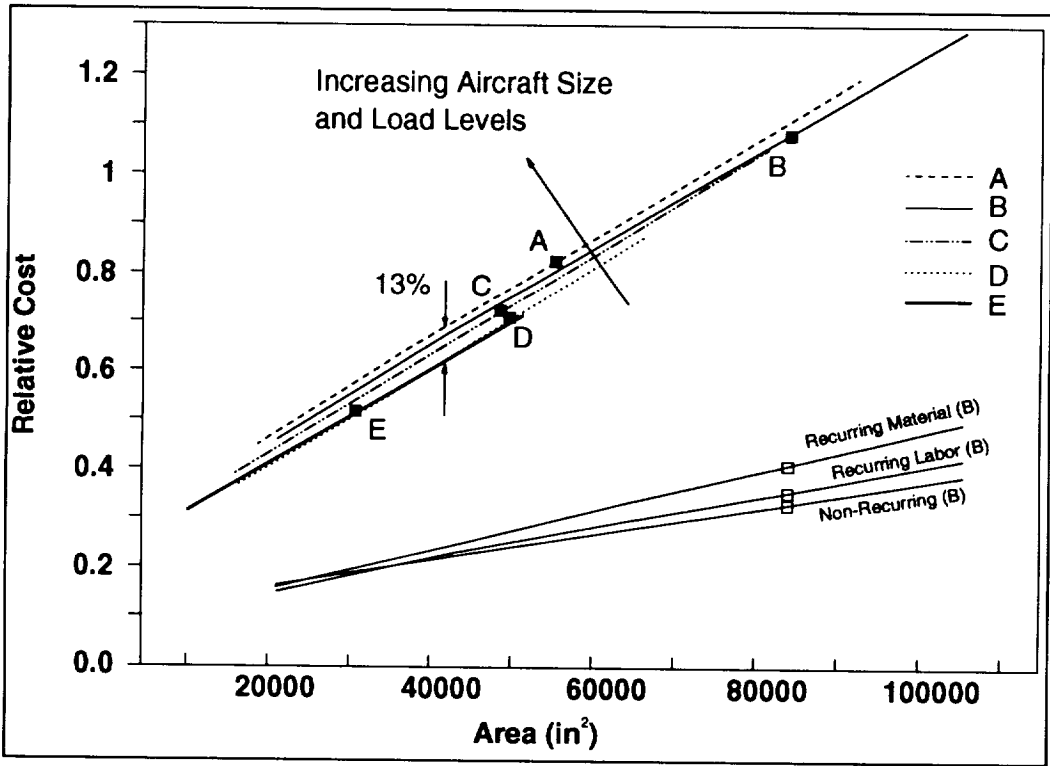**Figure 4:   Effect of Crown Panel Size on Weight**

Figure 5:   Effect of Crown Panel Size on Relative Cost

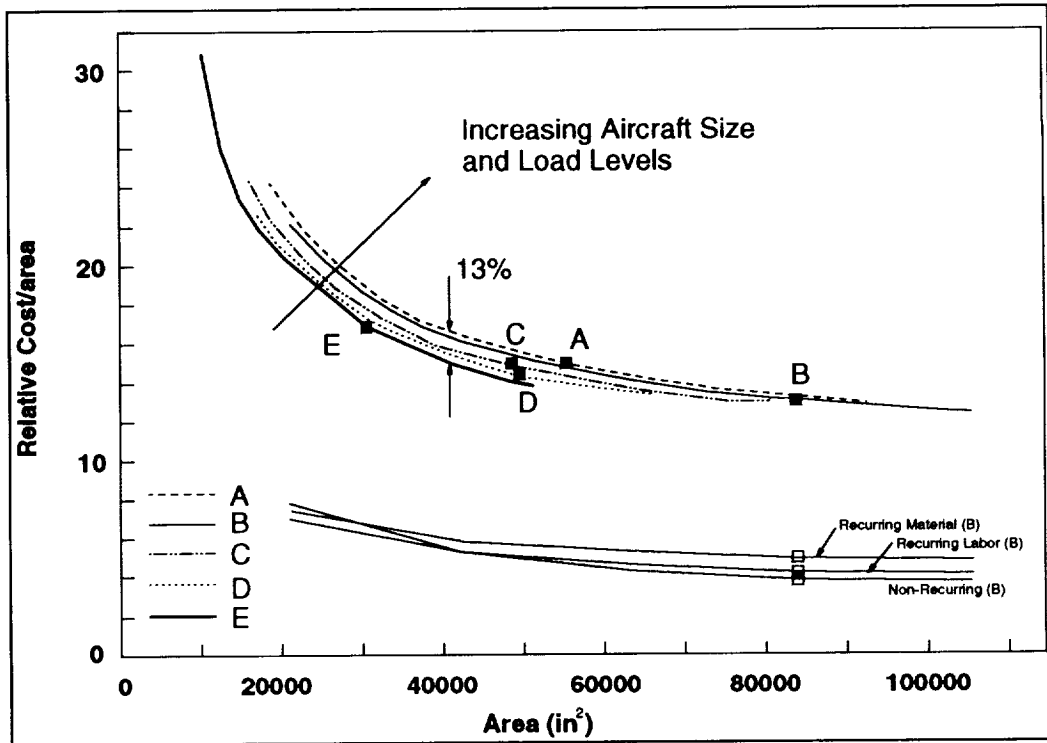**Figure 6: Effect of Crown Panel Size on Cost Normalized by Area**
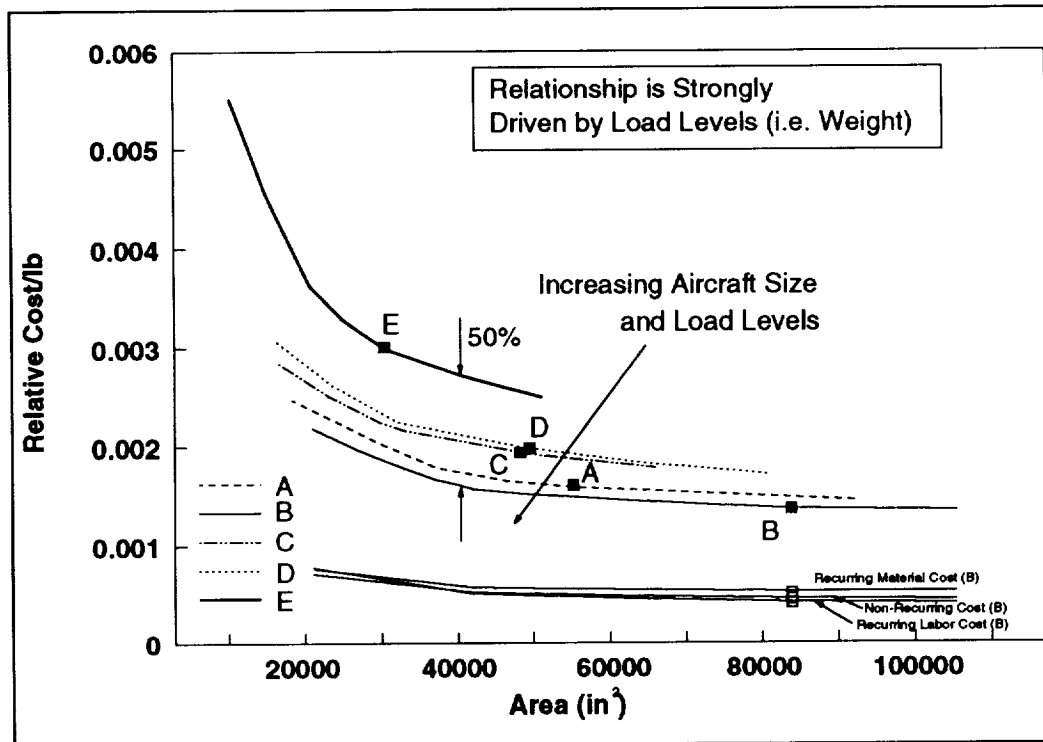

Figure 7: Effect of Crown Panel Size on Cost Normalized to Panel Weight

610

Cost and Weight Comparison of Composite and Metal Designs

It is often desirable to compare an emerging technology to an existing one to better understand the benefits and drawbacks of the new technology and where to invest future resources. In the world of composite materials, comparison is often made to the aluminum design, the backbone of today's commercial airframe industry. A comparison of aircraft configuration B is shown relative to an equivalent aluminum design in Figure 8. One of the most important points to note in the figure is that the metal design is not shown as a point but rather as a range of cost and weight. The aluminum design community is constantly looking for ways to lower the cost and/or weight of the design, as is the composite material community. The line in Figure 8 defining the "Current Aluminum Technology" represents the latitude that an airframe design engineer has based on a specific program's goals to minimize cost at the expense of weight, or visa versa. This decision relates to many economic factors. The location of an aluminum design within this range can greatly affect how the emerging composite design is compared.

For the composite design, the data point in Figure 8 represents a crown panel with hat stiffeners using a tow placed AS4/938 material system. This point represents the first hardware application of COSTADE. The details describing this data point are shown in Figure 9 along with some of imposed constraints. These constraints included higher axial loads and a larger crown panel size, based on updated load and geometry information since the original design [1]. Other constraints included that the aft laminate remain unchanged from the original design due to the existing material and laminate database and test plans on the Boeing ATCAS program. These load increases and laminate constraints are typical of real world design processes.

From this data point, the design engineer can trade a number of alternatives based on overall program direction to minimize cost, weight, or some combination of the two based on some level of dollars spent per pound of weight saved. The envelope drawn around the composite design point is an estimate of the range this particular crown panel design can be moved during cost and weight trade studies.

It is at this stage in the design process that many of the material decisions on aircraft programs are made. Material requirements for the tension load dominated crown panel may differ from the material requirements of other parts of the fuselage such as the keel [9]. For the particular crown panel application shown in Figure 8, the apparent toughness for large damage sizes exhibited by the tow placed AS4/938 material system suggests that it is a better choice than the resins typically described as tough [5, 6]. The medium modulus AS4 fibers were traded against higher modulus fibers which tend to be more expensive. The resulting weight savings associated with the higher modulus fiber was not sufficient to overcome the lower cost of the AS4 fiber for the assumed value of a pound of weight used in the current study.

Another material considered to have merit for crown panel applications is a hybrid material. In the current study, a hybrid material is defined as a material system that combines graphite and fiberglass fibers within a lamina to increase the damage tolerance. The resulting material has a lower effective modulus than an all graphite material, resulting in design criteria such as stiffness and stability to become more critical. As shown in Figure 8, hybrid applications tend to reduce cost due to the lower material cost, but tend to add weight due to an increased material density and the effect of stiffness and stability on the design. A lowering of these constraints tends to reduce the design weight without significant impact on cost.

The risk associated with the cost estimate is also shown in Figure 8. This is shown in two parts, the risk of the material price projections and the risk of emerging technologies not being developed. These risks are subjective, yet show that the costs are still at the same general cost level of the lower weight aluminum design space.
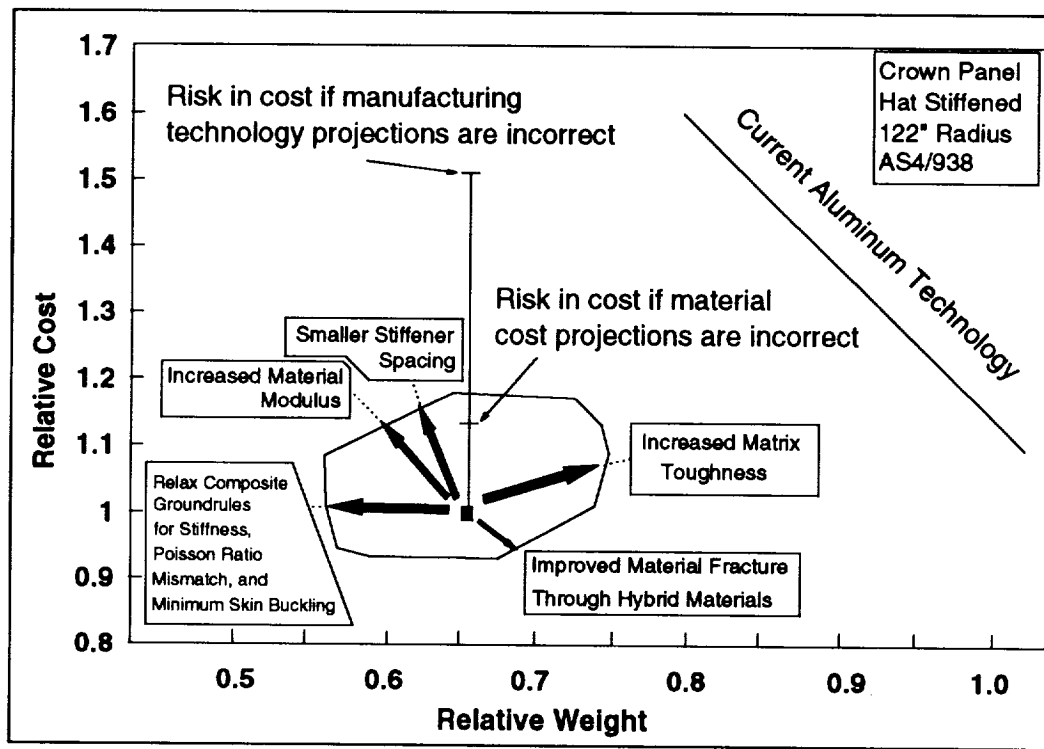
Figure 8: **Comparison of a Composite Crown Panel Design to an Equivalent Aluminum Design**

Weight Comparison of Composite and Aluminum Configurations

A comparison relating potential weight savings for composite structures to another design can be made by comparing the stress levels and densities of the aluminum design and the competing composite design. In general, for a given load, a higher stress level generally represents less required material in the design. If two materials are at an equivalent stress level, the weight savings would be directly proportional to their densities. In Figure 10, a comparison of the potential weight savings between the five composite crown panel designs and their aluminum counterparts is shown. Note that the shaded area represents the region of weight variance in the metal design described in Figure 8. It is important to note that the aluminum designs were based on many different assumptions over many years and may be different if designed in today's economy.

An assumed relationship between the potential weight savings and axial load level is shown in Figure 10 for the five configurations used in the current study. For a given diameter, a family of curves exists that relate the weight savings to the load level based on a maximum allowable aluminum stress level. If the maximum aluminum stress level is constant, the weight savings potential increases as the longitudinal load level increases, until a limiting composite stress level is reached.
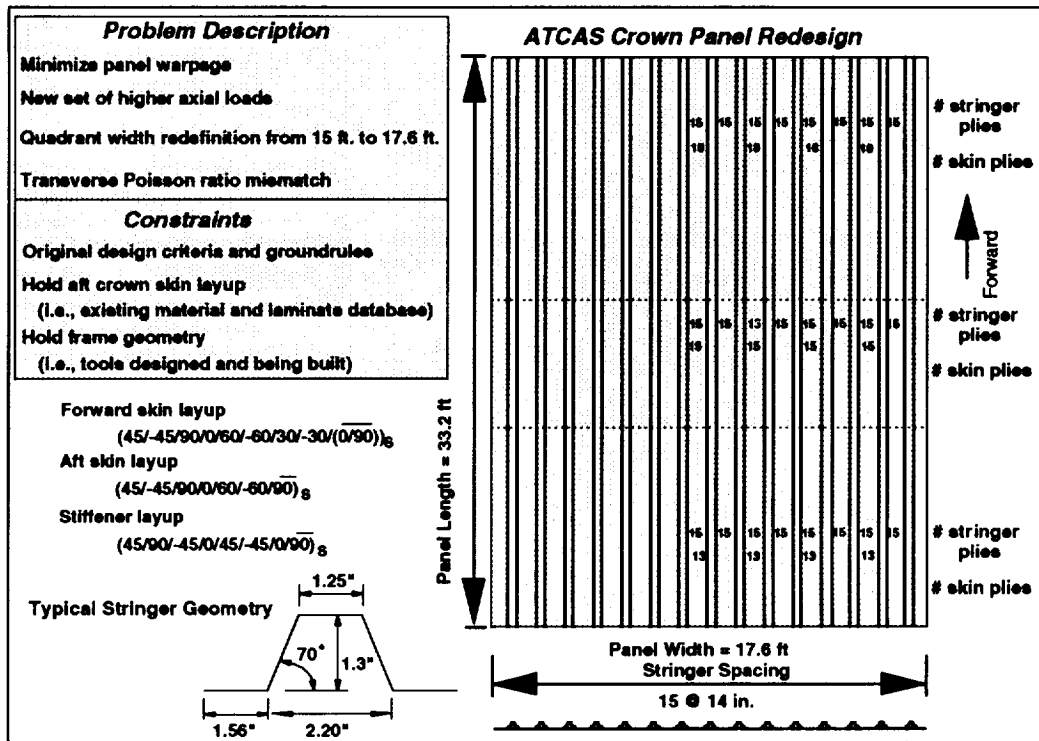
612

Figure 9: **Crown Panel Design Result Using COSTADE**

Note that a small, short fuselage tends to have low axial loads, making it appear unattractive for composite applications. However, the composite design was based on a material system chosen for minimum cost. When a weight emphasis is placed on the design, as was described in Figure 8, other material and criteria decisions may make the weight savings potential more attractive at this extreme end of the design envelope. In addition, longitudinal load levels logically increase as the fuselage length increases for a given diameter. For a typical family of airplanes, growth in the fuselage length typically occurs to satisfy customer requirements. A decision to choose a material system must take that growth into account. Finally, very light gage material is used in the aluminum design for the smaller aircraft. Additional weight, in the form of skin doublers (tear straps) under the frame are not reflected in the stress levels used to establish the trends in Figure 10. These factors will again tend to make the composite design more attractive.

PART 2: EFFECTS OF CONFIGURATION

To improve the buckling resistance of the skin panel, the introduction of a sandwich core to the skin, effectively creating a stiffened sandwich configuration, is compared to a more typical stiffened configuration. The tendency of sandwich core to reduce weight for larger spacings and its effect on cost is investigated. Previous studies [1, 4] discussed cost and weight trends for more typical stiffened skin designs. Decreased cost associated with fewer stiffeners (i.e. larger stiffener spacings) came at the penalty of increased weight. Optimum stiffener spacing was determined by the premium value (dollars per pound) that customer or manufacturer is willing to pay to save a pound of weight.

613

In this part of the current study, three configurations are considered: 1) a hat stiffened skin with no skin buckling allowed below Ultimate load, 2) a hat stiffened design with postbuckled skin, subject to a 38% minimum buckling requirement, 3) a hat stiffened sandwich panel constrained to be buckle resistant below Ultimate loads. A schematic of the three concepts and a photograph of some manufacturing demonstration articles representative of these concepts is shown in Figure 11.
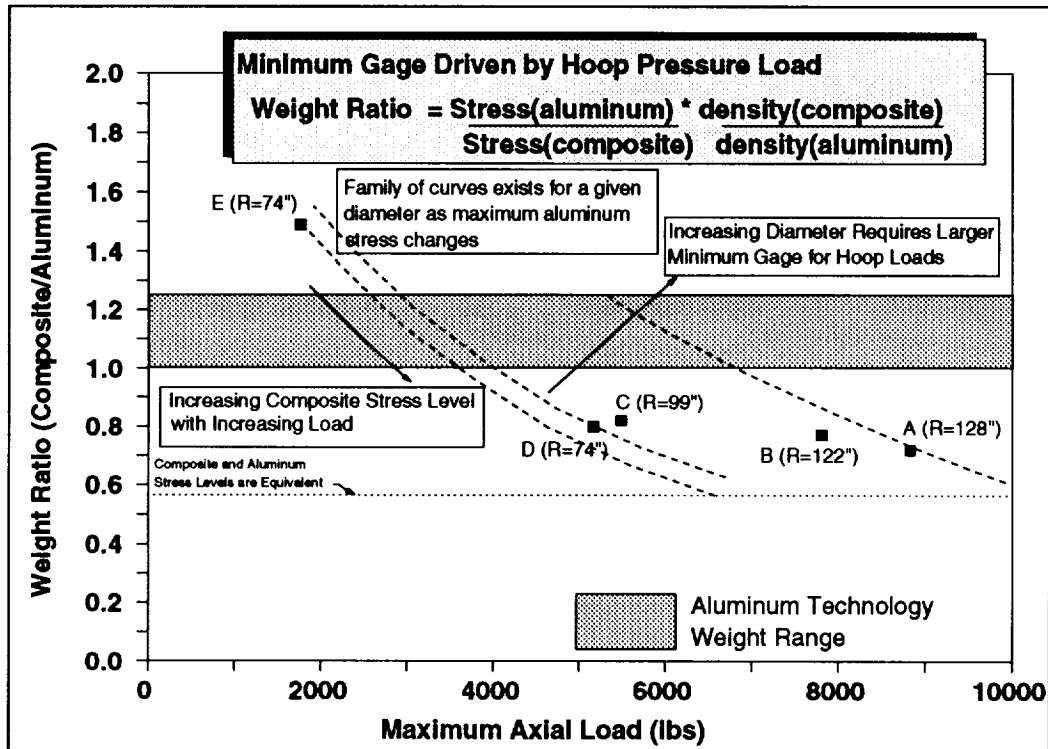

Figure 10: Comparison of a Composite Crown Panel Weight Savings Potential Relative to an Equivalent Aluminum Design

Effects of only applying stability constraints on stiffened sandwich design

Because skin stability plays such an important role in defining the crown panel stiffener spacing requirements, a simplified design study was undertaken that included only one compression load condition constrained by skin buckling, general instability, and local stiffener buckling. An additional constraint of face wrinkling was included for the stiffened sandwich designs. The three configurations shown in Figure 11 were optimized for minimum weight using the geometry of aircraft B from Figure 2. Two load levels, 1000 lb/in, typical of an aft crown panel compression load and 3000 lb/in, typical of a forward crown panel compression load, were considered. The cost values were based on the equations presented in Figure 1. This exercise was undertaken to provide information on whether the stiffened sandwich would be an attractive alternative without doing a fully constrained crown panel design.

The results of this study are shown in Figures 12 and 13. For the stiffened sandwich structure, a range of As/Ds values is used. This ratio couples the stiffener cross-sectional area (As) and the stiffener spacing (Ds). Note that in both figures, the weight is increasing with stiffener spacing while the cost is

614

decreasing with stiffener spacing. This is the same type of trend shown in [1, 4]. In these graphs, the value of saving weight is not accounted for in the relative cost relationships. Determining the benefits of lighter weight on the effective cost of different configurations would be needed to choose the most cost effective configuration.
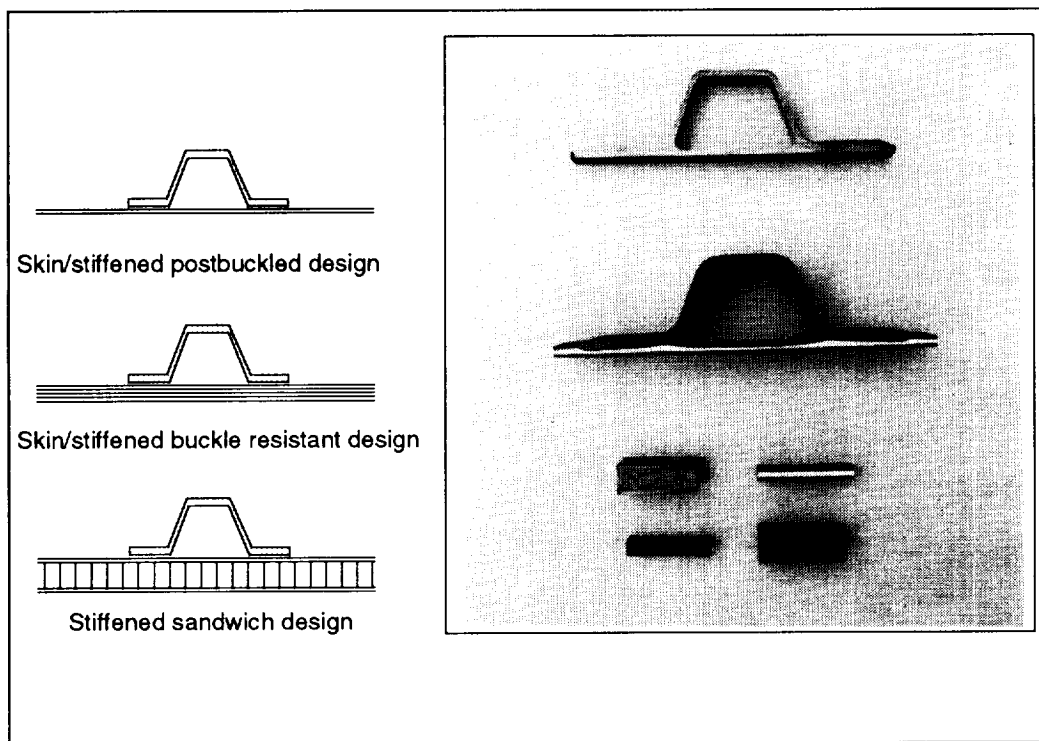

Figure 11: Configurations used in Stiffener Spacing Study

At 1000 lb/in, the postbuckled design is marginally heavier than the stiffened sandwich design, with the buckling resistant stiffened laminate significantly heavier. However, the postbuckled design was clearly more cost efficient than the other configurations at this load level. The cost saved by the lower weight of the stiffened sandwich did not overcome the additional cost of adding core to the design. The extra material used in the buckling resistant stiffened laminate made that configuration the heaviest. The cost, however, was essentially equivalent to the stiffened sandwich.

At 3000 lb/in compression, the relationships change. The best postbuckled design occurs when the stiffener spacing is small, yet it is not as weight efficient as the stiffened sandwich at any stiffener spacing. The cost effectiveness of the stiffened sandwich and the postbuckled design are essentially equivalent at this load level, with a slight advantage to the stiffened sandwich. The buckling resistant stiffened laminate is by far heavier and more expensive than the other two configurations.

The results of this initial, buckling only, study indicate that stiffened sandwich structure may be a good candidate to minimize the cost and weight of a stability dominated structure, given that the panel was subjected to relatively high compression loads. The next logical step, and the final part of the current study, is to determine the effects of stiffened sandwich given all of the design constraints outlined in Table 1 for a crown panel design.

615
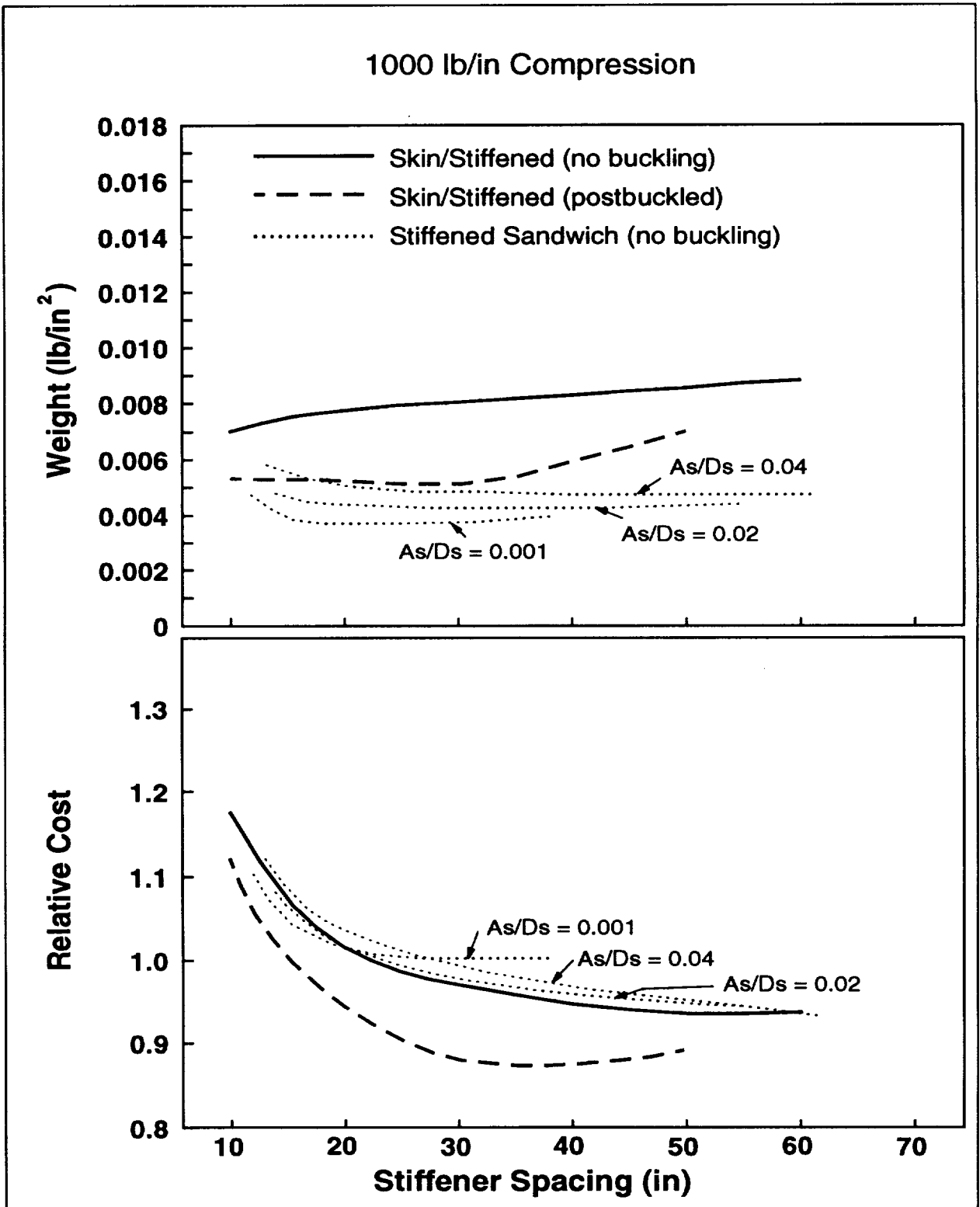
Figure 12: Cost and Weight Design Trends for Stiffened Panels Subjected Only to Buckling Constraints (1000 lb/in Compression)

616
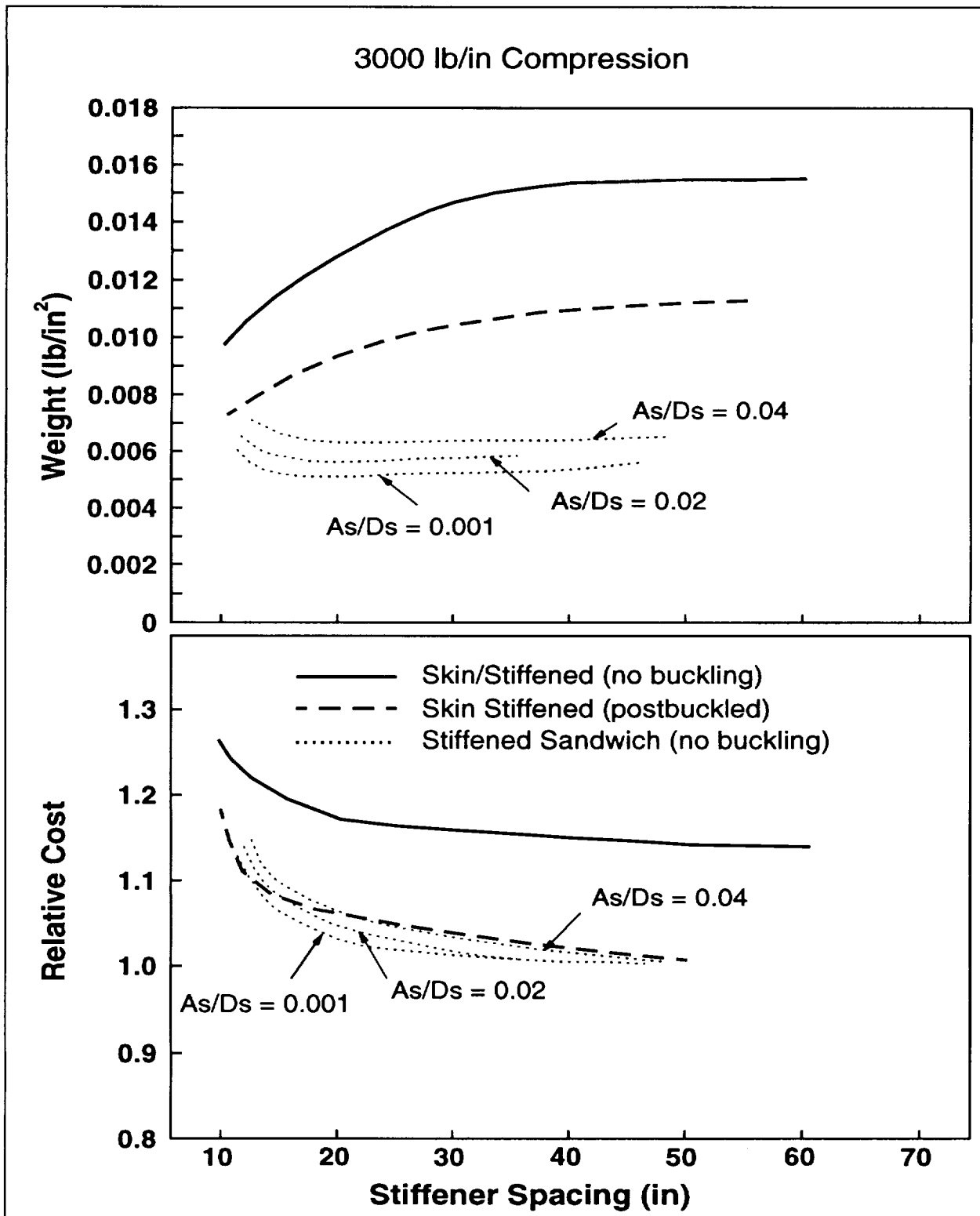
Figure 13: Cost and Weight Design Trends for Stiffened Panels Subjected Only to Buckling Constraints (3000 lb/in Compression)

Effects of Crown Panel Design Criteria on Stiffened Sandwich

In the last part of this study, a forward and an aft crown panel design are considered using the same three configurations presented in Figures 12 and 13. These designs include the effects of the criteria listed for the crown panel in Table 1. The costs do not reflect any additional value for weight savings.

Many of the analysis routines in the current version of the designer's cost model are based on simplified preliminary analysis design tools. Further development of some of these tools is ongoing and will be incorporated when complete. One of the analysis methods being developed is the tension damage tolerance assessment of a stiffened, orthotropic structure [5, 6, 10]. Currently, the cost model has a simplified damage tolerance routine that has proven to be inadequate for certain conditions. To address the effect of the stiffener spacing on the fully constrained design, some modifications to the tension damage tolerance analysis were made. It was assumed that up to a 20" axial damage size would be tolerated without any effect of load redistribution to the stiffening members for stiffener spacings larger than 20".

Other analysis routines that are to be added include a panel warpage assessment [11] and a stiffened and unstiffened sandwich analysis. Currently these are not yet incorporated. The sandwich analysis in this final part of the current study was calculated by a design engineer using the currently available design charts, spreadsheets, and lamination computer codes. It is interesting to note, and a big incentive for the cost model development, that the time needed to generate the analysis trends for the stiffened sandwich was on an order of magnitude longer than to develop similar trends using the cost model for both the buckling resistant and postbuckled hat stiffened panel designs. The many load cases and criteria that are checked in the process of developing a design can become cumbersome when doing an analysis by hand. The trends in time saved will be amplified even more when an entire panel is considered with many changing load levels. The blending function currently being developed for the cost model will address this situation.

The results of the fully constrained crown panel design for the lightly loaded (aft) and heavily loaded (forward) ends are shown in Figures 14 and 15, respectively. For small stiffener spacings, similar cost and weight trends to [1, 4] are observed for the stiffened laminate designs, with the post buckled design showing more cost and weight efficiency than the design constrained to resist buckling. At these smaller spacings, minimum skin buckling, axial and hoop damage tolerance, and axial stiffness tended to have the lowest margins of safety.

As stiffener spacings approached the frame spacing (22"), the revised damage tolerance analysis was implemented. In addition, the buckling mode shape of the skin between the stiffeners also approaches a critical point as the stiffener spacing approaches the same value as the frame spacing. Larger stiffener spacings affect the number of buckling waves across the skin bay between stiffeners. The required skin thickness to resist this buckling mode is such that the hoop damage tolerance is no longer critical for the larger stiffener spacings. The area labeled "transition zone" in the figure refers to the area where the critical design constraints are changing. Designs in this area are questionable in that small changes in any load or constraint may trigger different design constraints to be critical. Beyond this "transition zone," the design is driven by the axial damage tolerance and the buckling constraints. Little difference is seen between the postbuckled and buckle resistant stiffened skin designs in this region. For these larger stiffener spacings, postbuckling is no longer an effective way to save weight since a significant amount of material is needed to satisfy the minimum buckling constraint and the

618

stiffeners are less and less effective as the spacings increase. Reduced weight and cost as a function of stiffener spacing in this region is directly proportional to the reduction in the number of stiffeners.

The best stiffened sandwich design for both cost and weight is the limiting case of an unstiffened sandwich panel. The increased bending stiffness of the sandwich relieves the pressure effects at the notch tip and improves the buckling resistance of the skin. The addition of the core material, however, is a source of increased weight and significant cost. For the lightly loaded aft design, shown in Figure 14, the best sandwich design (large stiffener spacing) is both heavier and more expensive than the best postbuckled design (small stiffener spacing). The added axial and hoop loads along with the tension damage tolerance constraints require additional skin material in the sandwich beyond what is needed for the buckling constraint, resulting in somewhat different trends than shown in Figure 12 for stability only. The compression loads applied to both of these cases are similar.

For the more heavily loaded forward crown design shown in Figure 15, similar trends exist as were shown in Figure 14. The major difference is that the relationship between the best postbuckled design and sandwich design is much closer, suggesting that the higher loads make the sandwich design more favorable, a trend consistent with the buckling only results discussed earlier. For keel applications where very high compression loads occur, this trend would tend to favor the sandwich design [9].

Conclusions

A design study investigating the effects of size and configuration of a composite crown panel was undertaken. Results indicate that both aircraft geometry, load intensities, and material decisions can greatly affect the cost and weight of the designs. Larger crown panel sizes tended to be more economical. Comparison to aluminum technology utilized a concept of comparing feasible design regions, since both composite and aluminum designs can vary depending on weight and cost targets. The range of weight and cost in which a feasible design can be found is based on decisions that an engineer can make regarding material, geometry, and criteria.

The effect of stiffened sandwich, as compared to postbuckled and buckle resistant structure, suggests that for stability dominated designs, a stiffened sandwich concept can be weight effective without significant cost differences. This trend becomes more attractive for larger compression loads. When the remaining load conditions and design constraints typical of a crown panel are applied, the trend changes such that a sandwich structure without stiffeners is still not as efficient in both cost and weight as a postbuckled design. The trends suggest that as the load increases, the difference between these two concepts is less. A stiffened sandwich design may be a benefit for more heavily loaded compression panels.

The benefits of a design cost model in this type of study are evident. For the stiffened sandwich study, design constraints were not yet incorporated into the model, forcing a design engineer to run the trade studies using conventional analysis and available design tools. For the fully constrained crown panel design, the time needed to complete the trade study for the stiffened sandwich as compared to both stiffened laminate designs was an order of magnitude longer. The understanding gained by seeing the effects of the design on both cost and weight is a great benefit to an engineer. Further development of the cost model to include the sandwich constraints, along with warpage constraints, improved damage tolerance analysis, a blending function to handle load variations, and a more general cost framework are ongoing.
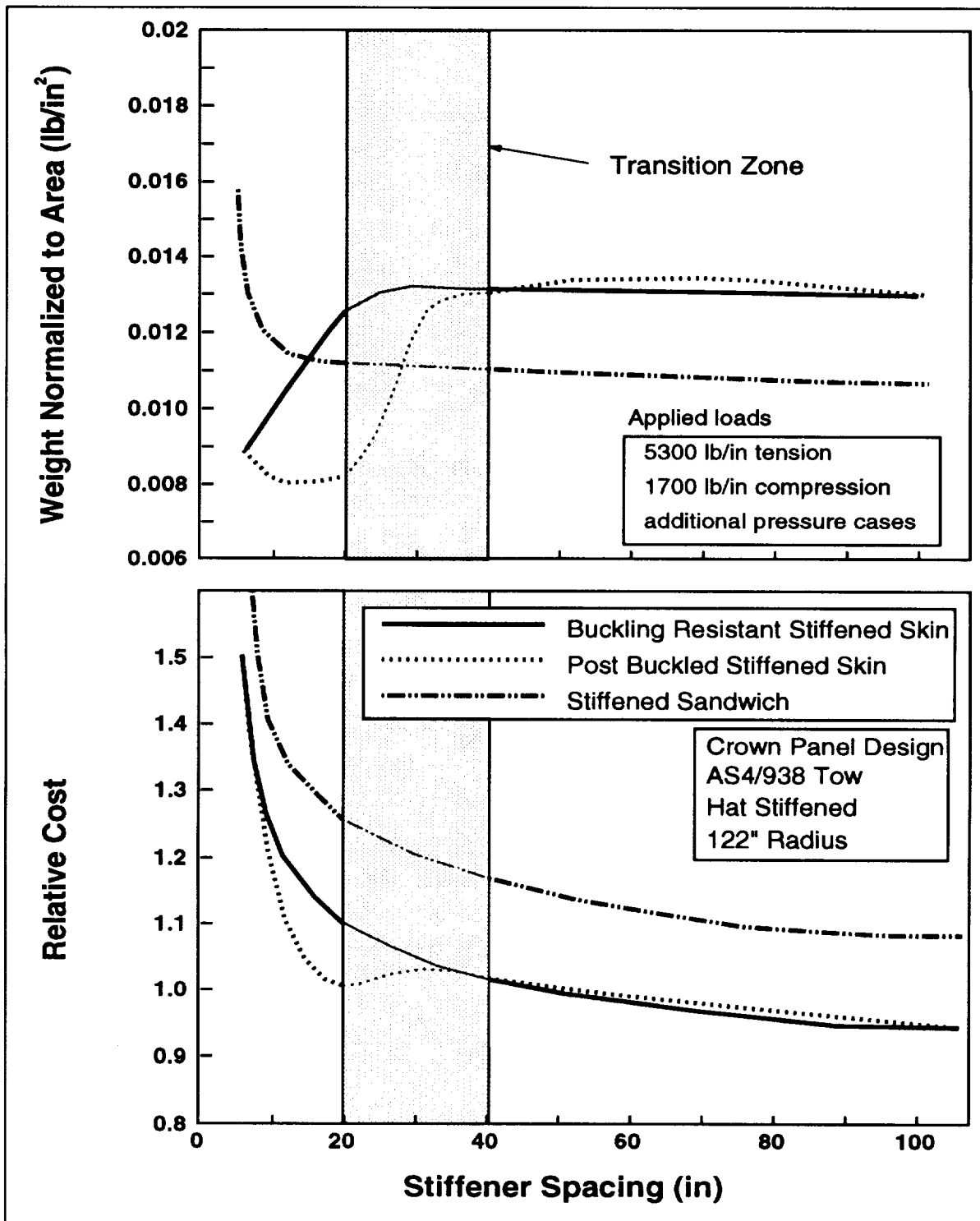
Figure 14: Cost and Weight Design Trends for Stiffened Panels
Subjected to Crown Panel Design Constraints (Lightly
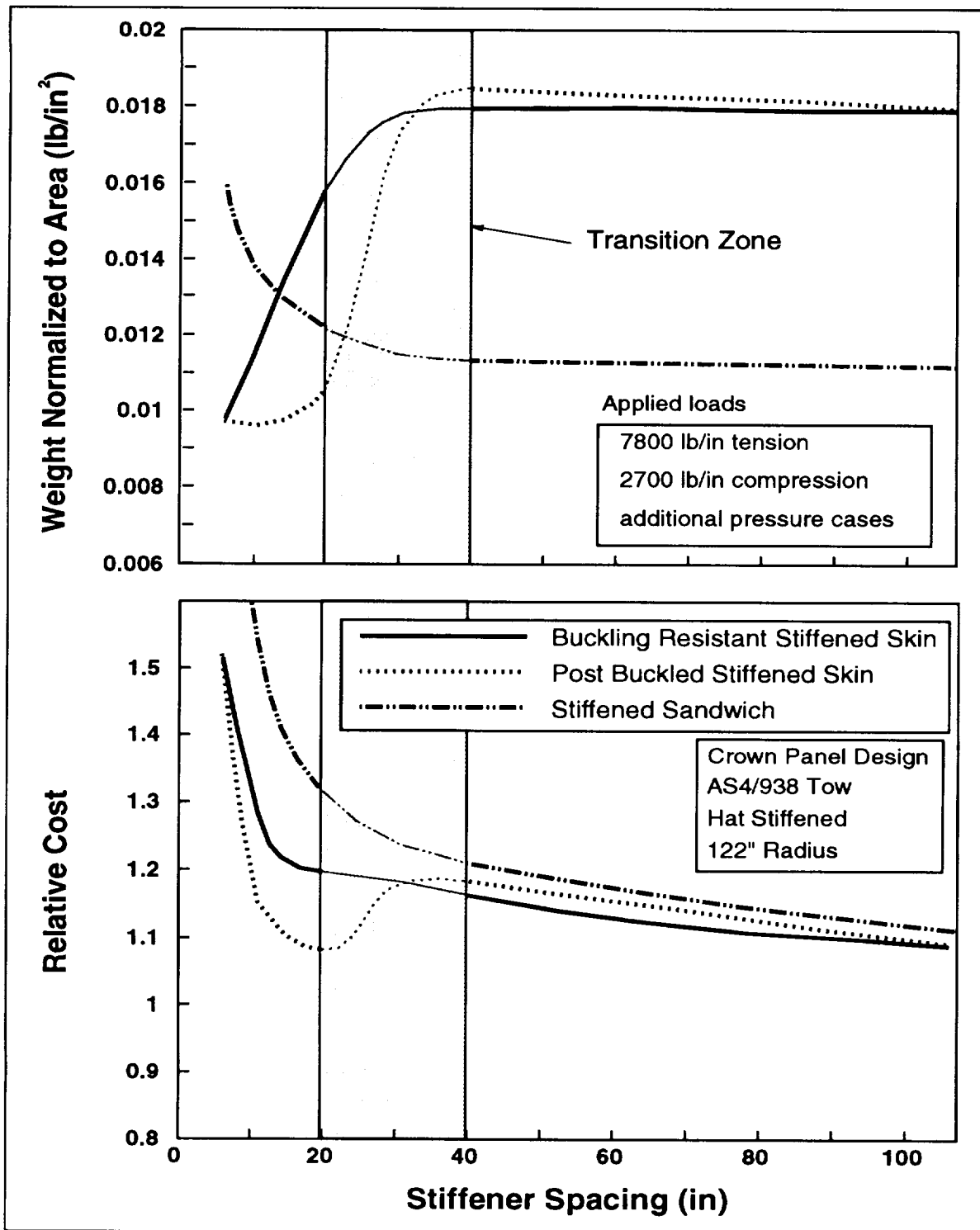Loaded Aft Crown Panel)

620

Figure 15: Cost and Weight Design Trends for Stiffened Panels
Subjected to Crown Panel Design Constraints (Heavily
Loaded Forward Crown Panel)

References

1. G. D. Swanson, L. B. Ilcewicz, T. H. Walker, D. Graesser, M. Tuttle, Z. Zabinsky: "Local Design Optimization for Composite Transport Fuselage Crown Panels," In Proceedings of the Ninth DoD/NASA/FAA Conference on Fibrous Composites in Structural Design, Lake Tahoe, NV, FAA Publication, 1991.

2. W. T. Freeman, L. B. Ilcewicz, G. D. Swanson, T. Gutowski: "Designer's Unified Cost Model," In Proceedings of the Ninth DoD/NASA/FAA Conference on Fibrous Composites in Structural Design, Lake Tahoe, NV, FAA Publication, 1991.

3. Z. Zabinsky, M. Tuttle, D. Graesser, G. D. Swanson, L. B. Ilcewicz: "Multi-Parameter Optimization Tool for Low-Cost Commercial Fuselage Crown Designs," First NASA Advanced Composites Technology (ACT) Conference, October 30 -November 1, 1990, NASA CP-3104.

4. C. Kassapoglou, A. J. DiNicola, and J. C. Chou: "Innovative Fabrication Processing of Advanced Composite Materials Concepts for Primary Aircraft Structures," NASA Contractor Report 189558, February 1992.

5. T. H. Walker, W. B. Avery, L. B. Ilcewicz, C. C. Poe, C. E. Harris: "Tension Fracture of Laminates for Transport Fuselage, Part 1: Material Screening," In Proceedings of the Ninth DoD/NASA/FAA Conference on Fibrous Composites in Structural Design, Lake Tahoe, NV, FAA Publication, 1991.

6. T. H. Walker, W. B. Avery, L. B. Ilcewicz, C. C. Poe, C. E. Harris: "Tension Fracture of Laminates for Transport Fuselage, Part 2: Large Notches," In Proceedings of the Third NASA Advanced Composites Technology (ACT) Conference, Long Beach, CA, June 8 - 11, 1992, NASA CP-3178.

7. L. B. Ilcewicz, T. H. Walker, K. S. Willden, G. D. Swanson, G. Truslove, and C. L. Pfahl: "Application of a Design-Build-Team Approach to Low Cost and Weight Composite Fuselage Structure," NASA Contractor's Report 4418, 1991.

8. T. H. Walker, P. J. Smith, G. Truslove, K. S. Willden, S. L. Metschan, C. L. Pfahl: "Cost Studies for Commercial Fuselage Crown Design," In Proceedings of the Ninth DoD/NASA/FAA Conference on Fibrous Composites in Structural Design, Lake Tahoe, NV, FAA Publication, 1991.

9. B. W. Flynn, M. R. Morris, S. L. Metschan, G. D. Swanson, P. J. Smith, K. H. Griess, M. R. Schramm, R. J. Humphrey: "Global Cost and Weight Evaluation of Fuselage Keel Design Concepts," Third NASA Advanced Composite Technology Conference, NASA CP-3178.

10. L. B. Ilcewicz, P. J. Smith, and R. E. Horton: "Advanced Composite Fuselage Technology," Third NASA Advanced Composite Technology Conference, NASA CP-3178.

11. G. Mabson, B. W. Flynn, R. Lundquist, G. D. Swanson, P. Rupp: "Dimensional Stability of Curved Panels with Cocured Stiffeners and Cobonded Frames," Third NASA Advanced Composite Technology Conference, NASA CP-3178.

Session VII

AUTOMATED FIBER PLACEMENT TECHNOLOGY

Session Chairman: William T. Freeman, Jr.
NASA Langley Research Center