REPORT DOCUMENTATION PAGE

Form Approved
OMB No. 0704-0188

1. AGENCY USE ONLY *(Leave blank)*

2. REPORT DATE
December 1991

3. REPORT TYPE AND DATES COVERED
Technical Paper

4. TITLE AND SUBTITLE
Optimization of Composite Sandwich Cover Panels Subjected to Compressive Loadings

5. FUNDING NUMBERS
WU 505-63-50-08

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7. PERFORMING ORGANIZATION NAME(S) AND ADDRESS(ES)
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8. PERFORMING ORGANIZATION REPORT NUMBER
L-16942

9. SPONSORING/MONITORING AGENCY NAME(S) AND ADDRESS(ES)
National Aeronautics and Space Administration
Washington, DC 20546-0001

10. SPONSORING/MONITORING AGENCY REPORT NUMBER
NASA TP-3173

11. SUPPLEMENTARY NOTES

12a. DISTRIBUTION/AVAILABILITY STATEMENT
Unclassified–Unlimited

Subject Category 24

12b. DISTRIBUTION CODE

13. ABSTRACT *(Maximum 200 words)*
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14. SUBJECT TERMS
Composites; Optimization; Sandwich structures; Structural design

15. NUMBER OF PAGES
10

16. PRICE CODE
A03

17. SECURITY CLASSIFICATION OF REPORT
Unclassified

18. SECURITY CLASSIFICATION OF THIS PAGE
Unclassified

19. SECURITY CLASSIFICATION OF ABSTRACT

20. LIMITATION OF ABSTRACT

Abstract

An analysis and design method is presented for the design of composite sandwich cover panels that includes transverse shear effects and damage tolerance considerations. This method is incorporated into a sandwich optimization computer program entitled SANDOP. As a demonstration of its capabilities, SANDOP is used in the present study to design optimized composite sandwich cover panels for transport aircraft wing applications. The results of this design study indicate that optimized composite sandwich cover panels have approximately the same structural efficiency as stiffened composite cover panels designed to satisfy identical constraints. The results also indicate that inplane stiffness requirements have a large effect on the weight of these composite sandwich cover panels at higher load levels. Increasing the maximum-allowable strain and the upper percentage limit of the 0° and ±45° plies can yield significant weight savings. The results show that the structural efficiency of these optimized composite sandwich cover panels is relatively insensitive to changes in core density. Thus, core density should be chosen by criteria other than minimum weight (e.g., damage tolerance, ease of manufacture, etc.).

Introduction

Composite materials are being widely considered for application to heavily loaded primary aircraft structures such as wing cover panels. To date, much of the research conducted on aircraft wing cover panels has focused on stiffened plate designs. The analysis of stiffened cover panels is well-understood, and tools exist to perform analysis and design optimization of these panels (refs. 1 and 2). Relatively less emphasis, however, has been placed on cover panels of sandwich construction.

The present paper describes an analysis and design method that has been developed for composite sandwich cover panels loaded in compression, including damage tolerance considerations. The analysis and appropriate design variables have been incorporated into a constrained sandwich optimization program entitled SANDOP. This program utilizes weight per unit area as the objective function to be minimized subject to several constraints. SANDOP is written in sizing and optimization language (SOL), a high-level computer language developed specifically for the application of numerical optimization methods to design procedures. (See refs. 3 and 4.)

As a practical demonstration of SANDOP, composite sandwich cover panels for transport aircraft wing applications have been designed subject to constraints appropriate for this kind of structure. These composite sandwich cover panels are compared with composite stiffened cover panels that were designed to satisfy identical constraints using a panel analysis and sizing code (PASCO). (See refs. 1 and 2.) Furthermore, the effect of changing the constraint values on the structural efficiency of these composite sandwich cover panels is investigated.

Symbols

A_{mn}	modal amplitudes (see eq. (4))
A_{11}, A_{66}	inplane stiffnesses of cover panel
$A_{11,\mathrm{min}}$, $A_{66,\mathrm{min}}$	minimum-required inplane stiffness of facesheets
$A_{11,\mathrm{min,bl}}$, $A_{66,\mathrm{min,bl}}$	baseline values of minimum-required inplane stiffness of facesheets
a	cover-panel length (see fig. 1)
b	cover-panel width (see fig. 1)
D_{Qx}, D_{Qy}	transverse shear stiffnesses of cover panel (see eqs. (2) and (3), respectively)
D_{11}, D_{12}, D_{22}, D_{66}	bending stiffnesses of cover panel
E_{c_z}	sandwich core modulus in z-direction
E_f	effective facesheet modulus in longitudinal direction
E_L, E_T	lamina modulus in longitudinal and transverse directions, respectively
G_{LT}	lamina shear modulus

$G_{xz},\ G_{yz}$	sandwich core transverse shear modulus in x- and y-directions, respectively
k_A	scaling factor for minimum-required inplane stiffness
$L_0,\ L_{45},\ L_{90}$	lower percentage limit of 0°, ±45°, and 90° plies, respectively
m	number of longitudinal half-waves for cover-panel buckling mode
N_x	applied longitudinal stress resultant (see fig. 1)
N_x^b	longitudinal stress resultant at buckling
N_x^w	longitudinal stress resultant for facesheet symmetric wrinkling
N_{xy}	applied shear stress resultant
N_y	applied transverse stress resultant
n	number of transverse half-waves for cover-panel buckling mode
t_c	core thickness (see fig. 1)
t_f	facesheet thickness (see fig. 1)
$t_0,\ t_{45},\ t_{90}$	thickness of facesheet 0°, ±45°, and 90° plies, respectively
$U_0,\ U_{45},\ U_{90}$	upper percentage limit of 0°, ±45°, and 90° plies, respectively
W	weight per unit area of cover panel
w	out-of-plane displacement of cover panel
x, y, z	Cartesian coordinate system (see fig. 1)
ε_x	longitudinal strain of cover panel
$\varepsilon_{x,\max}$	maximum-allowable longitudinal strain
ν_{LT}	lamina major Poisson's ratio
$\rho_{C/E}$	carbon/epoxy material density
ρ_{core}	core density

Analysis, Design, and Optimization Methodology

This section describes the analysis and design used in this study of sandwich cover panels with composite material facesheets. Dominant response mechanisms for composite sandwich cover panels are presented and analyzed. The analysis is combined with an optimization procedure to obtain structurally efficient designs. The objective function, design variables, and constraints for the structural optimization problem are explained in this section.

Figure 1. Panel geometry and loading. All edges simply supported.

The sandwich cover panel considered in the present study is shown in figure 1. This sandwich panel is rectangular, flat, and simply supported on all four edges. A single, uniform longitudinal stress resultant N_x is applied at opposite ends of the panel as shown in figure 1. The facesheets are symmetric composite laminates with specially orthotropic material symmetry. The sandwich core also exhibits specially orthotropic material symmetry in its transverse shearing stiffnesses. The corresponding transverse shearing stiffnesses of the core are denoted by G_{xz} and G_{yz}. The principal directions of the core material are assumed to coincide with the x and y coordinate directions. (See fig. 1.)

Response Mechanisms

Three response mechanisms are included in SANDOP for designing composite sandwich cover panels loaded in compression. These mechanisms are global buckling (including transverse shear deformation), symmetric facesheet wrinkling, and material failure. A brief description of each of these mechanisms is presented as follows:

Global buckling. The equation governing global buckling of sandwich panels, including transverse shear effects, is derived in reference 5 and is given by

$$
\begin{aligned}
&\left[D_{11}D_{66}\left(\frac{1}{D_{Qy}}+\frac{N_x}{D_{Qx}D_{Qy}}\right)\right]\frac{\partial^6 w}{\partial x^6} \\
&+\left[\frac{D_{11}D_{66}}{D_{Qx}}+\left(D_{11}D_{22}-D_{12}^2-2D_{12}D_{66}\right)\left(\frac{1}{D_{Qy}}+\frac{N_x}{D_{Qx}D_{Qy}}\right)\right]\frac{\partial^6 w}{\partial x^4\partial y^2} \\
&+\left[D_{22}D_{66}\left(\frac{1}{D_{Qy}}+\frac{N_x}{D_{Qx}D_{Qy}}\right)+\left(\frac{D_{11}D_{22}-D_{12}^2-2D_{12}D_{66}}{D_{Qx}}\right)\right]\frac{\partial^6 w}{\partial x^2\partial y^4} \\
&+\left[\frac{D_{22}D_{66}}{D_{Qx}}\right]\frac{\partial^6 w}{\partial y^6}-\left[D_{11}+N_x\left(\frac{D_{11}}{D_{Qx}}+\frac{D_{66}}{D_{Qy}}\right)\right]\frac{\partial^4 w}{\partial x^4} \\
&-\left[2(D_{12}+2D_{66})+N_x\left(\frac{D_{22}}{D_{Qy}}+\frac{D_{66}}{D_{Qx}}\right)\right]\frac{\partial^4 w}{\partial x^2\partial y^2}-D_{22}\frac{\partial^4 w}{\partial y^4}+N_x\frac{\partial^2 w}{\partial x^2}=0 \qquad (1)
\end{aligned}
$$

where the transverse shear stiffnesses for an orthotropic core material D_{Qx} and D_{Qy} are given in reference 6 by, respectively,

$$D_{Qx}=G_{xz}\frac{(t_c+t_f)^2}{t_c} \qquad (2)$$

and

$$D_{Qy}=G_{yz}\frac{(t_c+t_f)^2}{t_c} \qquad (3)$$

Solutions to the buckling equation for sandwich panels are determined directly by assuming a buckling mode shape that satisfies both the differential equation (eq. (1)) and the boundary conditions (simply supported on all four edges). A buckling mode shape that meets this criterion is expressed as

$$w=A_{mn}\ \sin\left(\frac{m\pi x}{a}\right)\ \sin\left(\frac{n\pi y}{b}\right) \qquad (4)$$

where

$$m=1,2,3,\ \ldots \quad (0\le x\le a)$$
$$n=1,2,3,\ \ldots \quad (0\le y\le b)$$

Substituting this mode shape into equation (1) yields a homogeneous linear algebraic equation that depends on the wave numbers m and n, and thus constitutes an eigenvalue problem. For nontrivial solutions, the resulting equation can be solved for N_x as a function of m and n. The global buckling stress resultant N_x^b is obtained by minimizing N_x with respect to m and n.

This formulation for global buckling includes shear crimping as a response mechanism for sandwich plates. Shear crimping is given by the degenerate case of global buckling for which the wave parameter m is very large.

Facesheet wrinkling. Another stability-related response mechanism for sandwich structures is facesheet wrinkling. For this mechanism, the facesheets buckle locally with a wavelength of the same order as the thickness of the sandwich core. Facesheet wrinkling can be symmetric or antisymmetric in form as shown in figure 2. In the present study, only symmetric facesheet wrinkling is included.

Since wrinkling in sandwich panels with honeycomb cores is usually of the symmetric type (ref. 7), the current wrinkling analysis is valid for honeycomb cores. The current wrinkling analysis may not be valid for sandwich panels with foam cores since they may buckle in an antisymmetric wrinkling mode.

Figure 2. Symmetric and antisymmetric facesheet wrinkling.

The equation used in the present study to determine the onset of symmetric facesheet wrinkling (ref. 7) is given by

$$N_x^w = 0.67 t_f E_f \left(\frac{E_{c_z} t_f}{E_f t_c} \right)^{1/2} \tag{5}$$

Material failure. For a given panel design, the facesheet material may fail before the onset of either of the stability mechanisms previously discussed. Material failure is determined by specifying a maximum-allowable longitudinal strain criterion. Specifically, the onset of material failure is assumed to occur when the axial strain ε_x exceeds a maximum strain value $\varepsilon_{x,\max}$. This maximum strain value is based on a lower limit compression-after-impact failure strain of the composite facesheet that was experimentally determined. The use of this allowable strain criterion implicitly incorporates a damage tolerance constraint into the design process.

Objective Function and Design Variables

Structural efficiency is defined by a minimum cover-panel weight for the given design loads. The objective function used in this study is the weight per unit area W of the cover panel.

The design variables used in this study are classified as either facesheet design variables or core design variables. The composite facesheets are considered to be homogeneous through the thickness and to consist of 0°, ±45°, and 90° plies only. These two assumptions allow the facesheets to be completely defined by using only the three design variables t_0, t_{45}, and t_{90}, which are the thicknesses of the 0°, ±45°, and 90° plies, respectively, in the facesheet laminates. Both facesheets are symmetric, specially orthotropic, and identical. The sandwich core is defined by the two design variables t_c and ρ_{core}, the core thickness and core density, respectively. The three core material properties used in the analysis, G_{xz}, G_{yz}, and E_{c_z}, are determined by the core type, core material, and core density.

Constraints

The constraints used to perform the optimization are based on the response mechanisms for sandwich panels previously described and on current design practices for composite facesheets and sandwich cores. A brief description of the constraints is presented as follows:

Response mechanism constraints. The cover-panel designs for the present study are constrained to have buckling and wrinkling stress resultants N_x^b and N_x^w greater than the applied stress resultant N_x. In addition, the longitudinal strain ε_x due to the applied N_x is constrained to be less than the maximum-allowable longitudinal strain $\varepsilon_{x,\max}$. This maximum-allowable strain corresponds to the inherent residual compressive strength for an impact-damaged composite laminate, and it is an empirical value.

Facesheet and core constraints. Constraints are placed on the laminate and the inplane stiffnesses of the composite facesheets. The laminate is constrained by placing upper and lower limits on the relative thicknesses of each ply group (plies with the same orientation) with respect to the total facesheet thickness. These constraints are written as

$$L_0 < \frac{t_0}{t_0 + t_{45} + t_{90}} < U_0 \tag{6}$$

$$L_{45} < \frac{t_{45}}{t_0 + t_{45} + t_{90}} < U_{45} \tag{7}$$

$$L_{90} < \frac{t_{90}}{t_0 + t_{45} + t_{90}} < U_{90} \tag{8}$$

where L and U denote the lower and upper percentage limits, respectively, for a given ply group. These constraints are used to exclude laminate designs that are dominated by one ply orientation. Practical laminate designs are often required to have fibers oriented in several directions to satisfy requirements not specifically considered herein, e.g., repair requirements (ref. 8).

The composite facesheet designs are also required to satisfy minimum inplane stiffness constraints. The facesheet stiffnesses A_{11} and A_{66} are required to be greater than some specified minimum stiffnesses $A_{11,\min}$ and $A_{66,\min}$, respectively. The minimum stiffnesses used in this study are discussed in the "Results and Discussion" section.

The sandwich core density is constrained to a range of densities that are practical for aircraft cover panels. Upper and lower limits for core density are specified for the present study.

SANDOP

The design and optimization method described above has been incorporated into a sandwich optimization computer program entitled SANDOP. This program is written in sizing and optimization language (SOL), a high-level computer language developed specifically for the application of numerical optimization methods to design problems. (See refs. 3

and 4.) SANDOP allows the user to optimize composite sandwich cover panels. The input parameters available to the user are the facesheet and core material properties, the panel dimensions, the design stress resultant N_x, and the parameter values for the various constraints. SANDOP can be modified to expand the present analysis and constraints.

Results and Discussion

As a demonstration of the capabilities of SANDOP, the program was used to design optimized composite sandwich cover panels for transport aircraft wing applications. These optimized sandwich panels are compared with stiffened composite cover panels designed to satisfy identical constraints. The effect of the constraints on the optimal design is also investigated.

Baseline Design

A baseline set of design parameters and constraints was selected to establish a reference design for subsequent comparison. These design parameters and constraints are typical of those used to design sandwich cover panels for transport wing applications. All the cover panels considered in the study are assumed to be square, with 30-in. side dimensions. Cover panels were optimized for load levels ranging from 3 000 to 24 000 lb/in.

The unidirectional composite material properties used for the facesheets are those of Hercules IM6 carbon fibers and American Cyanamid 1808I epoxy interleaved material given in reference 9 as shown in table 1. The core material used in this study is Hexcel 5052 aluminum-alloy hexagonal honeycomb, whose properties were obtained from reference 10; a few typical values are shown in table 2. Since core material properties are available only for specific values of the core density, SANDOP interpolates these data to obtain core properties at densities other than those given in reference 10.

Table 1. Properties of IM6/1808I Carbon/Epoxy Tape[a]

Longitudinal Young's modulus, E_L, Msi	18.5
Transverse Young's modulus, E_T, Msi	1.09
Shear modulus, G_{LT}, Msi	0.70
Major Poisson's ratio, ν_{LT}	0.33
Density, $\rho_{C/E}$, lb/in^3	0.058

[a]Values were obtained from reference 9, except for the density which is estimated.

Table 2. Properties of Hexcel 5052 Aluminum-Alloy Honeycomb Core[a]

ρ_{core}, lb/ft^3	E_{c_z}, ksi	G_{xz}, ksi	G_{yz}, ksi
1.0	10.0	12.0	7.0
6.0	235.0	96.0	40.5
9.5	420.0	105.0	53.0

[a]Values were obtained from reference 10.

Table 3. Constraint Values of Baseline Design

$\varepsilon_{x,\max} = 0.0045$ in/in.	
$A_{11,\min} = f(N_x)$ $A_{66,\min} = g(N_x)$	See figure 3
$L_0 = 0.125$ $L_{45} = 0.125$ $L_{90} = 0.125$	$U_0 = 0.375$ $U_{45} = 0.375$ $U_{90} = 0.375$
1.0 lb/ft^3 $< \rho_{\text{core}} <$ 9.5 lb/ft^3	

The constraints used for the baseline design are shown in table 3. The minimum-required inplane stiffnesses $A_{11,\min}$ and $A_{66,\min}$ are functions of the load level as indicated by figure 3. This correlation between the minimum-required inplane stiffness and N_x is based on historical data for transport aircraft wings that were presented in reference 11. The limits on the relative thickness of each ply group, with respect to the total facesheet thickness, is based on the recommendations of reference 8. These recommendations are designed to yield laminates suitable for bolted and riveted joints. A maximum-allowable strain of 0.0045 in/in. was selected to provide acceptable damage tolerance capability consistent with current composite material systems.

Figure 3. Minimum-required inplane stiffnesses for cover panels (ref. 11).

Figure 4. Weight comparison between sandwich and stiffened-plate composite cover panels.

Comparison With Stiffened Cover Panels

The structural efficiency of composite sandwich cover panels optimized with SANDOP is shown in figure 4. In this figure the weight per unit area of the cover panel W is shown as a function of N_x. In addition, the structural efficiency of hat- and blade-stiffened composite panels optimized with PASCO (refs. 1 and 2) is shown in figure 4 for comparison. Optimum designs for both the sandwich and the stiffened cover panels were determined using the baseline material properties and constraints. The composite sandwich cover panels have approximately the same structural efficiency as the composite stiffened cover panels when designed to identical constraints. This behavior is to be expected since the maximum-allowable strain and the inplane stiffness requirements are the active constraints for the optimum designs. These two constraints determine the amount of composite material required by both the sandwich and stiffened cover panels. Since the weight of the composite material constitutes the major component of the cover-panel weight, the structural efficiencies of both the sandwich and the stiffened cover panels are approximately equal.

Effect of Varying Constraints on Optimum Design

To assess the sensitivity of the structural efficiency of composite sandwich cover panels to changes in the constraints, new sets of optimum composite sandwich cover panels were designed while varying the constraints one at a time. By comparing these new cover-panel designs with the baseline designs, the effect of varying the constraints is identified.

Figure 5. Effect of maximum-allowable longitudinal strain on structural efficiency of composite sandwich cover panels.

Effect of varying maximum-allowable strain. The effect of varying the maximum-allowable strain constraint on the structural efficiency is shown in figure 5. This figure shows the structural efficiency W of optimized sandwich cover panels as a function of N_x for three values of the maximum-allowable strain $\varepsilon_{x,\max}$. For the baseline design, $\varepsilon_{x,\max} = 0.0045$ in/in. This maximum-allowable strain is an active constraint for N_x greater than 15 000 lb/in. Increasing the maximum-allowable strain to 0.006 in/in. yields significant improvements in the structural efficiency at load levels above 15 000 lb/in. Increasing the maximum-allowable strain beyond 0.006 in/in. yields little or no further improvements since $\varepsilon_{x,\max}$ is replaced by the minimum inplane stiffness requirements as one of the active constraints. If $\varepsilon_{x,\max}$ is decreased to 0.003 in/in., the maximum-allowable strain becomes the active constraint for load levels of 7500 lb/in. and above. The weight of sandwich cover panels designed with a maximum-allowable strain of 0.003 in/in. increases for load levels above 7500 lb/in. as compared with the baseline design.

Effect of varying minimum inplane stiffness requirements. The effect of varying the minimum inplane stiffness requirements on structural efficiency is shown in figure 6. This figure shows the structural efficiency W of optimized sandwich cover panels as a function of N_x for three values of the inplane stiffness requirements. In this figure, k_A is a scaling factor for the baseline values of $A_{11,\min}$ and $A_{66,\min}$. When $k_A = 1.0$, $A_{11,\min}$ and $A_{66,\min}$ are

Figure 6. Effect of minimum-required inplane stiffness on structural efficiency of composite sandwich cover panels.

the baseline values. When k_A has a value other than 1.0, the baseline values of $A_{11,\min}$ and $A_{66,\min}$ are multiplied by k_A at all load levels. Since the minimum inplane stiffness constraint is active for the baseline design at load levels below 15 000 lb/in., letting $k_A = 0.5$ reduces the weight of the cover panels at load levels below 15 000 lb/in. Further reductions in the minimum inplane stiffness requirements yield little or no further improvements since $\varepsilon_{x,\max}$ replaces $A_{11,\min}$ and $A_{66,\min}$ as one of the active constraints. Letting $k_A = 2.0$ increases the weight of the cover panels at all load levels considered. The minimum inplane stiffness requirements become an active constraint at all load levels, thus replacing $\varepsilon_{x,\max}$ as the active constraint at load levels above 15 000 lb/in. This is an important trend since the inplane stiffness requirements are likely to increase for newer technology transport aircraft with higher aspect ratio wings. For such a wing, stiffness may become a more important consideration than a higher $\varepsilon_{x,\max}$ for improved damage tolerance in the selection of appropriate materials for future transport aircraft.

Effect of varying upper percentage limit of all ply group thicknesses. The results of this study indicate that the upper limit on the percentage of 0° and ±45° plies (U_0 and U_{45}, respectively) is an active constraint at all load levels. The fact that this constraint is active indicates that the structural efficiency of these cover panels can be increased by allowing laminates with higher values of U_0 and U_{45}.

Figure 7. Effect of upper percentage limit of all ply orientations on structural efficiency of composite sandwich cover panels.

The structural efficiency W of optimized sandwich cover panels is shown in figure 7 as a function of N_x for two values of the upper percentage limits of all ply groups (U_0, U_{45}, and U_{90}). The upper curve in figure 7 is for the baseline value of this constraint ($U_0 = U_{45} = U_{90} = 0.375$), whereas the lower curve shows the effect of setting $U_0 = U_{45} = U_{90} = 1.0$. In both cases the lower percentage limits for all ply angles (L_0, L_{45}, and L_{90}) are equal to 0.125.

Figure 7 shows that the weight of all cover panels is reduced by allowing higher values of U_0, U_{45}, and U_{90}. For the loading case investigated ($N_{xy} = N_y = 0$, $N_x \neq 0$), the optimum percentage of 0° layers lies between 48 and 54 percent, whereas the optimum percentage of ±45° layers lies between 33 and 40 percent. The optimization procedure always drives the percentage of 90° layers to its minimum-allowable value, 12.5 percent in this case. The weight savings achieved by using higher values of U_0, U_{45}, and U_{90} indicate the importance of developing ways to understand and utilize laminates in which a high percentage of the plies are oriented in one direction.

Effect of varying core density. The optimum core density at all load levels is quite low, typically about 1.0 lb/ft^3. For reasons other than minimum weight, it may be preferable to use cores with a higher density. Thus, the effect of increasing the core density on the structural efficiency was investigated.

The results in figure 8 indicate the structural efficiency W of sandwich cover panels using cores of two different densities: $\rho_{\text{core}} = 1.0$ and 9.5 lb/ft^3.

Figure 8. Effect of core density ρ_{core} on structural efficiency of composite sandwich cover panels.

As can be seen from this figure, the weight of these sandwich cover panels is not very sensitive to changes in the core density; a ninefold increase in core density increases the weight by approximately 11 percent. There are two reasons for this behavior. First, the core is only a small percentage of the total weight of the sandwich cover panel; large differences in the core density have a small effect on the total weight. Second, as the core density is increased, so are its transverse shear stiffnesses G_{xz} and G_{yz}. Thus, the core thickness required to prevent global buckling from occurring is reduced. As can be seen from the data in table 4, the core thickness is reduced by up to 33 percent when the core density is increased from 1.0 to 9.5 lb/ft^3. Also note that the facesheet thickness does not vary as the core density is increased. Since the facesheet thickness t_f is mainly determined by the maximum strain and inplane stiffness constraints, changing the core density has no effect on t_f.

Since weight is relatively insensitive to changes in the core density, the selection of core density is probably best made based on criteria other than minimum weight, e.g., damage tolerance and ease of manufacture.

Concluding Remarks

An analysis and design method has been developed for the design of composite sandwich cover panels, including transverse shear effects and damage tolerance considerations. This method has been incorporated into a sandwich optimization computer program entitled SANDOP.

A set of optimized sandwich cover panels was designed with SANDOP with input values typical of those used for transport aircraft wing applications. Based on the designs generated by SANDOP, several observations can be made about the use of composite sandwich cover panels for transport aircraft wing applications. The composite sandwich cover panels considered in this study have approximately the same structural efficiency as composite stiffened-plate cover panels designed to identical constraints when the dominant design load is axial compression. Increasing the maximum-allowable strain from 0.0045 to 0.006 in/in. decreases the weight of composite sandwich cover panels at the higher load levels considered while having no effect on weight at the lower load levels. Increasing the maximum-allowable strain beyond 0.006 in/in. has little or no effect on the weight of the composite sandwich cover panels considered in this study. Decreasing the inplane stiffness requirements reduces the weight of composite sandwich cover panels at the lower load levels while having no effect on weight at the higher load levels. Increasing the inplane stiffness requirements induces a weight increase at all load levels. Increasing the upper limit on the percentage of 0° and ±45° plies of the facesheet laminate reduces the weight of composite sandwich cover panels. The weight of the sandwich cover-panel designs in this study is not very sensitive to changes in the core density. The core density

Table 4. Core and Facesheet Thicknesses

Thickness element	Values of N_x, lb/in., of—			
	3000	7500	15 000	24 000
Core thickness, in., for $\rho_{\text{core}} = 1.0$ lb/ft^3	0.30	0.44	0.61	0.63
Core thickness, in., for $\rho_{\text{core}} = 9.5$ lb/ft^3	0.28	0.40	0.52	0.42
Facesheet thickness, in., for $\rho_{\text{core}} = 1.0$ and 9.5 lb/ft^3	0.117	0.154	0.186	0.298

selection is probably best made on the basis of criteria other than those included in the present analysis (e.g., damage tolerance and ease of manufacture).

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November 7, 1991

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